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REPORT ZP-8-048
DATE 7 December 1956
MODEL F-102A

TITLE

STUDY
INSTALLATION OF AN INFRARED AUTOMATIC SEARCH AND POINT SYSTEM IN THE F-102A

PREPARED BY P. W. Snijders
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GROUP Preliminary Design

REFERENCE

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Chief, Pre-Design

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INTRODUCTION

The purpose of this report is to invite the attention of USAF personnel to the employment of the Avion Model 333 Infrared Automatic Search and Point System in the F-102A interceptor.

A study was made to determine possible locations for installation of the retractable detector head of the Model 333 ASP package, and the method of interconnection with the existing power supply sources in the F-102A airplane.

In addition, the possibility of using the control panel of the existing radar unit for the ASP system will be determined by further study.

Note: This is a preliminary study of the installation problems only. Further studies must be made to determine the desirability of incorporating the infrared system in the F-102A before a final report can be made.

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SUMMARY

This study shows that a retractable ASP package can be installed on the F-102A airplane with a minimum of structural changes. A minor reduction in speed during extensive operation will occur due to the aerodynamical interference effects.

The location in the lower nose assembly and in the upper nose assembly (Drawing SD-56-08084), allow the installation of a retractable ASP package with a square search pattern of 22°.

These locations cause no interference with the electronic compartments.

The weight increase for installation of the ASP system is approximately 55 lbs., including hydraulic actuators, doors, fittings, cables, switches, clamps, etc.

No aerodynamic feasibility tests have been made.

[redacted]

RECOMMENDATIONS

1. It is recommended that operational analysis studies be made to determine the performance improvements of the weapon system that can be obtained by the employment of the Avion Model 333 Automatic Search and Point Infrared System.

2. A simulated retractable ASP package should be installed in a fixed position on an F-102A to determine the aerodynamic interference effects during actual operation.

3. It is strongly recommended that the manufacturers of the Avion Model 333 ASP system in close cooperation with Convair Engineers study the following necessary improvements to make practical use of this system in high speed aircraft.

a. Design and install a light-shut-off valve in front of the sensitive photocell to make a fixed installation of the head possible.

b. Reduce present 5" diameter of the head of the ASP package to comply with the aerodynamical requirements for high speed aircraft.

c. Study the possibility of a 10 feet long attachment cable between the head and the electronics and power supply housed in the cylindrical container, which automatically increases the choice of separate locations for the head and for the cylindrical container.

[redacted]

DISCUSSION

The Avion Model 333 ASP is an Automatic Search and Point System, sensitive to near infrared radiation.

The associated electronics enable the optical system to search for targets through a square search pattern of up to 22°, either manually or automatically, and to track them either manually or automatically, as desired.

There are four distinct modes of operation.

1.1. Manual

In the Manual mode, ASP will indicate the presence of a target, but will not track it automatically.

2. Track

With the function switch in track position, ASP, if it sees a moving target, will track it automatically.

3. Search

With the function switch in search position, ASP will search for a target through a square pattern up to 22°.

4. SALO (Search Automatic Lock-On)

With the function switch in SALO position, ASP will behave exactly as in Search, in the absence of a target. When a target is observed, however, ASP acquires and tracks it automatically. SALO is thus a combined search-and-track mode.

[redacted]

The unit consists of a head, a cylindrical container with the electronics and power supply, and a control panel, which is attached to the electronics through a cable.

The Head is a short cylinder 5" in diameter and 4" long, with a dome of special Dow-Corning 0160 glass which passes IR (Infra-Red) within the operating passband. It contains the optical system and the photo cell.

The electronics and the power supply are housed in an aluminum cylinder 5" in diameter and 24" long, which may be attached to the Head directly or through a short cable.

The control panel contains all switches and controls plus a cross pointer bearing meter which at all times shows the relative position of the gyro with respect to the Head.

The ASP system draws 35 watts from a 28 volt DC source, and 45 watts from a 115 volt, single-phase source of precision (± 5%) 400 cycle current.

On the control panel is a DC outlet, while a cable is attached to the 400-cycle source.

The choice of location and the installation of the ASP package are complicated by the following requirements:

1. For proper operation, there should be no magnetic materials in the vicinity of the Head and no metal at all for a distance of 4" around the Head.

2. Strong targets like direct sun rays, search light, nuclear weapons, etc., should not come within the field of view in order to prevent damaging the sensitive photo-cell.

3. The square search pattern of up to 22° is only possible if a location is chosen which allows a view, unobstructed by the aircraft structure.

[redacted]

4. To obtain the highest accuracy in performance of the ASP system, vibration of the ASP package by aerodynamic wave effects must be avoided.

5. The center of the square search pattern of the ASP system must be boresighted with the center of the search pattern of the radar system of the F-102A.

Drawing SD-56-08081 - Installation Avion Model 333 ASP

3 View - F-102A - Various Locations (not recommended)

Pos. 1 shows the location of the retractable ASP package on top or in the leading edge of the wing.

Due to the presence of the fuel tanks the installation of the ASP package in the leading edge of the wing is complicated.

Installing the package on top or under the wing will affect the flight characteristics of the F-102A.

Pos. 2 shows the location of the retractable ASP package in the tail structure. Disadvantages are the unstability of the top of the tail structure, which causes inaccuracies of the Search and Point system, and the extreme distance from the power-source.

Pos. 3 shows the location of the retractable ASP package in the inlet duct structure. In this location the required square search pattern of up to 22° is not possible, caused by interference of the aircraft structure in this search pattern.

A complicated inlet duct structure re-design would be required to utilize this location.

[redacted]

Pos. 4 shows the location of the retractable detector Head of the ASP package on top of the canopy and the location of the cylindrical container behind the canopy of the F-102A.

Due to the high bow wave effect at this location of the retractable Head this location is not recommended.

Drawing SD-56-08084 - Installation Avion Model 333 ASP View Inboard - F-102A (recommended locations)

Pos. 5 shows the location of the retractable ASP package in the upper panel of the nose section of the F-102A. A fairing on top of the nose is shown.

Pos. 5A shows the retractable detector Head with fairing of the ASP package at the same location. The cylindrical container is located separately in the upper panel assembly. The power supply cable runs from the cylindrical container through the legs of the retract-structure to the Head.

Note: The location mentioned in Pos. 5 and Pos. 5A is recommended for installation of a retractable inflight refueling probe as described in Convair, San Diego Report No. ZP-8-046, titled "A report on Inflight Refueling Potentialities with the F-102A Interceptor as a Receiver Airplane."

Technically it is possible to install both the retractable inflight refueling probe and the retractable Head of the ASP package in the upper panel assembly of the nose section of the F-102A.

With the exception of two small wells in the top shelf of the electronic compartment there is no interference with the electronic packages layout.

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Pos. 6 shows the retractable ASP package located in the lower panel assembly of the nose section of the F-102A.

At the same location a retractable Head with fairing can be installed leaving ample space to locate the cylindrical container separately.

However, after tracking the target the ASP package has to be retracted before a missile is fired.

A coil spring is built in the retract-system as a safety feature in the event of hydraulic system failure.

Drawing SD-56-08094 - Installation Avion Model 333 ASP Leading Edge Wing F-102A

This drawing shows the location of a fixed Detector Head in the leading edge of the wing of the F-102A airplane.

This location is only feasible if the design of the Avion Model 333 ASP Infrared System is improved as described in this study-report on Page 3, Recommendations 3 a.b.c.

The drawing shows a 3½" and a 5" diameter for the Detector Head. A minimum diameter of 7" is required in the event, that a light-shut-off valve or cover has to be designed outwards of the Detector Head.

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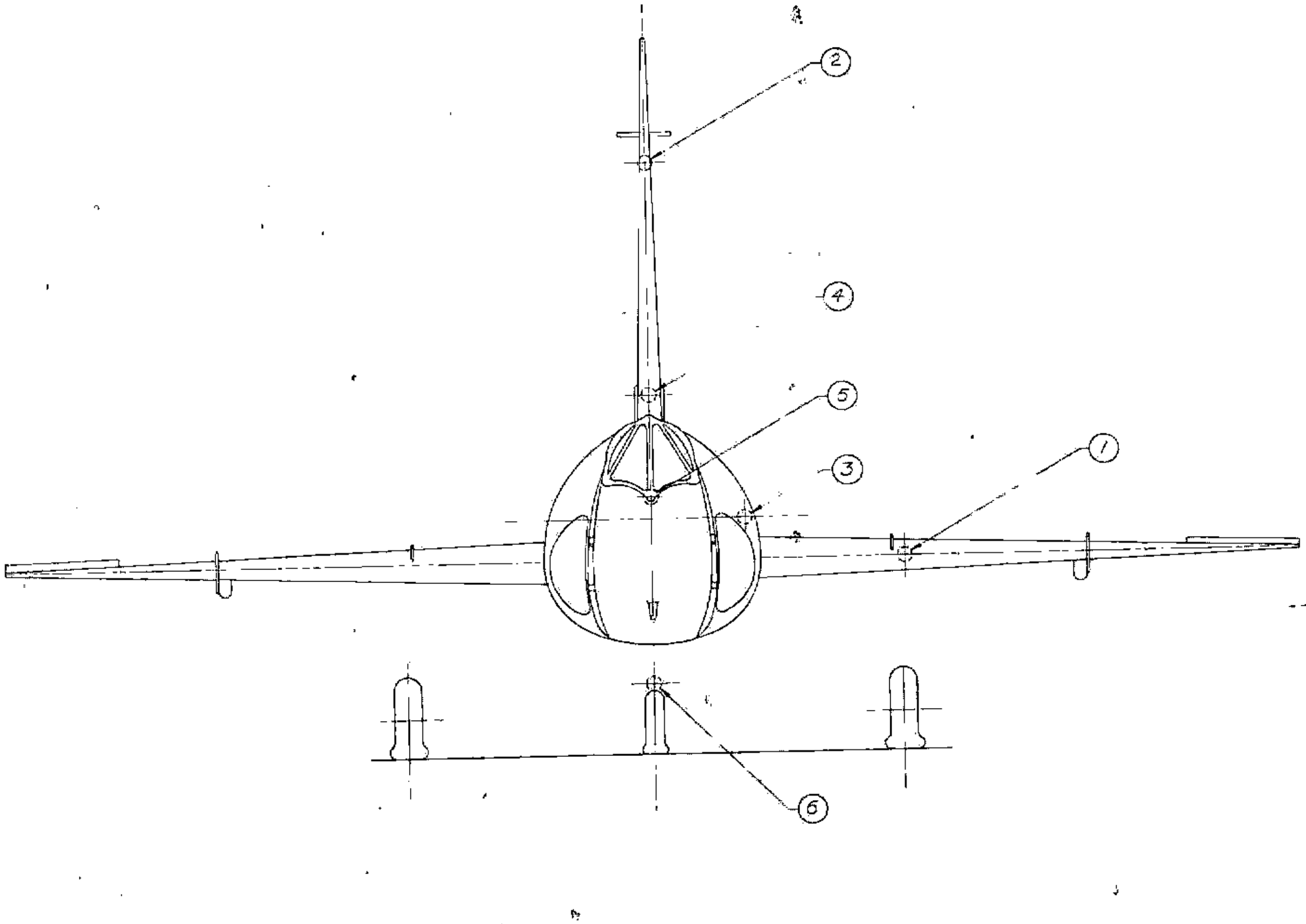

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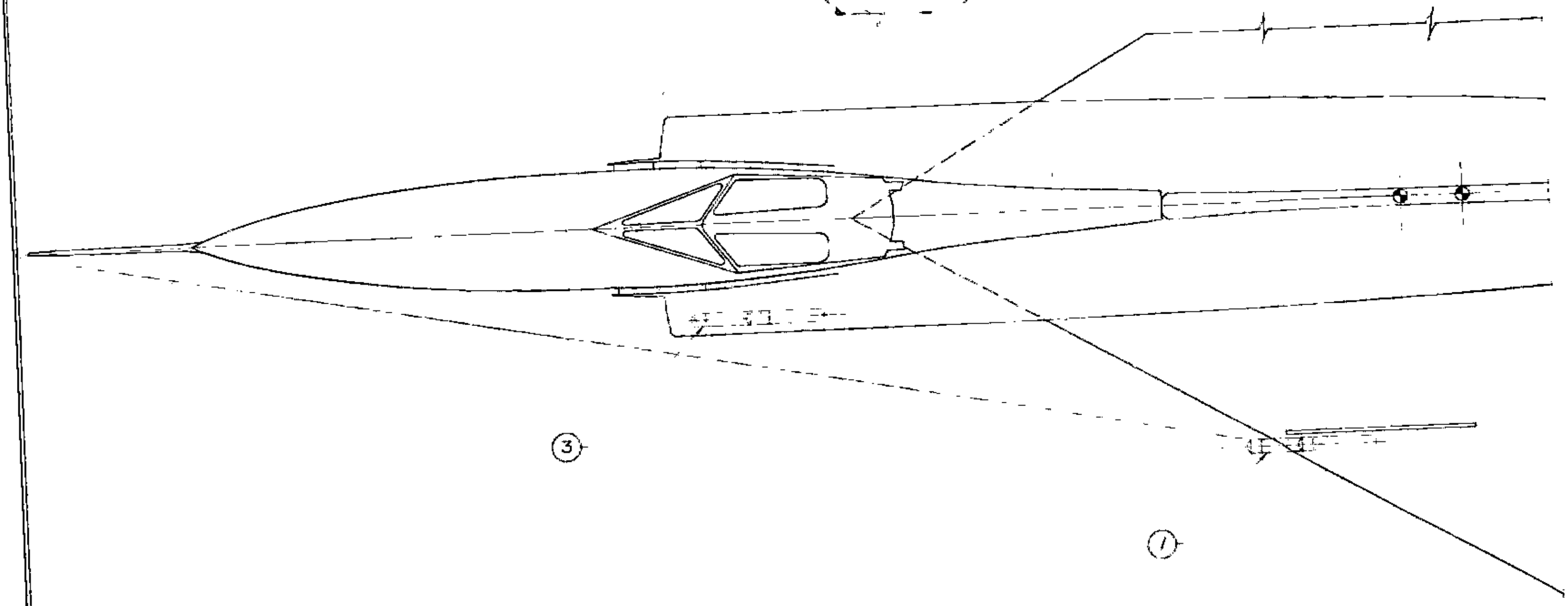

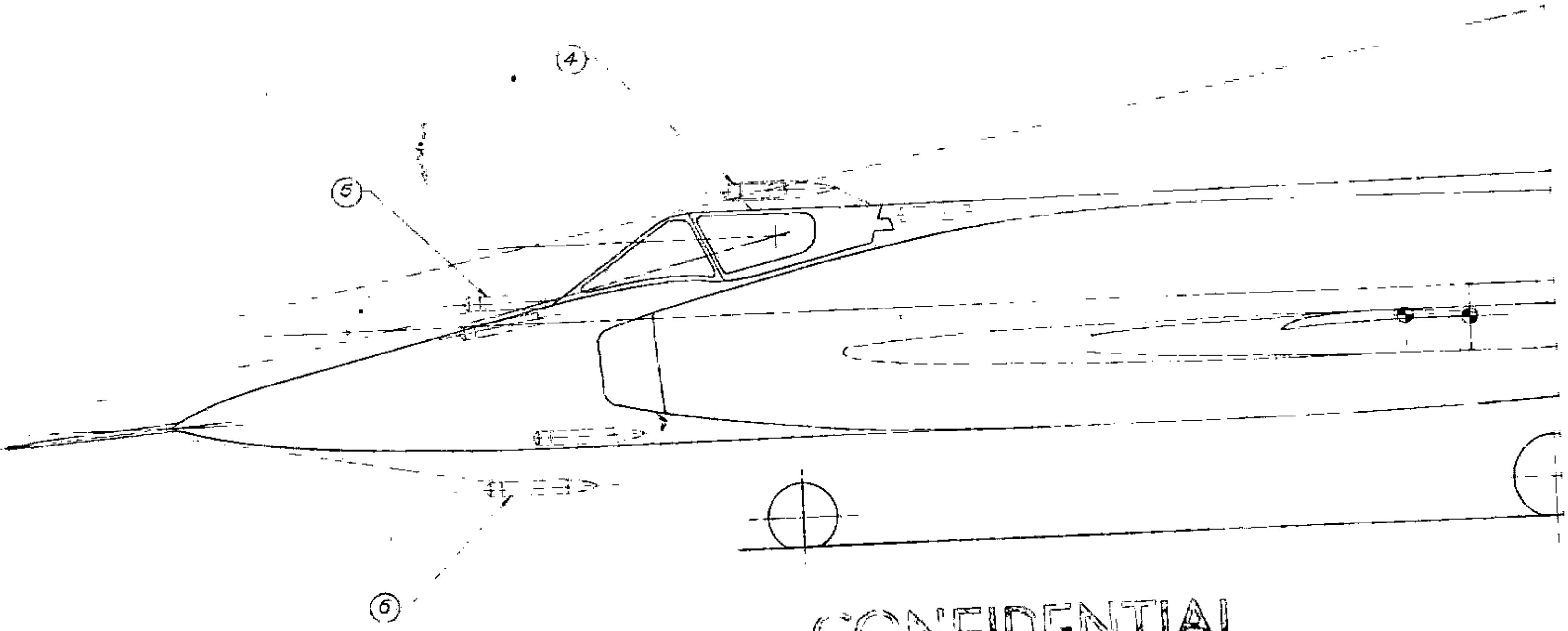

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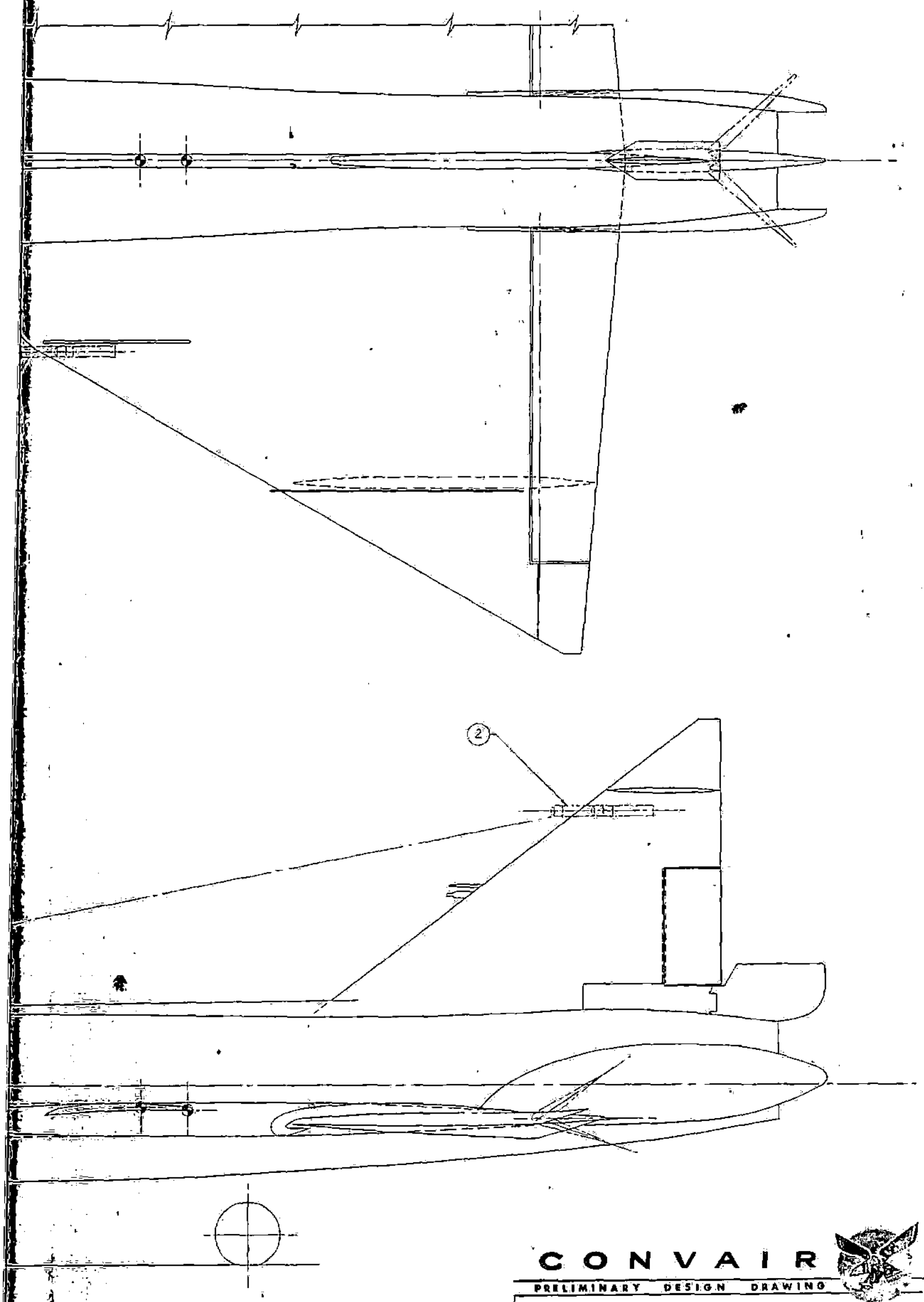

C O N V A I R

PRELIMINARY DESIGN DRAWING

INSTALLATION AVION MODEL 333 ASP
VARIOUS LOCATIONS F-102 A

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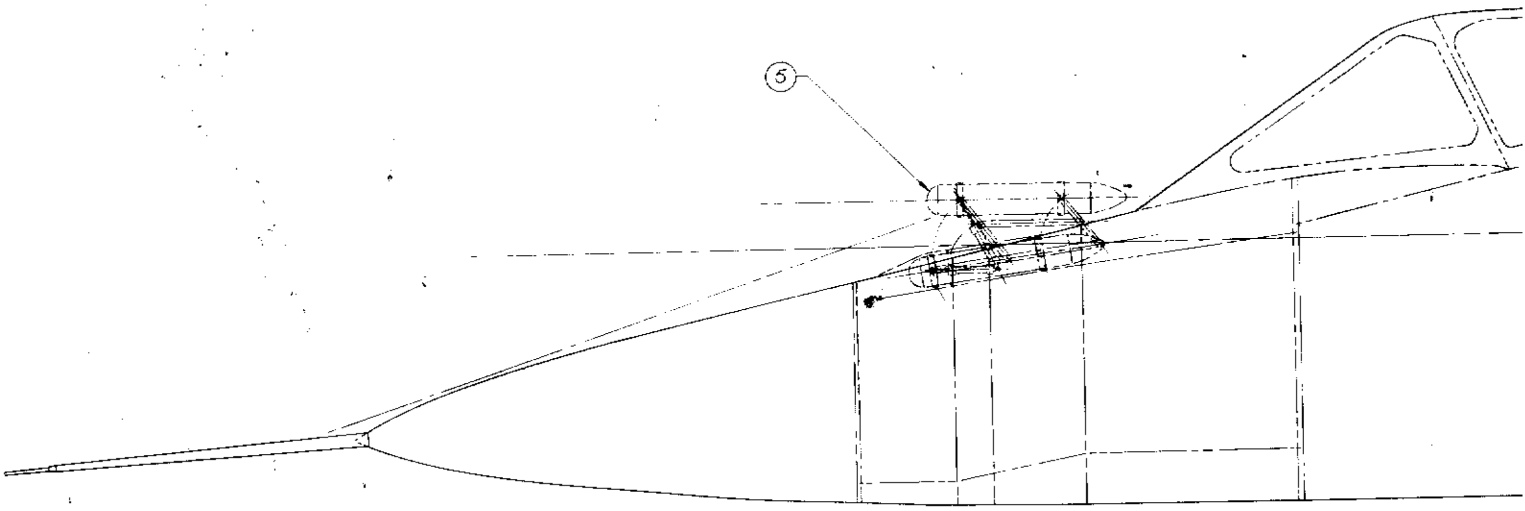
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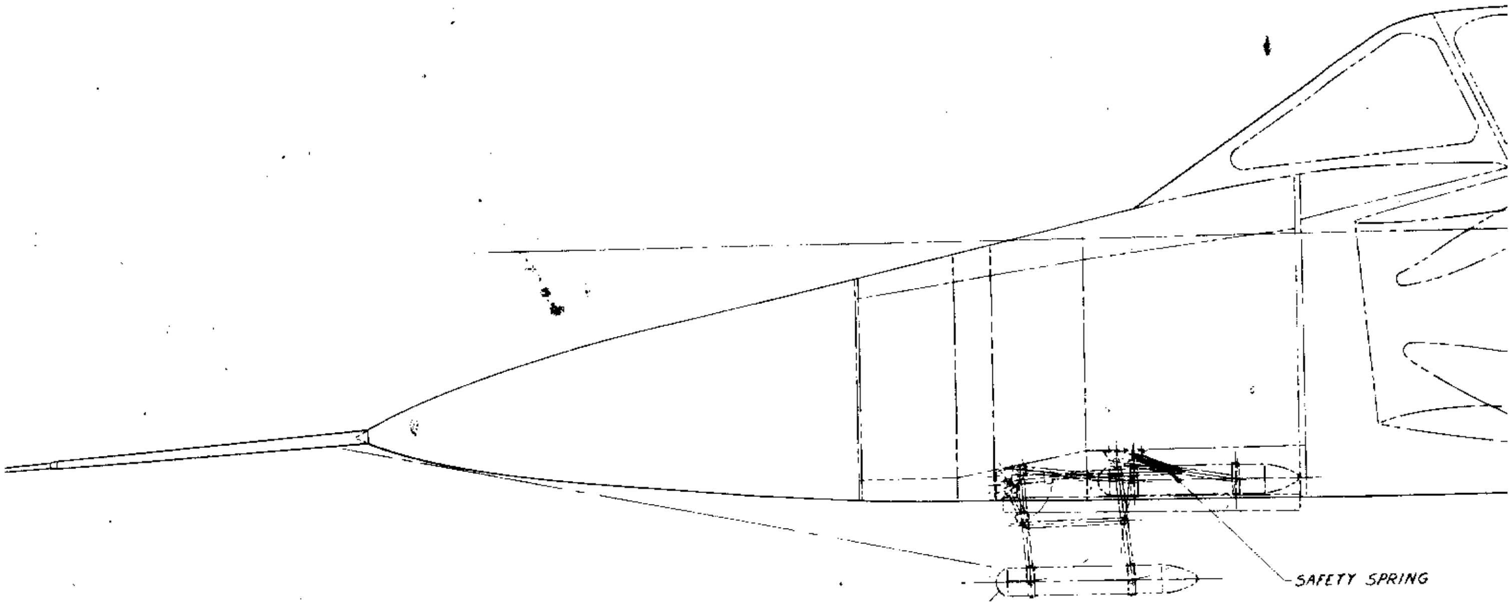
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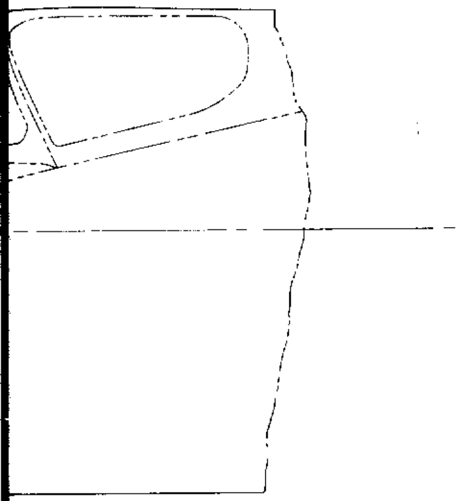

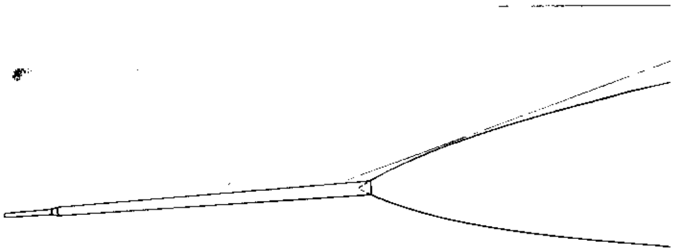

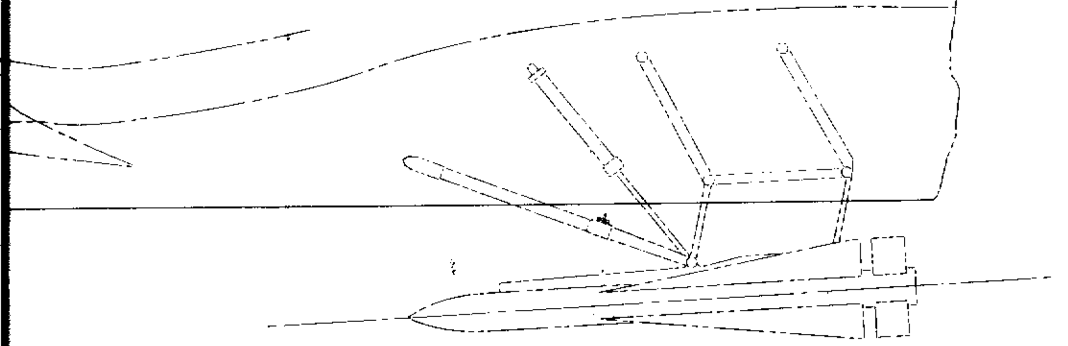

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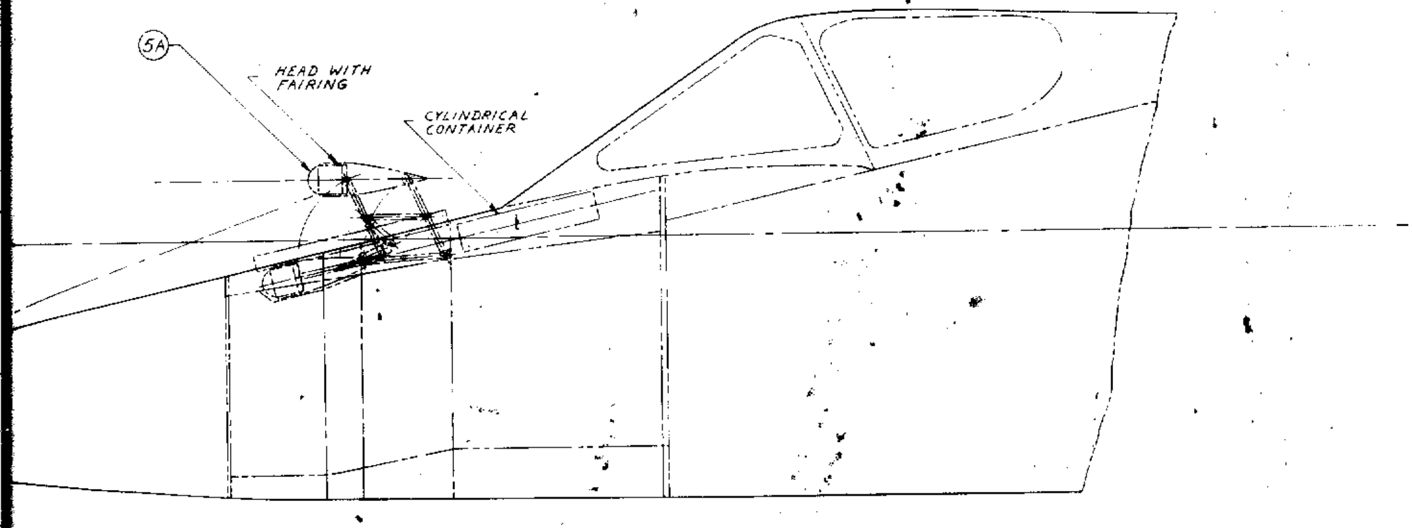

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PRELIMINARY DESIGN DRAWING

INSTALLATION AVION MODEL 333 ASP
VIEW INBOARD F-102 A

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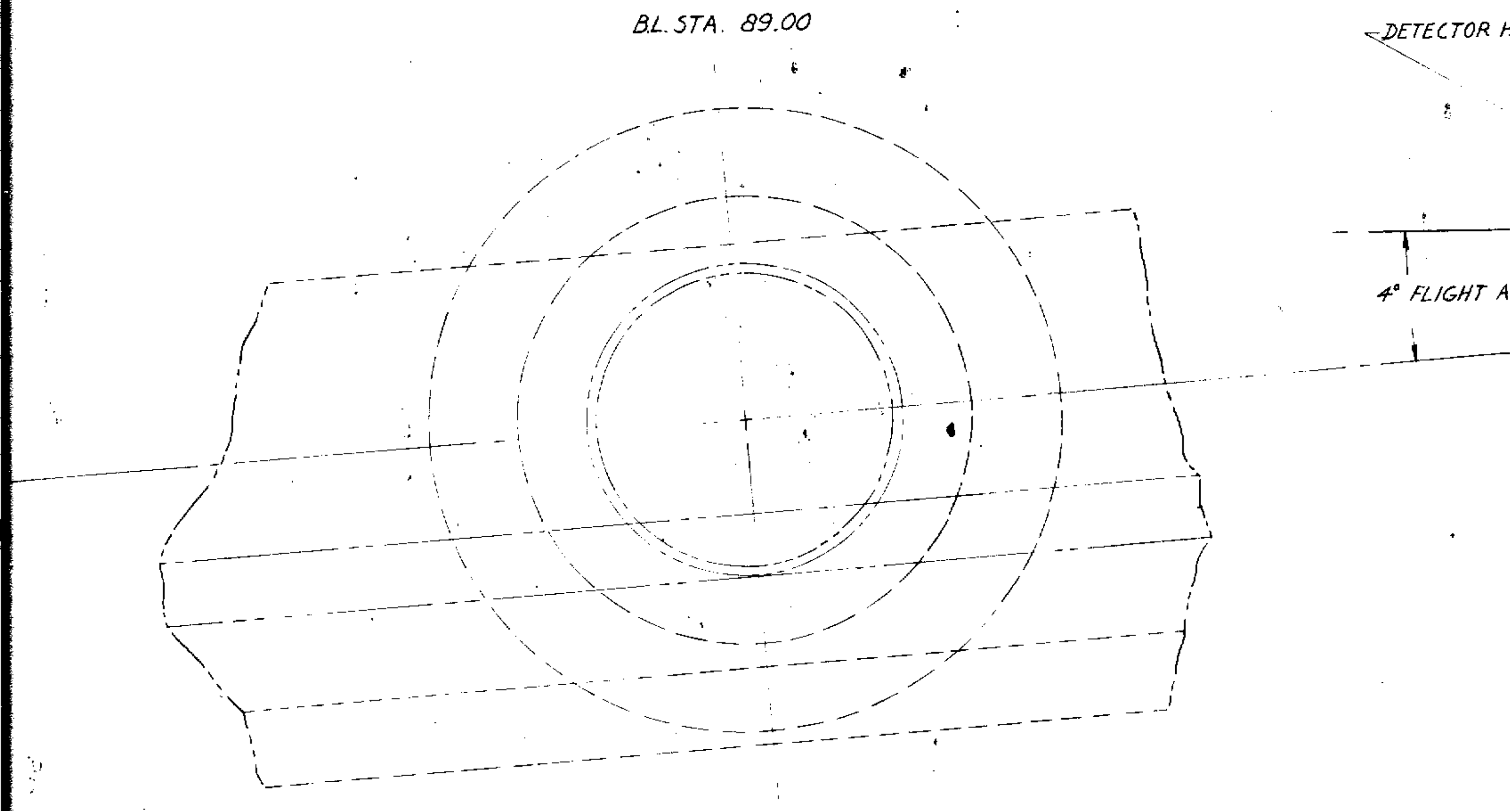

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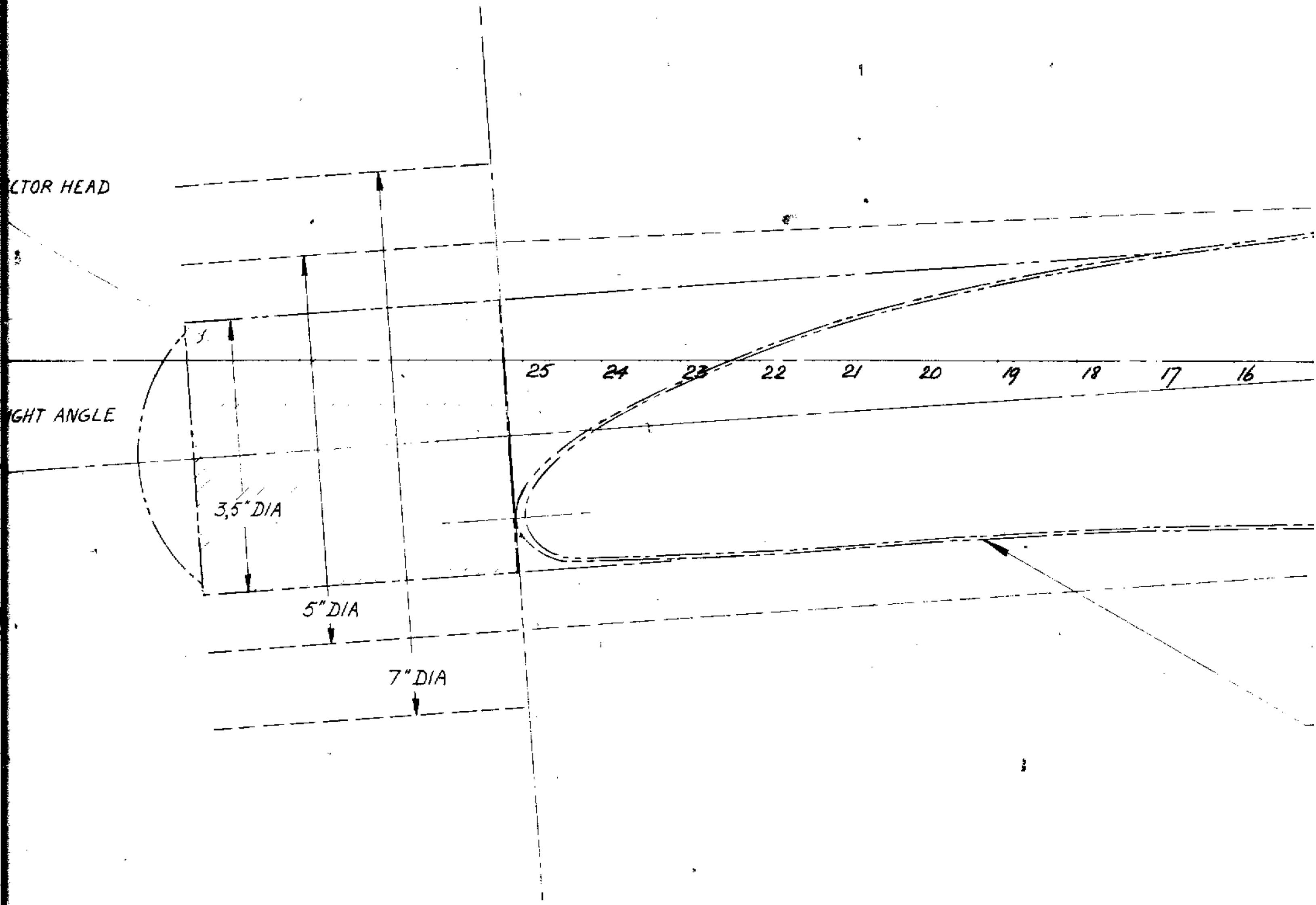

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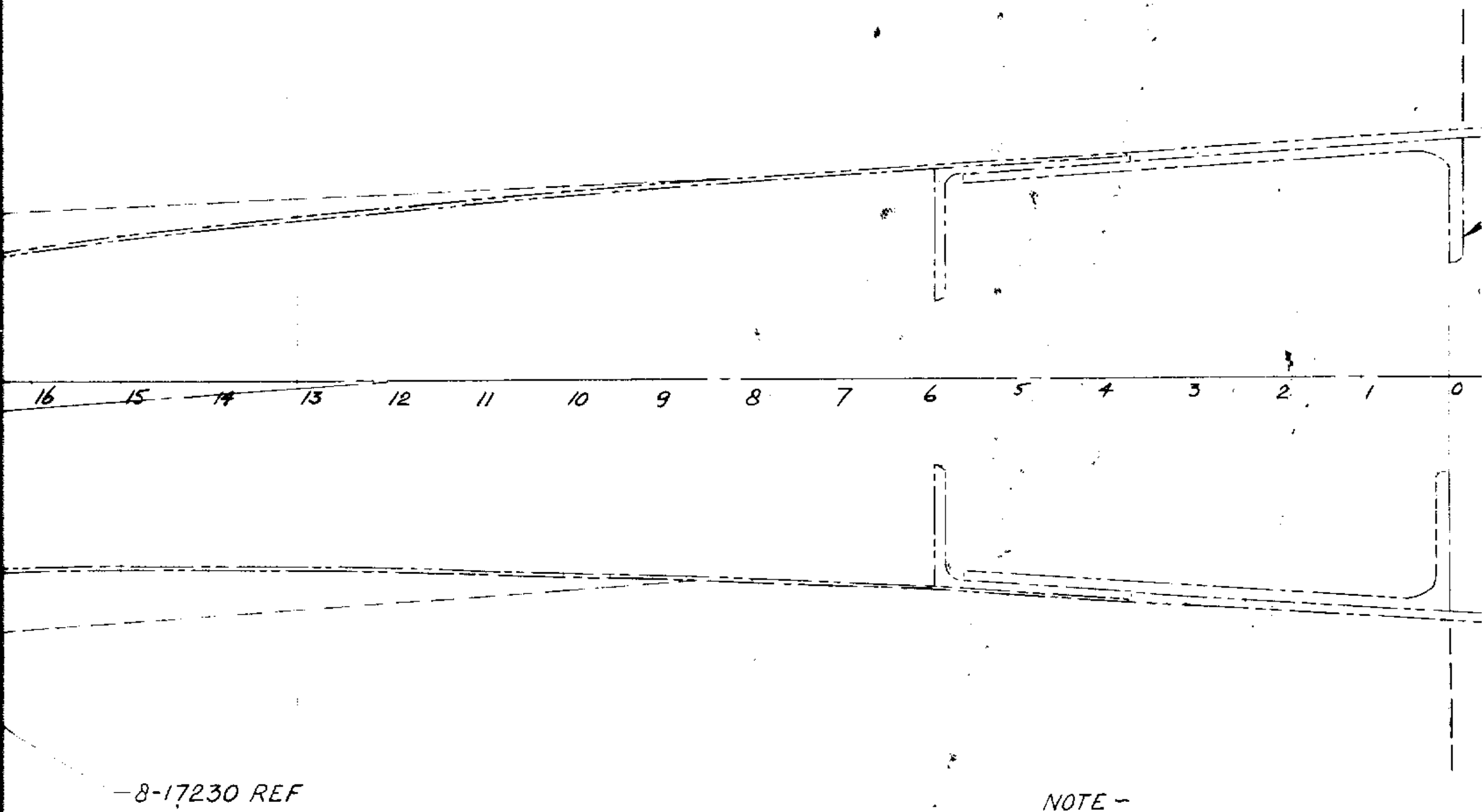

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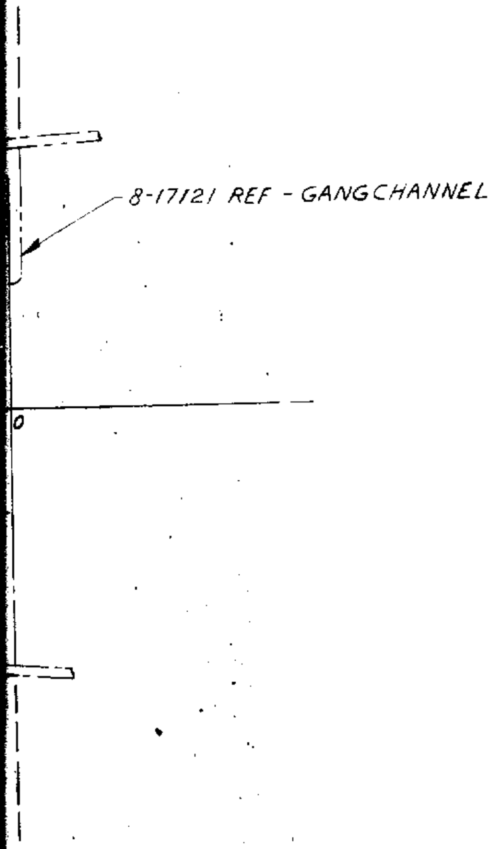

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VAIR

DESIGN DRAWING

AVION MODEL 333 ASP
EDGE WING F-102A

	APPROVED	SCALE FULL	DATE 12-5-1956

IEGO, CALIFORNIA
DYNAMICS CORPORATION
56-08094

GDC 66-062

GD GENERAL DYNAMICS | CONVAIR

REPORT GDC-66-062
DATE 10 June 1966
MODEL F-106B

TITLE

F-106B NASA SST
TEST BED STUDY
(Phase II)

Prepared under NASA Contract
NAS 3-7915, Task Order No. 1

PREPARED BY Collette
J. G. R. Collette

GROUP DYNAMICS

CHECKED BY R. E. Martin
R. E. Martin

REFERENCE

CHECKED BY R L Fefferman
R. L. Fefferman

APPROVED BY
C. S. Brandt
Program Manager
F-106B NASA SST Test Bed

NO. OF PAGES 44
NO. OF DIAGRAMS 23
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INTRODUCTION

A study was conducted to examine the feasibility of using an F-106B airplane as a flight test bed for two nacelles containing J-85-13 engines. The close, underwing installation requires a 26-inch section of the elevon surface to be removed from each side to allow for mounting of the nacelles.

The work consisted of evaluating the effects of the nacelle installation and the resulting loss of control effectiveness, on the performance, stability and control of the airplane. The study makes the assumption that torsional rigidity is retained such that aeroelastic losses are proportional to the basic airplane. It further assumes that the yaw damper and turn coordinator meets the requirements of the new configuration.

SUMMARY

With two possible exceptions, the Phase II results indicate that the test missions can be carried out with no modifications other than those proposed. Hot day operation at 40,000 feet in the low supersonic region represents a marginal condition which may necessitate special operating techniques to obtain certain test points. The longitudinal characteristics exhibited by the airplane fall somewhat below the handling qualities criterion that was chosen for stability-augmented operation at the higher test altitudes. The unmodified damper does provide improvement to the test bed airplane inherent characteristics; the only question is one of acceptability of the available damping for carrying out the missions successfully.

1. WEIGHT AND C.G. MANAGEMENT

The takeoff gross weight and c.g. range is dependent upon the amount of fuel required to accomplish the test mission as well as the fuel sequencing to be used. Two missions were examined using 3 different fuel sequencing concepts. It soon became apparent that due to the fuel required by the J-75, it would be necessary to make some of the missile bay fuel available to the J-75 engine.

The first sequence examined early in the program was one in which the missile bay tanks were integrated into the existing system (see Figure 1-3). This provided the basis for the weight-inertia parameters for the stability study and performance analysis. It is not recommended, however, since it requires too much ballast.

Subsequently, another scheme was considered wherein the missile bay tanks were left separate from the basic airplane fuel system and used primarily for feeding the J-85 test engines. With separate fuel systems, a number of highly undesirable conditions exist which would unduly complicate the test program (see Figure 1-2). For example the center of gravity travel becomes different each time there is a change in time scheduling of the J-85 use (for example a Mach 1.2 test vs. a Mach 2.0 test) and the same for any variation from that plan (such as delays in afterburner light-off or engine start, flame-outs, etc.). The c.g. becomes unpredictable in these events and the amount of ballast required to insure a stable airplane in all cases would create forward c.g. problems at other points. The aforesaid conditions will exist regardless of how the J-75 fuel system is sequenced. The concept of providing manual transfer of fuel by the pilot in the event of an emergency is also impractical since he would not know where the c.g. was at the time of an emergency and would therefore risk the likelihood of creating a too far forward c.g. while attempting to avoid an aft c.g., or vice versa.

In order to provide a simple system, the scheme shown in Figure 1-1 was derived and is recommended for adoption. This system makes use of the F-106B fuel sequence intact, (except that the "T" tank is eliminated in all cases) the only difference being that the missile bay tanks are placed between the existing system and the engines, with all engines feeding from a common manifold line. This system will provide a c.g. travel which can be guaranteed between the existing airplane c.g. limits at all times regardless of mission or emergencies, and will require the least amount of ballast and therefore the lightest weight. The proposed system would then operate as follows:

1.	48 gallons from tank 1	Existing F-106B
2.	All of tank "F"	
3.	Remainder of tank 1	
4.	All of tank 2	
5.	All of tank 3	
6.	Forward missile bay	
7.	Aft missile bay fuel last.	

There is also a potential of removing all the ballast from the nose, if subsequent investigations should show that reducing load factors to 5 would relieve that portion of the aft c.g. limit which is set by structural considerations. If this should turn out to be possible, the ramp weight would be reduced to approximately 40,450 pounds with a takeoff c.g. of approximately 28%.

2. AERODYNAMIC DATA

Revision of the aerodynamic coefficients was accomplished primarily to support studies for determining whether or not the existing damper and turn coordinator systems will provide satisfactory airplane flight characteristics, and for investigating airplane response to thrust chops. Basic airplane aerodynamic coefficients, including elastic effects, were obtained from Reference 1.

2.1 CONTROL DERIVATIVES

Effect of the elevon cut-out on rigid airplane control derivatives was obtained by determining a reduced control effectiveness parameter, α_δ, from a collection of test data. Subsonic values for α_δ are shown on Figure 2-1 as a function of control surface area ratio. Also shown on Figure 2-1 is the variation of α_δ (with the elevon cut-out) with Mach number. It was assumed that Mach number effects on α_δ remained the same as for the basic airplane. Revised elevon control derivatives are given, with elastic losses included, on Figures 2-2 through 2-6 ($C_{L_{\delta_e}}$, $C_{m_{\delta_e}}$, $C_{y_{\delta_a}}$, $C_{\ell_{\delta_a}}$, $C_{n_{\delta_a}}$). The elevon control derivatives have been reduced by the ratio of α_δ with and without the elevon cut-out, and elastic attenuation factors are the same as for the basic airplane.

2.2 STABILITY DERIVATIVES

Subsonic lift and pitching moments were estimated from Reference 2, and the data from Reference 3 were used as a guide for determining Mach number effects. The nacelle lift curve slope and pitching moment (for c.g. at $.305\bar{c}$) increments are shown as a function of Mach number on Figure 2-7. Effect of the test nacelles on static longitudinal stability, $C_{m_{C_L}}$, for the elastic airplane is shown on Figure 2-8 where the magnitude of $C_{m_{C_L}}$ indicates that the test nacelles should have an insignificant effect on the airplane's static longitudinal stability. Since the test nacelles are axisymmetric, side force, C_{y_β}, was assumed to be equal to lift force, C_{L_α}. The derivatives C_{n_β} and C_{ℓ_β} were obtained by multiplying C_{y_β} by the appropriate moment arms. Increments in the lateral-directional static stability derivatives due to the test nacelles are shown on Figure 2-9. Comparison with the values of C_{y_β}, C_{n_β} and C_{ℓ_β} for the basic airplane indicates that the test engines should also have an insignificant effect on static lateral-directional stability.

2.3 TEST ENGINE DRAG

External drag increment of the test nacelles was estimated, assuming the engines to be operating with no additive drag due to inlet spillage. Wave, skin friction and subsonic pressure drag were obtained from Reference 4. Total external drag for the two test nacelles is shown on Figure 2-10. Drag due to a dead test engine was approximated from windmilling engine drag for the Convair 990 (Reference 5) converted to the test engine nacelle cross section area and F-106B reference wing area. Additive drag, assuming a mass flow ratio of zero, was estimated from Reference 6. Total engine-out drag increment (per engine) is shown on Figure 2-11.

3. PERFORMANCE

3.1 TAKEOFF AND LANDING

Since the test engines decrease airplane ground clearance angle, a rough check of takeoff and landing speeds and distances was conducted to determine performance degradation. Examination of F-106A wind tunnel data in Reference 5 showed that the elevon cut-out would not cause the airplane to be elevator limited during takeoff and landing. Performance for the basic F-106B airplane is determined from Reference 7. Trimmed lift coefficients at maximum ground clearance angle, as a function of center of gravity location, are shown on Figure 3-1 for the basic and test airplanes. Test airplane data on Figure 3-1 were estimated by determining ratios of trimmed C_L for the basic airplane at minimum flight test demonstrated takeoff and landing speeds (from Reference 7) to the trimmed C_L at a ground clearance angle of 16 deg. (from wind tunnel data in Reference 6). These ratios were applied to wind tunnel data at a ground clearance angle of 13 degrees to obtain test airplane trimmed C_L curves. Using the takeoff and landing charts in Appendix B of Reference 7 and trimmed C_L from Figure 3-1 typical takeoff and landing performance was computed for the test airplane and compared with a basic airplane at an equivalent gross weight and center of gravity location. Results are tabulated below:

TAKE OFF (MAX A/B POWER)						
	Weight (lbs)	c.g. (% Mac)	Trimmed C_L	Takeoff Speed (Kts)	Ground Roll (ft)	Distance Over 50 ft.Obstacle (ft.)
Test Airplane	39,200	25.9	.40	204	4600	6700
Basic Airplane	39,200	25.9	.51	181	3700	5600

LANDING						
	Weight (lbs)	c.g. (%Mac)	Trimmed C_L	Touch Down Speed (Kts)	Ground Roll (ft)	Distance Over 50 ft.Obstacle (ft.)
Test Airplane	30,334	30.0	.50	161	4500	5950
Basic Airplane	30,334	30.0	.64	142	3100	4500

NOTE:

Landing distance is for speed brakes extended and drag chute deployed.

3.2 MISSION PROFILES

Fuel required to accomplish the two most critical test missions was determined. Description of these missions is as follows:

A. For Both Missions
   1. Climb to 25,000 ft. with J-75 in A/B power, J-85 off.
   2. Light J-85 and climb to 40,000 ft. with J-75 in A/B power, J-85 at MIL power.

B. Mission #1
   1. Accelerate to M = 1.2, J-85 and J-75 in A/B power.
   2. Cruise at M = 1.2 for 15 min, J-85 in A/B power, J-75 as required to maintain M = 1.2.

C. Mission #2
   1. Accelerate to M = 2.0, J-85 and J-75 in A/B power.
   2. Cruise at M = 2.0 until depletion of fuel, J-85 in A/B power, J-75 as required to maintain M = 2.0.

A summary of the fuel required to accomplish these missions is presented on Table 3-1,based on a ramp weight of 40,993 lbs. Mission #1 can be accomplished with reserve fuel of 3279 lbs. for the J-75 and 1562 lbs. for the J-85. For Mission #2, 3.8 min. cruise at M = 2.0 is available if separate J-75 and J-85 fuel systems are used, and 6.7 min. is available if the fuel systems are manifolded. It should be noted, however, that minimum speed of the basic airplane at test gross weights and 40,000 ft. is about 1.47 (J-75 minimum A/B power), while maximum speed with MIL power is about M = .98. A check was therefore made to determine whether or not the M = 1.2 cruise portion of Mission #1 can be accomplished. Results are shown on Figure 3-2 where thrust required, at 40,000 ft. is shown for the basic airplane at W = 36,000 lbs. and for the basic airplane plus external drag of the test engines. Thrust available with J-85 at max. A/B power and J-75 at MIL power is also shown. It is seen that M = 1.2 can be sustained with this power on a standard day, but hot-day operations might necessitate special operating techniques, such as operation at lower altitudes or deceleration through the desired test speeds.

4. MANEUVERING CAPABILITIES

4.1 LOAD FACTOR CAPABILITY

A brief examination of the effect of the elevon cut-out on load factor capabilities was accomplished. A check of basic airplane hinge moment coefficients showed that the elevon cut-out had a negligible effect so that test airplane hinge moments would be changed only by the reduction in elevon area. Since the maximum hinge moment available from the hydraulic actuators is unchanged, maneuvering capability will be degraded only in the regions where the airplane is elevator limited. Load factor capability for the test airplane is shown on Figure 4-1 compared with the basic airplane.

4.2 ROLLING PERFORMANCE

At this stage of the analysis, possible restrictions on allowable aileron deflection (such as those imposed with external tanks) have not been established. Therefore, no quantitative roll performance will be presented. However, the effect of the elevon cut-out is to reduce roll rate by about 12% at maximum aileron delfection. Rolling performance of the basic F-106B (from Air Force Category II tests) is given in Reference 8.

5. ENGINE FAILURE MODE

Addition of outboard engines to the F-106B airplane raises two questions relative to the loss of thrust from one outboard engine: Is there adequate control power to maintain control of the airplane? Are the resulting loads, particularly on the vertical tail, within the design limits?

To evaluate the dynamics of the engine-out mode, a digital computer program was written, incorporating the equations of lateral-directional motion (Eq. 1-3) and of the rudder (Eq. 4-11) and aileron (Eq. 12) control systems. The equations are given in Figure 5-1. Hinge moment limits (although never reached) were applied in the rudder case but not for the aileron considering that the elevon actuators are several times more powerful than the rudder actuator.

For simplicity, the equations of longitudinal motion were deleted. The assumption of constant speed does not degrade the usefulness of the results, considering that the significant events occur within a few seconds following engine failure. Indeed, the equations of motion could have been further simplified with no significant effect.

Three flight conditions were investigated:

(1) M = 2.0 at 30,000 feet, GW = 37,000 lbs.

(2) M = 1.2 at 55,000 feet, GW = 34,000 lbs.

(3) M = 2.0 at 55,000 feet, GW = 34,000 lbs.

The first condition represents maximum dynamic pressure, the others were selected to investigate controllability in the low-damping regime.

Certain assumptions were made relative to the nominal engine failure simulation:

a. The right-hand engine fails,

b. The afterburner thrust decays linearly to zero in 0.2 seconds,

c. The basic (J-85) engine thrust decays exponentially with a time constant of 1.5 sec. (based on experimental data for the JT3D engine),

d. The lift increment resulting from engine thrust loss had a maximum value of 1600 lbs. at condition (1) above, and varied directly with q_c.

e. The pilot's corrective action lags behind the start of engine failure.

These assumptions were, on occasion, varied such as to aggravate the problems of controllability or vertical tail load. Deactivation of the yaw damper-turn coordinator was examined, as was the omission of pilot-applied rudder.

Table 5-1 summarizes the significant parameters for the computer solutions, together with the peak values of roll angle, sideslip angle and vertical tail load.

Condition (1) results indicate a nominal maximum load on the vertical tail of 5000 lbs. whether or not the pilot applies corrective rudder. In the latter case, the turn coordinator introduces -0.35 deg. of steady state rudder, holding the airplane in -0.6 deg. of sideslip. This corresponds to a steady-state tail load of approximately 4500 lbs.

For case (1c) the engine thrust decay time constant, τ_e, was reduced to 0.1 sec. and the pilot-applied rudder was delayed 2.0 sec. The result of this very conservative simulation is shown in Figure 5-2. The peak vertical tail load of 7450 lbs. is well below the design limit of 24,000 lbs. The simulation did not include any reduction of rudder pedal force; therefore a meaningful steady-state condition is not achieved. It is demonstrated that there is sufficient rudder power to reduce the sideslip angle to zero with sufficient margin to reverse it to 0.3 deg. or more. Roll control constitutes no problem, with the bank angle never exceeding five degrees. The momentary application of full aileron (7°) results from the choice of gain factors in equation (14) and is not necessarily typical of a pilot's response. The momentary tail load due to aileron deflection at t = 1.5 seconds would, if applied at t = 0.6 seconds, increase the peak load on the tail. Nevertheless, the total would still be far below the design limit. Comparing cases (1a) and (1d), one can see the load effect of applying corrective aileron with less delay.

Application of the q_c factor to the thrust-induced lift (assumption d, above) gives a lift increment of 150 lbs. for M = 1.2 at 55,000 feet. For conservatism, this was increased to 200 lbs. for case (2a) and to 400 lbs. for cases (2b), (2c) and (2d). As expected, the tail loads were low. Neither were there any problems of controllability. In every case the sideslip was reversed despite an inadvertent restriction of rudder deflection to the limits of condition (1). This resulted from the failure to change the value of K_p, Equation (7), with the change in q_c.

Condition (3) is represented by cases (3a) and (3b), applying asymmetric lift of 500 lbs. and 1000 lbs., respectively. No problems with loads or control are evident. The same comments as above apply relative to sideslip and available rudder deflection.

In summary, the addition of J-85 engines under the wings of the F-106B does not cause any problems with either the tail loads or the controllability associated with engine failure.

6. DYNAMIC STABILITY

6.1 LONGITUDINAL

The modifications proposed for the F-106B have little effect on the inherent longitudinal stability of the airplane.

The phugoid mode presents no problem on the USAF airplane, and no attempt was made to analyze the mode formally. The increase in drag on the test bed is the only significant parameter affecting this mode and its effect should be to improve the damping.

The variations in the stability parameters of the short period mode, caused by the a.c. shift resulting from the engine installation, are negligible. The effect of higher gross weight is more pronounced, yielding lower frequencies and damping, as expected. Overall, the characteristics exhibited by the modified F-106B without the damper system still provide positive damping with relatively small changes in period throughout the flight regime. Figure 6-1 shows a comparison of the characteristics exhibited by both airplanes.

Stability Augmentation

Augmentation was examined using the present damper system with nominal gains and time constants together with the 2 degree-of-freedom short period equations. The study included a cross-section of gross weights and c.g's sufficient to cover the proposed c.g. control pattern described in Figure 1-3. The conclusions reached indicate trends which can be generalized to some extent and adapted to other c.g. patterns. The altitude Mach envelope covered is described in the following table.

Altitude	Mach Range
S.L.	0.25 to 1.0
15 M	0.6 to 1.2
30 M	0.8 to 1.5
45 M	0.8 to 2.0
55 M	0.9 to 2.0

With periods expected to range from approximately 1 second to somewhat less than 5 seconds, the damping of oscillations to 1/10 amplitude in one cycle was used as the criterion for good handling qualities. This criterion is equivalent to a damping ratio characteristic (ζ = damping/critical damping) of 0.344.

The results indicate that the present damper is adequate to handle the modified airplane at low altitudes regardless of Mach number, gross weight and c.g. condition. This is illustrated in Figures 6-2 and 6-3. Thus no problems are anticipated in the take-off, initial climb and approach maneuvers.

The combined effects of increased gross weight and especially loss of control effectiveness through the elevon cut-outs, show somewhat marginal characteristics with respect to the criterion, at both ends of the speed range, at the test altitude (see Figure 6-4). The same curves show satisfactory damping in the transonic region.

Some general trends can be observed from the computed data for the higher altitudes:

a. Effect of G.W.: lower damping and frequency with increasing weight.

b. c.g. effect:

   1. Subsonic flight: damping improves as the c.g. moves aft.
   2. Supersonic flight: damping improves as the c.g. moves forward, with the cross over occurring in transonic flight.

c. Effect of speed: the best damping is obtained in the transonic region; it deteriorates slowly with speed increases. A more rapid fall off is observed in the lower speed regime.

d. The computed frequencies for the modified airplane show only small differences when compared with the F-106B at similar G.W. and c.g. conditions.

The loss of control effectiveness has the effect of lowering the damper loop gain. It appears that a simple upward revision of the pitch rate gain would improve the flying qualities of the airplane. The behavior of the frequency parameter suggests that the compensating network values can be left untouched.

It should be emphasized that there is no magic about the criterion selected from the handling qualities standpoint. A lower damping factor is merely a question of acceptability by the human controller, e.g. $\zeta = 0.25$ is equivalent to 1/10 amplitude in 1.4 cycles which is quite acceptable to some pilots. In the present situation we would recommend that the airplane be flown at least initially, with the present damper system. To preclude the possibility of having to delay or stop the tests to analyze a situation where the system may prove unsatisfactory in flight, Convair suggests that such analysis be done prior to flight in order to have a firm, positive standby solution if the need arises. From results now available, such a solution should not entail a change from the present "black-boxes".

6.2 LATERAL-DIRECTIONAL

No study was made of the lateral-directional characteristics under this part of the program. Although this can be justified in this initial phase by the nature of the modification, we suggest that a brief analysis be carried out prior to flight.

A quick approximation shows that somewhat lower frequency and damping should result from the increased inertias. Variations in the static aerodynamic derivatives are of the order of 2% and should contribute little change from the basic airplane. A plus factor is the fact that control effectiveness is maintained. Altogether it appears that the present yaw damper can handle the airplane.

The Turn Coordinator applies rudder in proportion to aileron deflection with a gain K_{δ_a} which is a function of dynamic pressure, q_c. The change in the $\dot{\phi}/\delta_a$ transfer function due to the elevon cutouts and increased inertias, indicates that K_{δ_a} should be revised downward. This should be investigated from the handling qualities standpoint.

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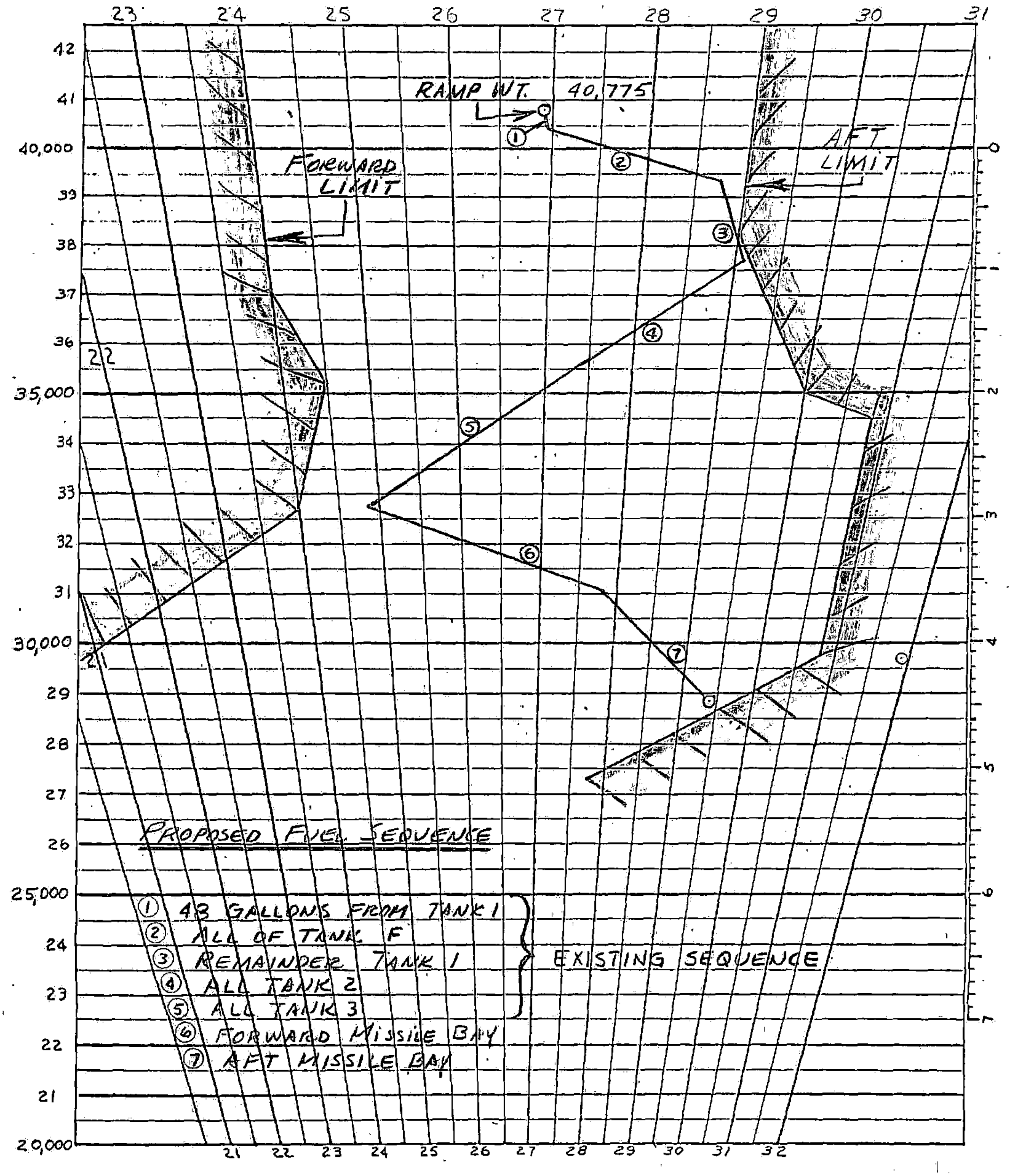
6. Giles, H. F. and Bulinski, R. J. "Additional Low-Speed Wind-Tunnel Tests of a .0991-Scale Model of the F-106A Airplane to Determine the Aerodynamic Effect of Tail Surfaces, External Stores, and Ground Plane Simulation" Convair Report CVAL-181C, 7-15-57.
7. Bohannon, W. C. and Zombek, W. S. "Performance Data Substantiation Report for the F-106B Flight Manuals T. O. 1F-106B-1 (Appendix I) dated 1 September 1961 and the F-106B Standard Aircraft Characteristics Charts dated 1 October 1961 "General Dynamics/Convair Report ZA-8-516, 1 September 1961.
8. Creed, R. L. and Wood, J. W. "F-106B Category II Performance, Stability and Control Tests" Air Force Flight Test Center, Report AFFTC-TR-60-13, June 1960.

MOM SCALE :
142,550 IN#/IN

Figure 1-1

GROSS WEIGHT VS. C.G.
FOR
RECOMMENDED FUEL SEQUENCE



PROPOSED FUEL SEQUENCE

① 48 GALLONS FROM TANK 1
② ALL OF TANK F
③ REMAINDER TANK 1
④ ALL TANK 2
⑤ ALL TANK 3
⑥ FORWARD MISSILE BAY
⑦ AFT MISSILE BAY

} EXISTING SEQUENCE

BL 412.16
MOM SCALE:
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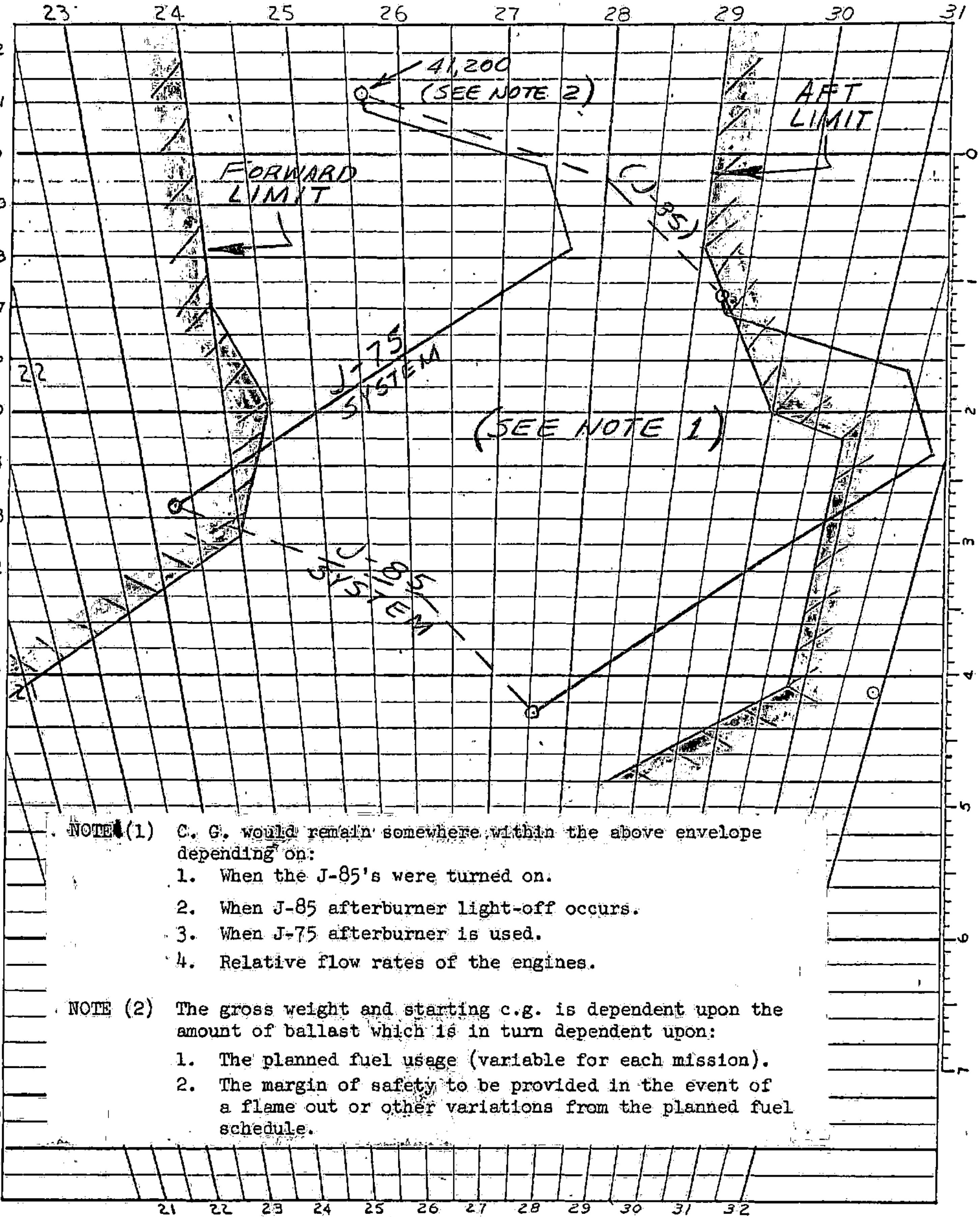
GROSS WEIGHT VS. C.G. FOR SEPARATE FUEL SYSTEMS



NOTE (1) C. G. would remain somewhere within the above envelope depending on:

1. When the J-85's were turned on.
2. When J-85 afterburner light-off occurs.
3. When J-75 afterburner is used.
4. Relative flow rates of the engines.

NOTE (2) The gross weight and starting c.g. is dependent upon the amount of ballast which is in turn dependent upon:

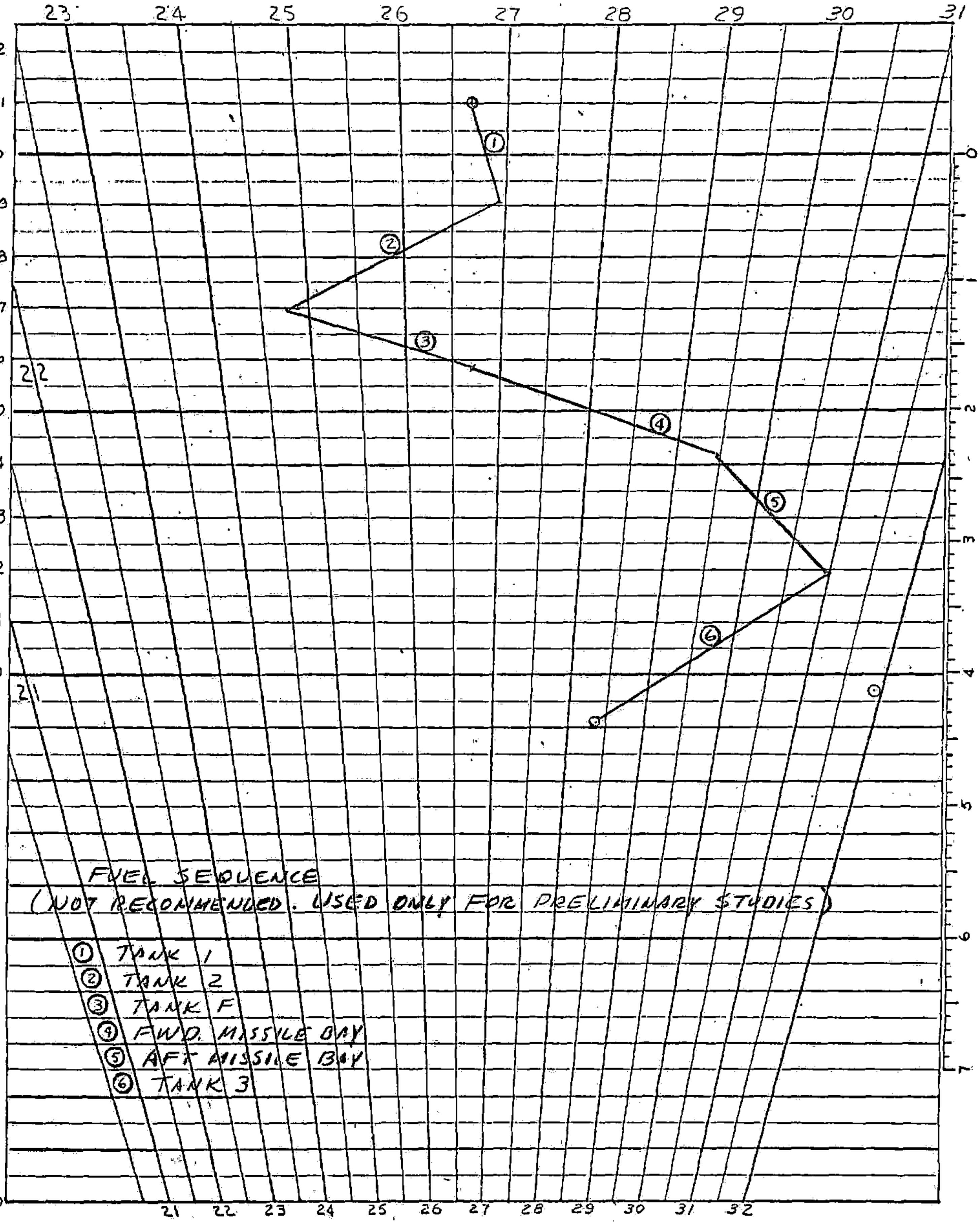
1. The planned fuel usage (variable for each mission).
2. The margin of safety to be provided in the event of a flame out or other variations from the planned fuel schedule.

Figure 1-3

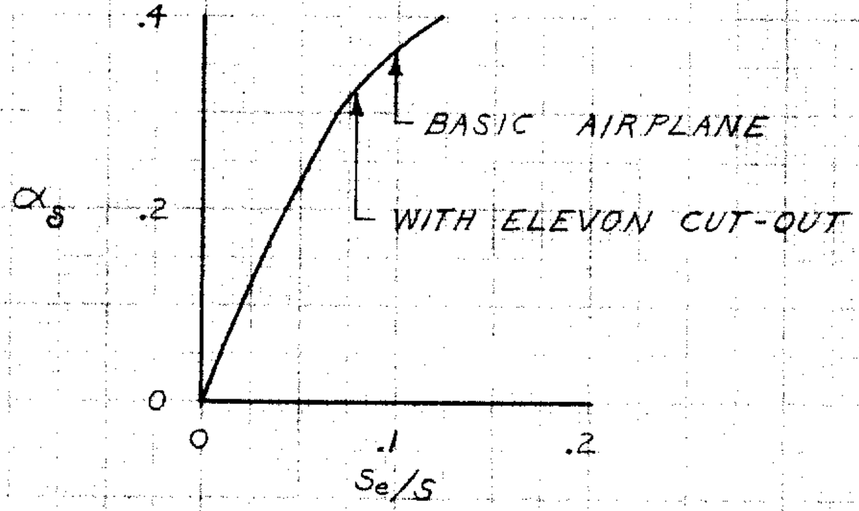
MOM SCALE :
142,550 IN#/IN

GROSS WEIGHT VS. C.G.

(SEQUENCE USED FOR EARLY STABILITY STUDY)
(NOT RECOMMENDED)



F-106B SST ENGINE TEST BED
ELEVON CONTROL EFFECTIVENESS



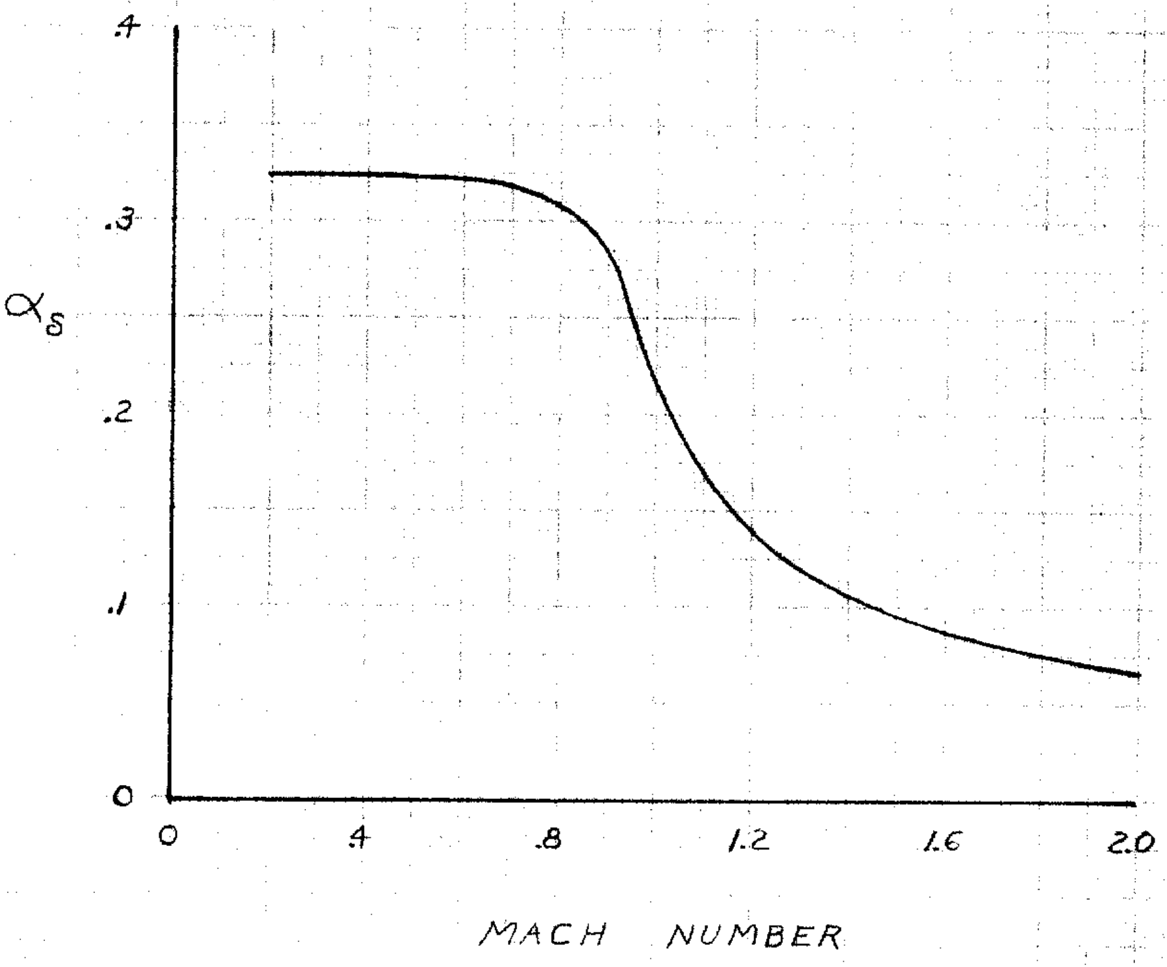
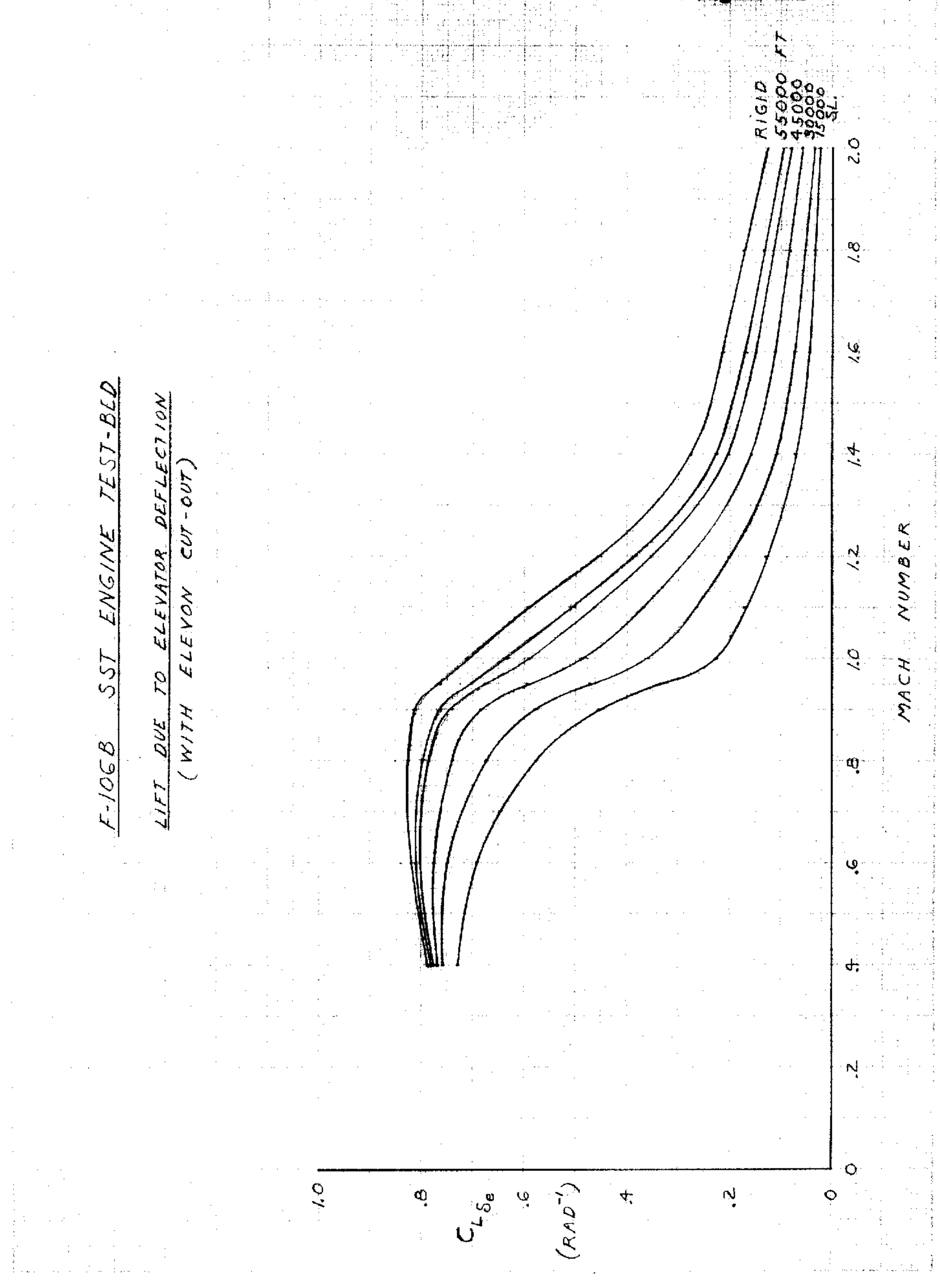


Figure 2-2

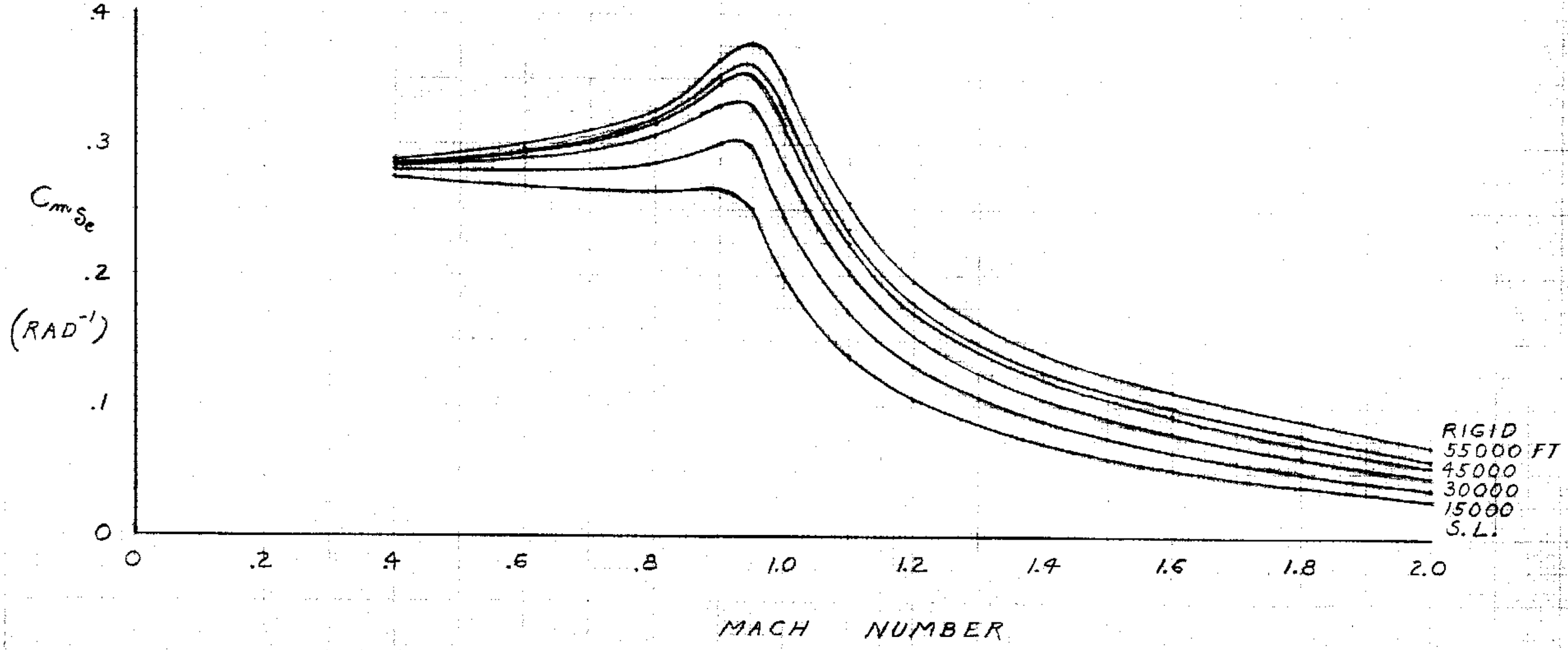


F-106B SST ENGINE TEST-BLD

LIFT DUE TO ELEVATOR DEFLECTION
(WITH ELEVON CUT-OUT)

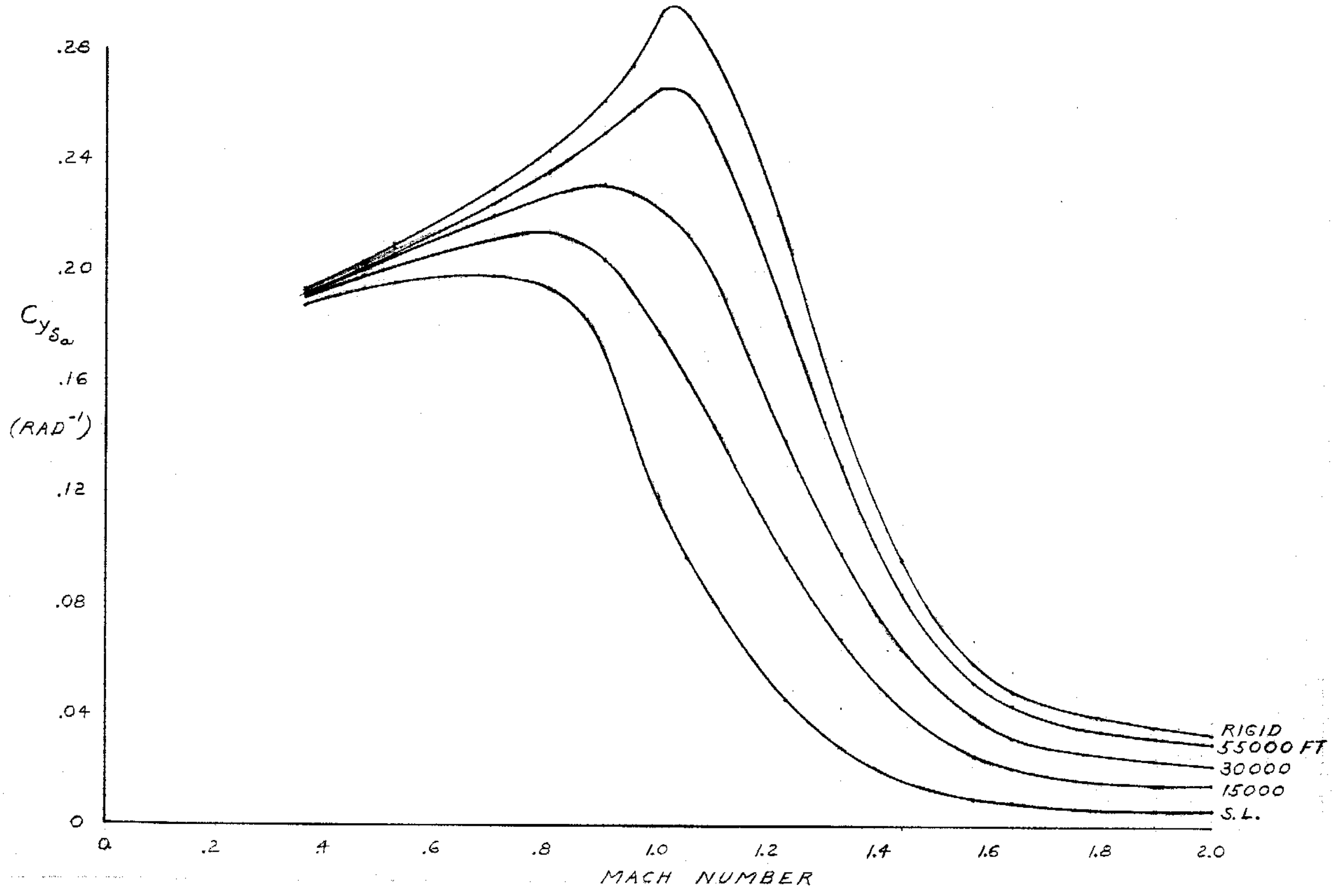
F-106B SST ENGINE TEST-BED

PITCHING MOMENT DUE TO ELEVATOR DEFLECTION
(WITH ELEVON CUT-OUT)



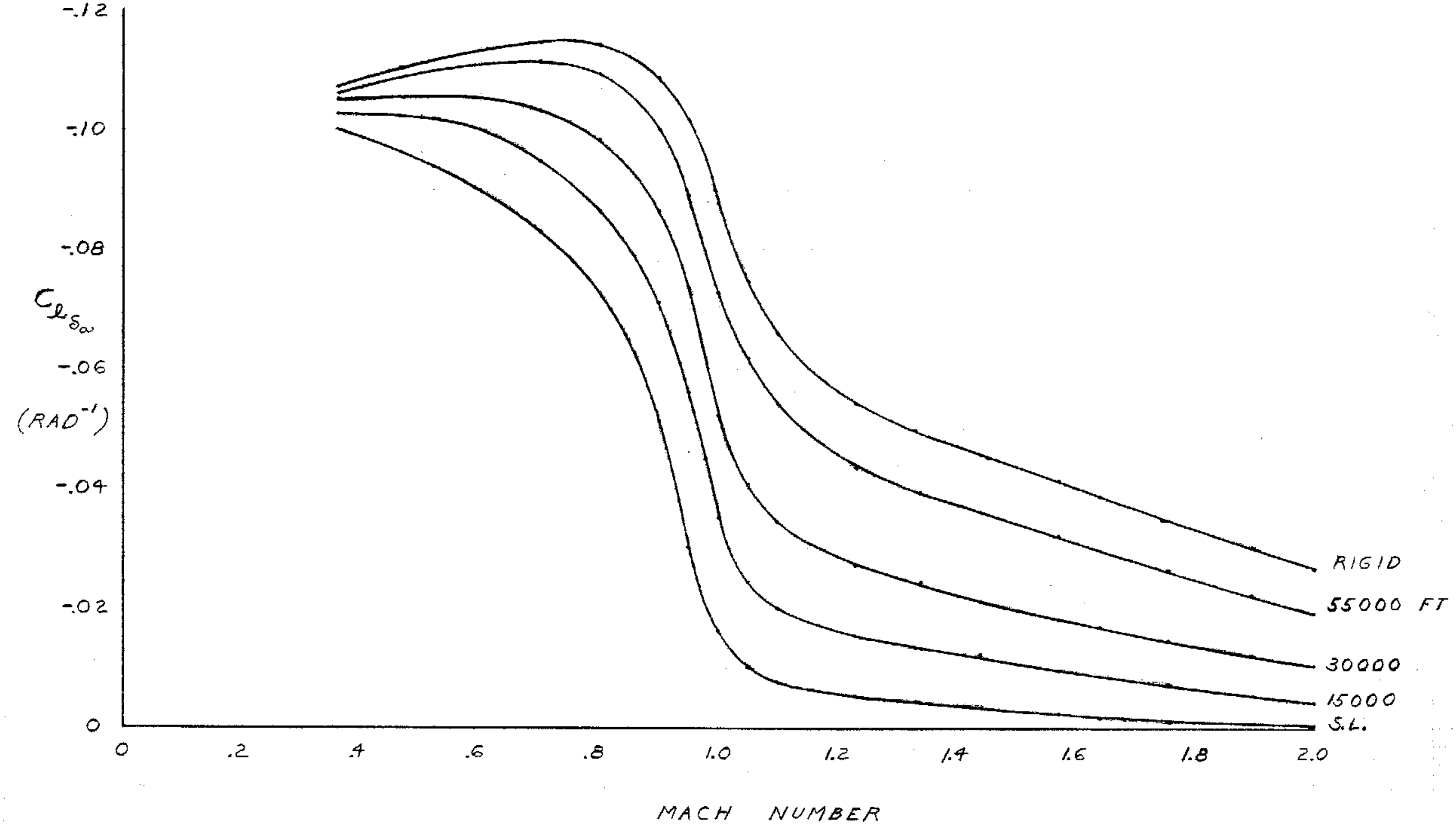
F-106B SST ENGINE TEST-BED

SIDE FORCE DUE TO AILERON DEFLECTION
(WITH ELEVON CUT-OUT)



F-106B SST ENGINE TEST-BED

ROLLING MOMENT DUE TO AILERON DEFLECTION
(WITH ELEVON CUT-OUT)



F-106B SST ENGINE TEST-BED

YAWING MOMENT DUE TO AILERON DEFLECTION

(WITH ELEVON CUT-OUT)

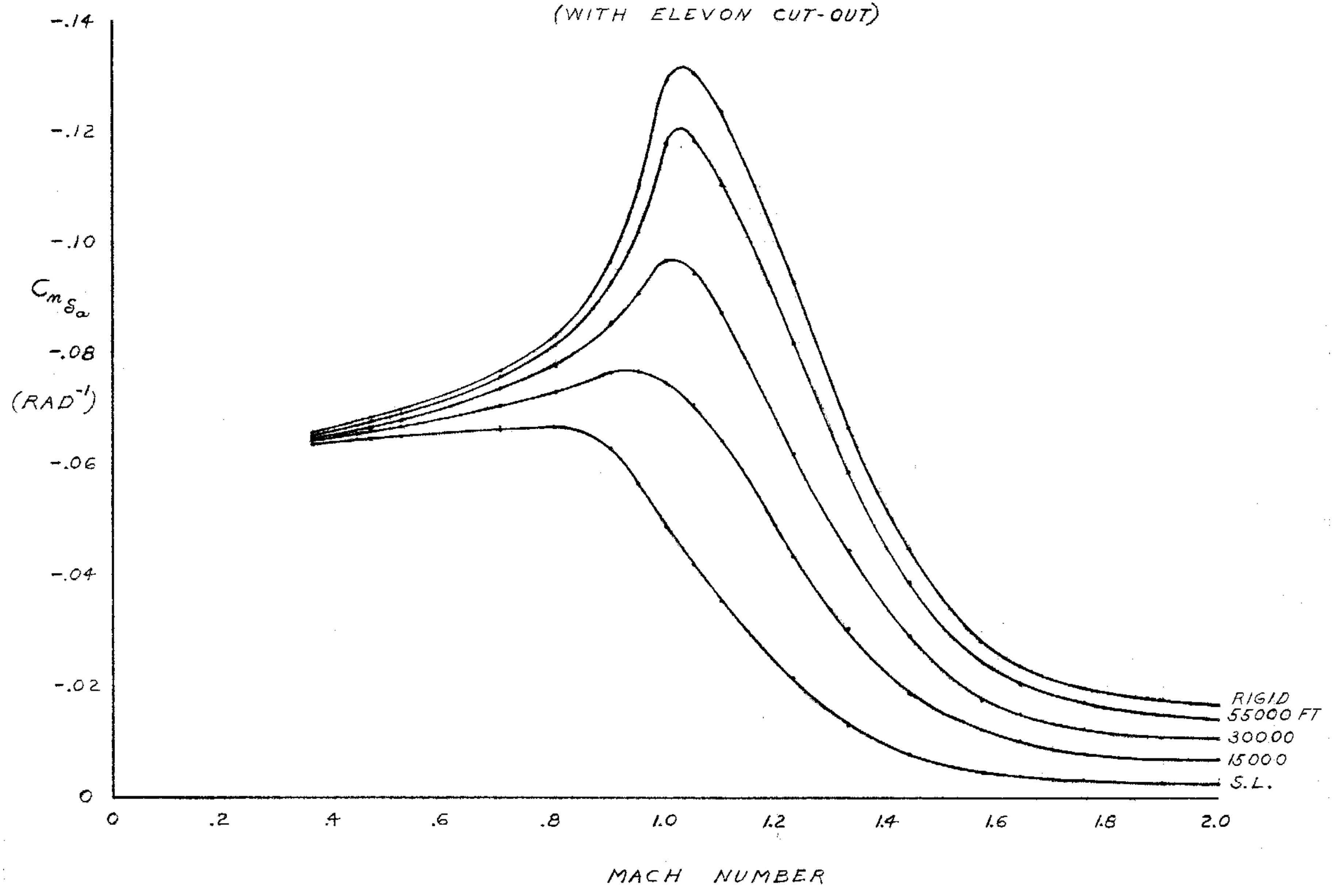
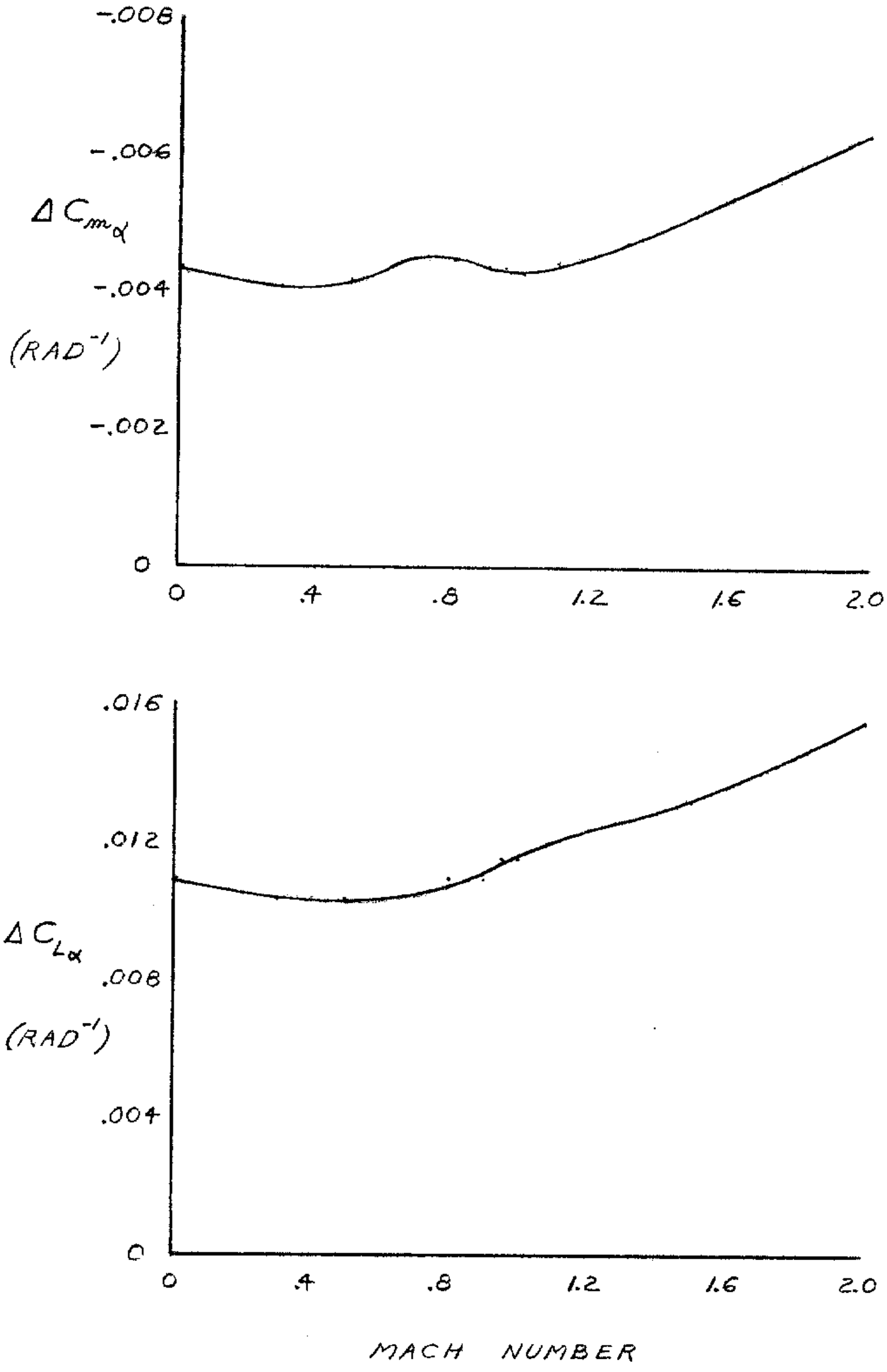


Figure 2-7

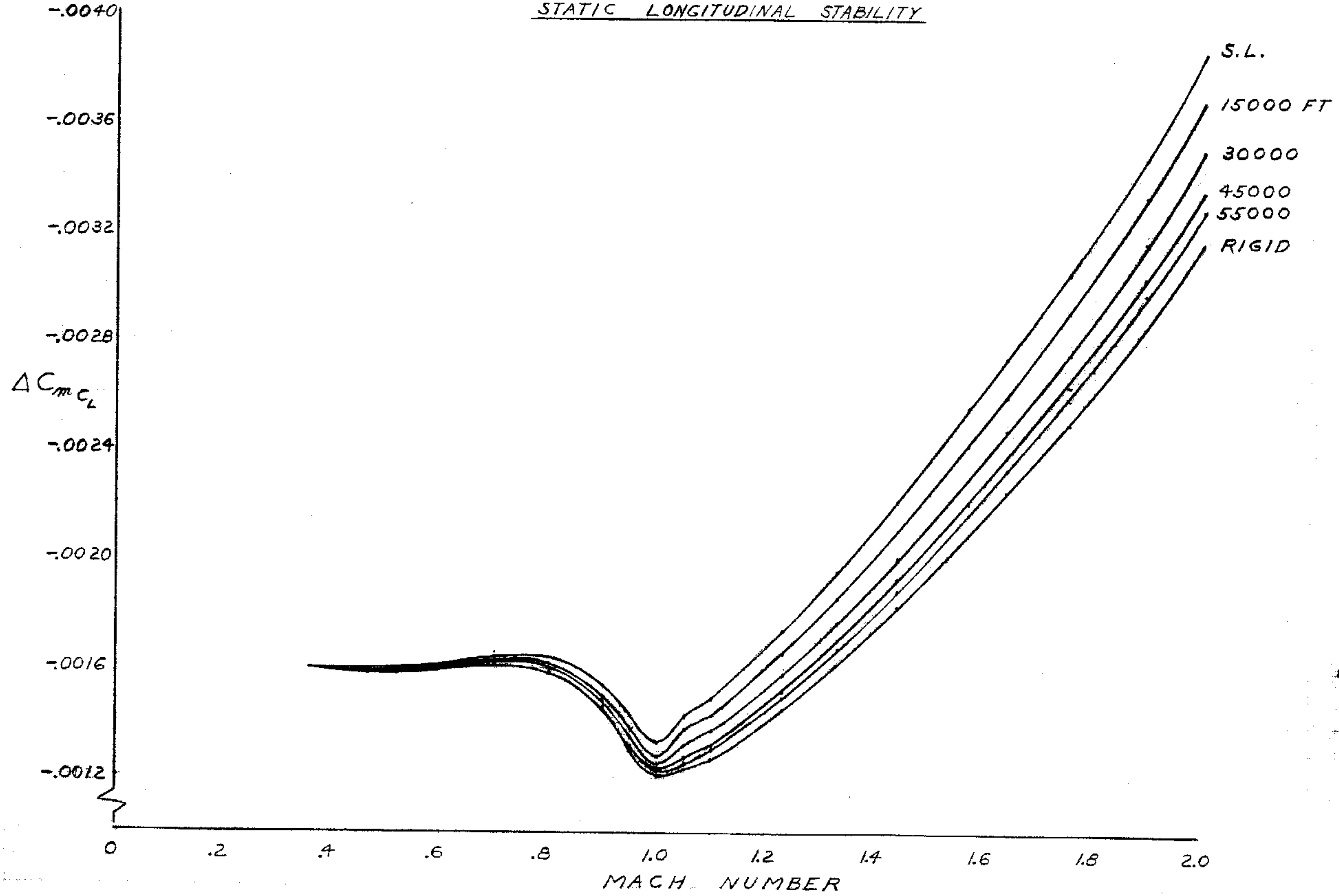
F-106B SST ENGINE TEST BED

TEST ENGINE NACELLE LIFT & PITCHING MOMENT



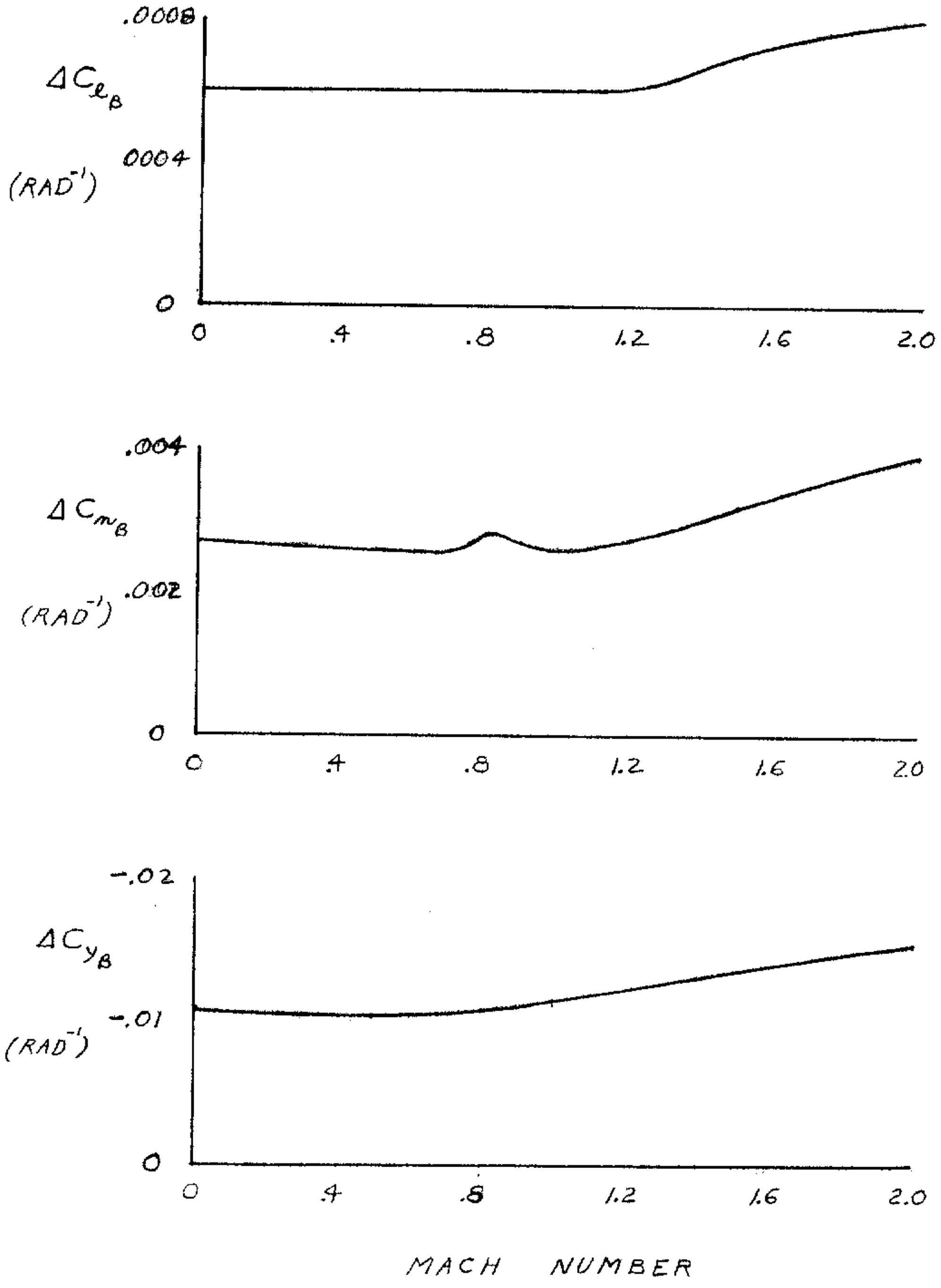
F-106B SST ENGINE TEST-BED

EFFECT OF TEST ENGINE NACELLES ON STATIC LONGITUDINAL STABILITY



F-106B SST ENGINE TEST BED

EFFECT OF TEST ENGINE NACELLES ON LATERAL-DIRECTIONAL STABILITY DERIVATIVES



F-106B SST ENGINE TEST-BED EXTERNAL DRAG INCREMENT OF TEST ENGINE NACELLES

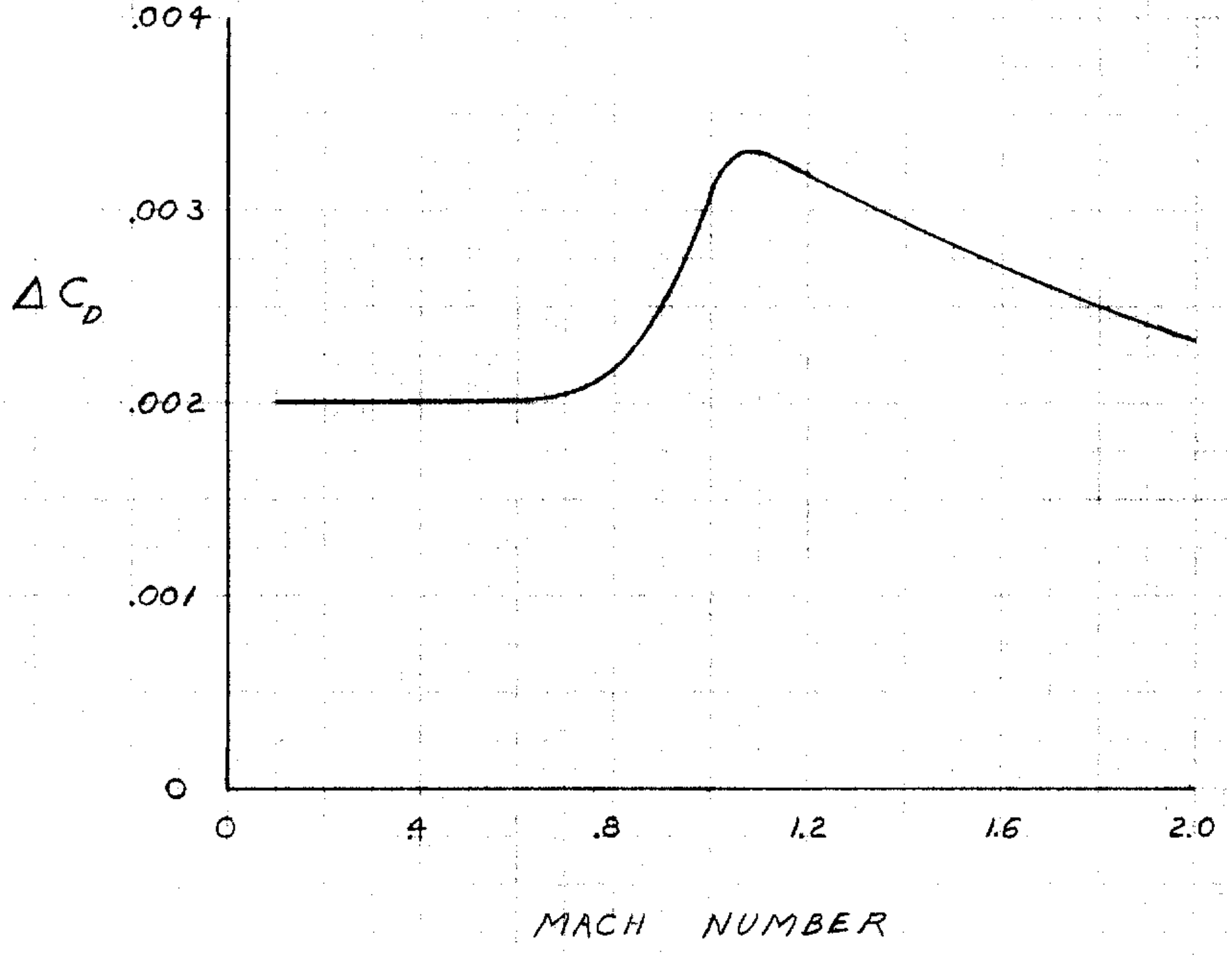


Figure 2-11

F-106B SST ENGINE TEST-BED

J-85 ENGINE-OUT DRAG INCREMENT

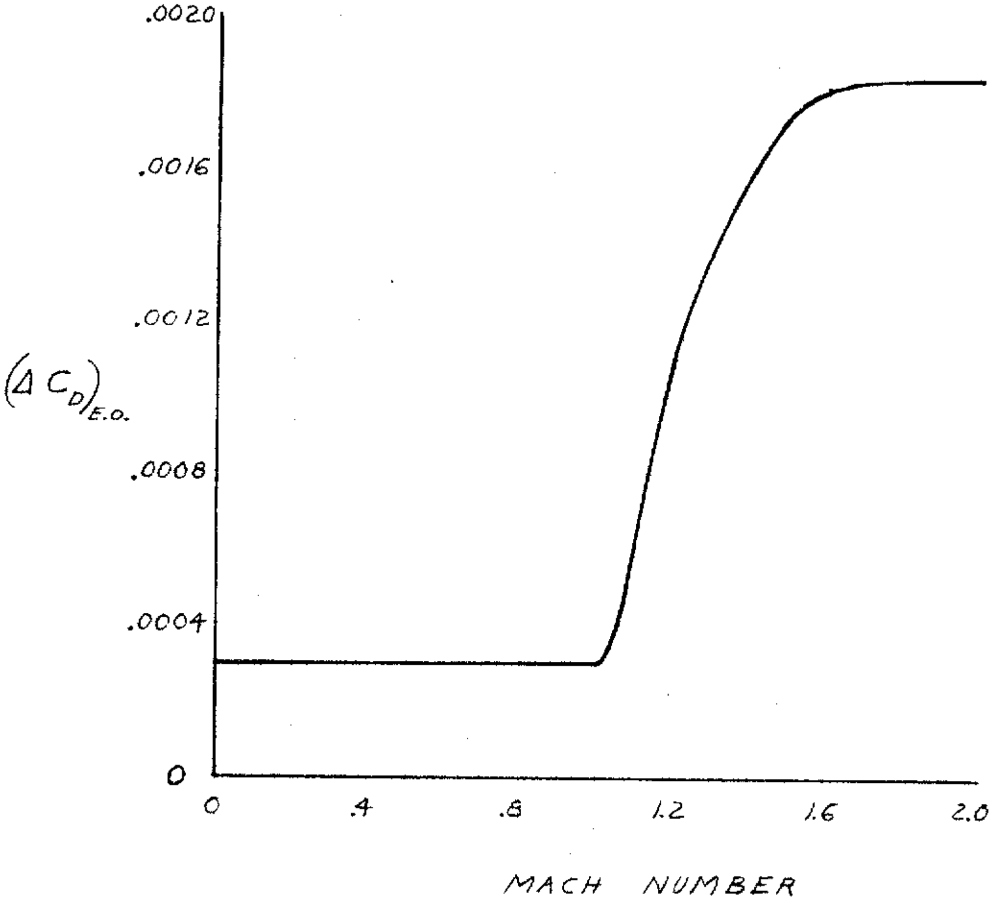


TABLE 3-1

Flight Condition	Fuel, Used, (Lbs)	
	J-75	J-85
Both Missions		
Takeoff and Accelerate to Climb Speed	1070	----
Climb to 25,000 ft.	923	----
Climb to 40,000 ft.	560	*
TOTALS	2553	*
Mission #1		
Accelerate to M = 1.2	335	118
Cruise at M = 1.2 for 15 min.	1425	2220
Descent	390	----
TOTALS	2150	2338
TOTALS FOR MISSION #1:	4703	2338
RESERVE:	3279	1562
Mission #2		
Accelerate to M = 2.0	1899	681
Cruise at M = 2.0 for 3.8 min.	1940	965
Descent	390	----
TOTALS	4229	1646
TOTALS FOR MISSION #2:	6782	1646
RESERVE:	1200	2254
** Cruise at M = 2.0 for additional 2.9 min	1494	760

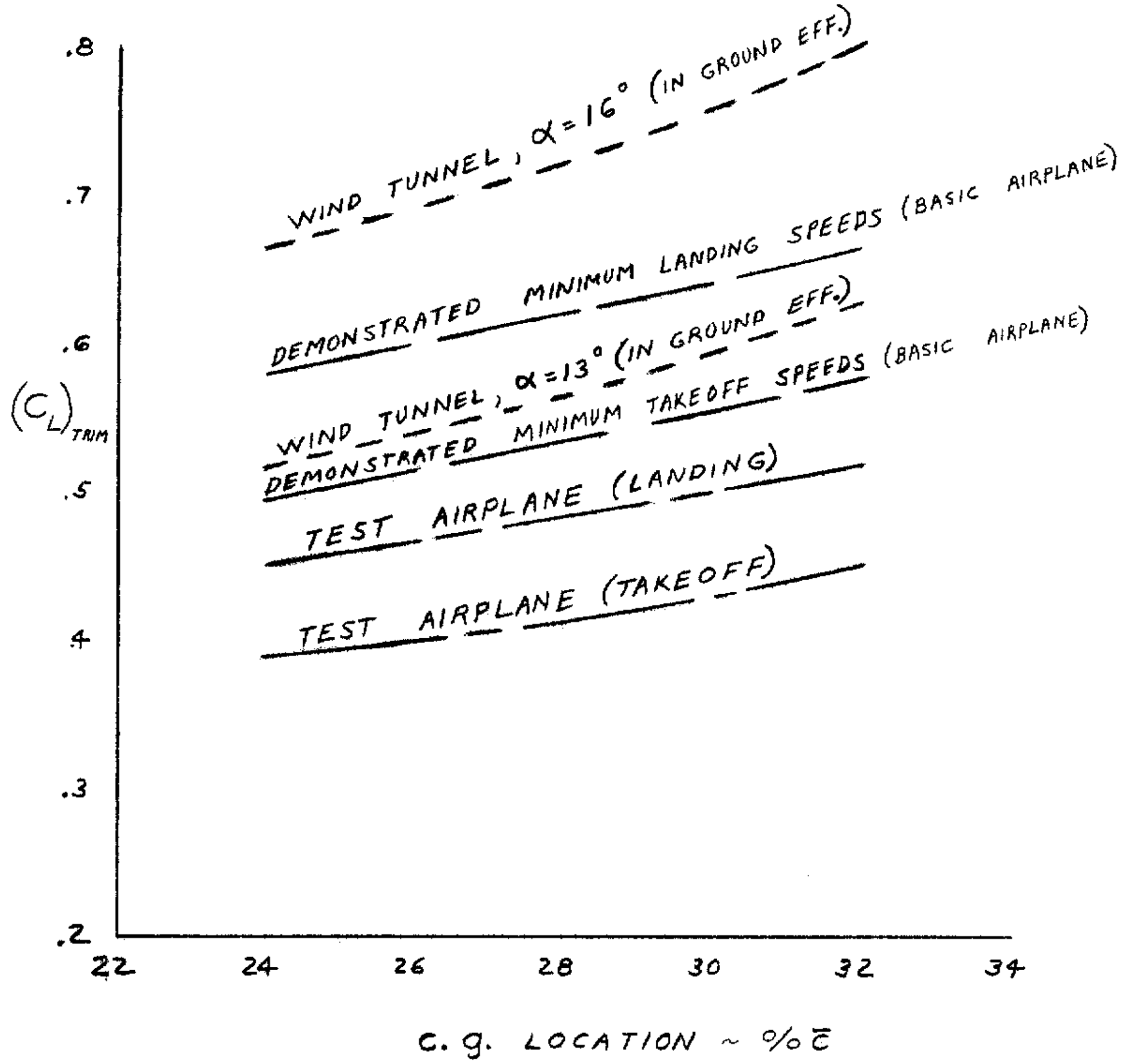
NOTES: *J-85 fuel flow at Mil Power not available at Convair. Time to climb from 25 K to 40 K is 1.3 min.

**If the remaining 2254 lbs of J-85 fuel is shared with the J-75, an additional 2.9 min. cruise at M = 2.0 is available.

Figure 3-1

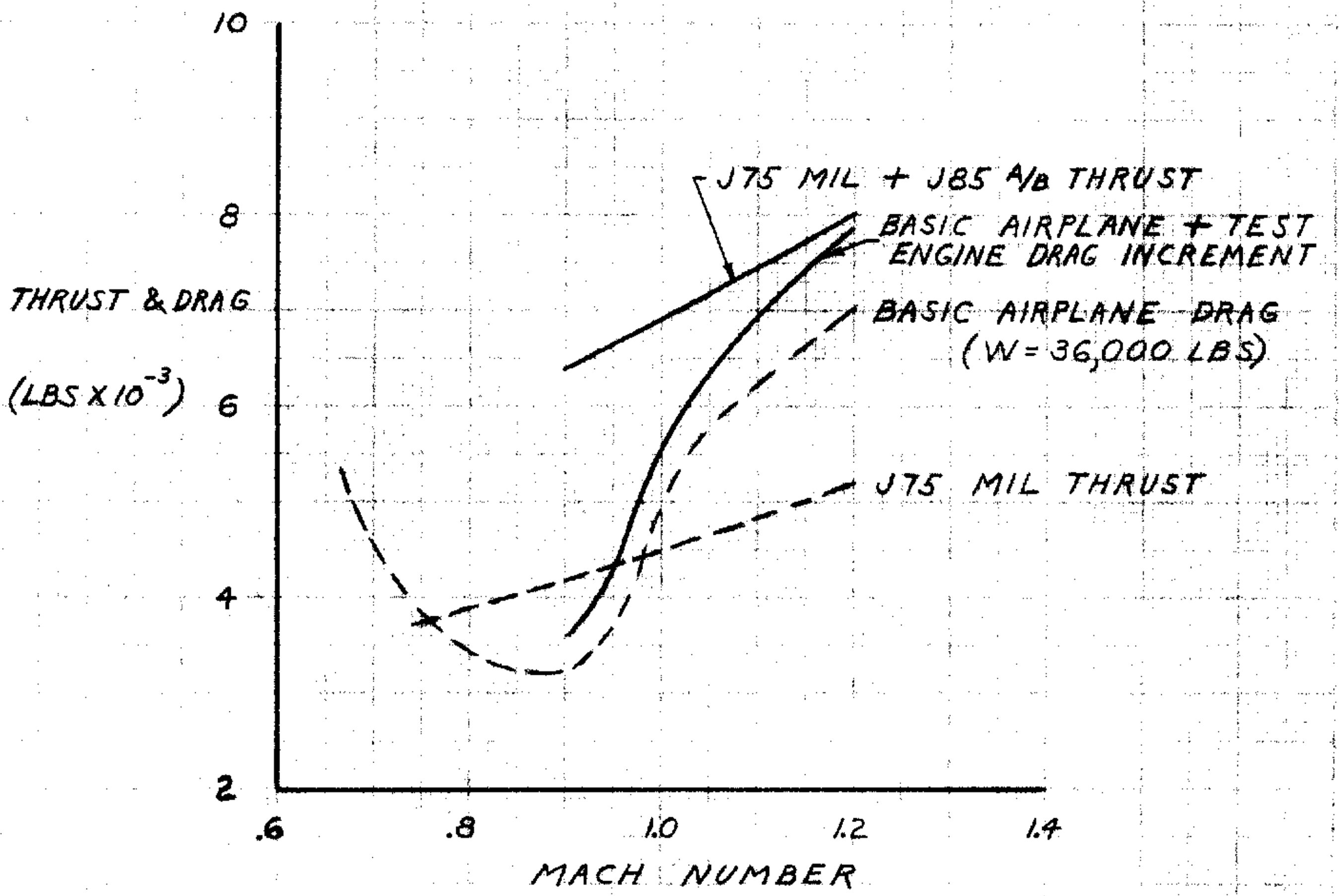
F-106B SST ENGINE TEST-BED

TRIMMED LIFT COEFFICIENTS FOR TAKEOFF AND LANDING



F-106B SST ENGINE TEST-BED

THRUST AND DRAG AT 40000 FT



F-106B SST ENGINE TEST-BED

LOAD FACTOR CAPABILITY

NOTE:
EXTREMETIES OF FLIGHT ENVELOPE
ARE FOR BASIC AIRPLANE

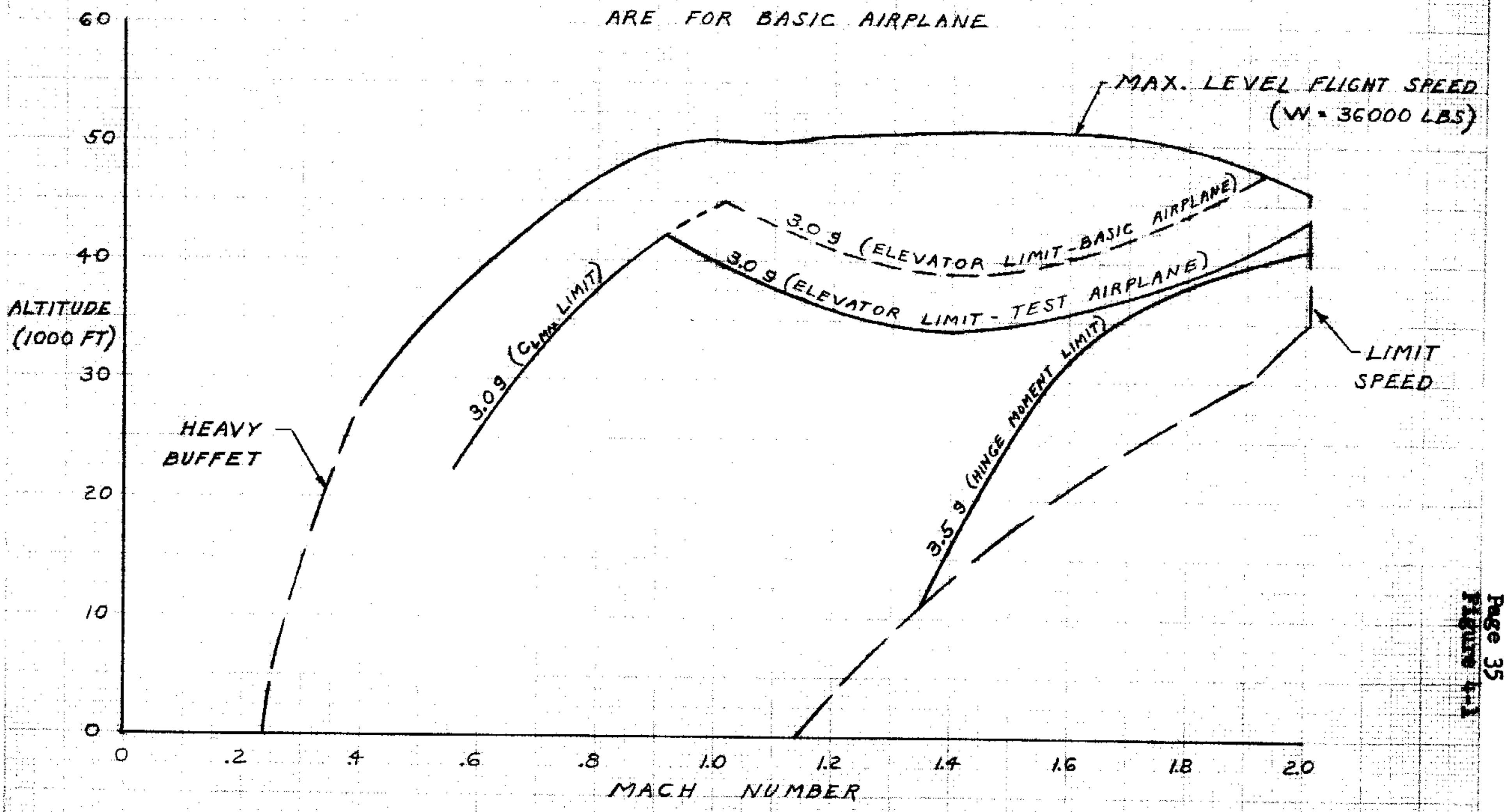


TABLE 5-1

SUMMARY OF ENGINE FAILURE SIMULATIONS

Case No.	Alt.	Mach No.	G.W.	Asym. Lift	MIL Thrust Decay	Pilot Time Delays		Pedal Force	Y.D. & T.C.	Extreme	Extreme	Max Tail Load
	h (ft.)	M	W (lbs)	Z_e (lbs)	τ_e (sec)	t_{δ_A} (sec)	t_{δ_R} (sec)	F_p (lbs)		ϕ (deg)	β (deg)	(lbs)
1a	30,000	2.0	37,000	1600	1.5	1.5	1.5	180	ON	-3.3	-.7	5000
1b	"	"	"	"	"	"	"	0	"	-3.2	-.7	5150
1c	"	"	"	"	0.1	"	2.0	180	"	-4.8	-.7	7450
1d	"	"	"	"	1.0	0.5	"	"	OFF	-.6	-1.0	5750
2a	55,000	1.2	34,000	200	"	1.5	"	"	ON	-.7	-.79	2000
2b	"	"	"	400	"	"	"	"	"	-1.6	-.95	1350
2c	"	"	"	"	"	"	"	0	"	-1.6	-.95	1350
2d	"	"	"	"	"	"	"	180	OFF	-1.7	-1.00	1430
3a	"	2.0	31,000	500	"	"	"	"	ON	+1.5	-.59	2700
3b	"	"	"	1000	"	"	"	"	"	-3.0	-.68	2610

FIGURE 5-1

AIRPLANE AND CONTROL SYSTEM EQUATIONS

$$(1) \qquad \dot{\beta} = -\dot{\psi}\cos\phi + \alpha_o\dot{\phi} + \frac{g}{U_o}\sin\phi + \frac{\bar{q}S}{mU_o}\Big[C_{y_\beta}\beta + C_{y_{\delta_A}}\delta_A + C_{y_{\delta_R}}\delta_R + \frac{b}{2U_o}\big(C_{y_p}\dot{\phi} + C_{y_r}\dot{\psi}\big)\Big]$$

$$(2) \qquad \ddot{\phi} = \left(\frac{I_{xz}}{I_{xx}}\cos\phi + \theta_o\right)\ddot{\psi} + \left(\frac{I_{zz}-I_{yy}}{I_{xx}}\right)\dot{\psi}^2\sin\phi\cos\phi + \frac{\bar{q}Sb}{I_{xx}}\Big[C_{l_\beta}\beta + C_{l_{\delta_A}}\delta_A + C_{l_{\delta_R}}\delta_R + \frac{b}{2U_o}\big(C_{l_p}\dot{\phi} + C_{l_r}\dot{\psi}\big)\Big] + \frac{y_e}{I_{xx}}C_{z_{F_e}}F_e$$

$$(3) \qquad \ddot{\psi} = \sec\phi\left\{\frac{I_{xz}}{I_{zz}}\left(\ddot{\phi} - \dot{\psi}^2\cos\phi\sin\phi\right) + \left(\frac{I_{xx}-I_{yy}+I_{zz}}{I_{zz}}\right)\dot{\phi}\dot{\psi}\sin\dot{\phi} + \frac{\bar{q}Sb}{I_{zz}}\Big[C_{n_\beta}\beta + C_{n_{\delta_A}}\delta_A + C_{n_{\delta_R}}\delta_R + \frac{b}{2U_o}\big(C_{n_p}\dot{\phi} + C_{n_r}\dot{\psi}\big)\Big] - \frac{y_e}{I_{zz}}F_e\right.$$

WHERE

All angles are in radians,

y_e is the distance of the outboard engines from the XZ plane, feet,

$\bar{q} = \frac{\rho U_o^2}{2}$ is the dynamic pressure, lb/ft^2,

$C_{z_{F_e}}$ is the asymmetric lift per unit thrust resulting from failure of the right hand engine,

F_e is the thrust change for the right hand engine (negative for thrust loss), lbs.

FIGURE 5-1 (Continued)

In the following equations, the variables are in degrees. They were converted to radian measure prior to substitution in Equation (1)-(3).

$$\hat{\delta}_R = \hat{\delta}_{R_1} + \hat{\delta}_{R_2} + \hat{\delta}_{R_3} \tag{4}$$

$$\dot{\hat{\delta}}_{R_1} = K_{\dot{\psi}}\dot{\psi} - K_{\dot{\phi}}\dot{\phi} - \hat{\delta}_{R_1} \tag{5}$$

$$\hat{\delta}_{R_2} = K_{\delta_A}\delta_A \tag{6}$$

$$\dot{\hat{\delta}}_{R_3} = \frac{1}{\tau_p}\left(\frac{F_p}{K_p} - \hat{\delta}_{R_3}\right) \tag{7}$$

where

$K_{\dot{\psi}}$, $K_{\dot{\phi}}$ and K_{δ_A}, gains of the yaw damper/turn coordinator, are functions of compressible dynamic pressure, q_c

$\hat{\delta}_{R_3}$	is the pilot-commanded rudder angle, deg.
τ_p	is the pilot's lag, sec.
F_p	is the applied rudder pedal force, lbs.
K_p	is the rudder feel spring stiffness, lbs/deg.

$$\dot{\delta}_R = K_V\left[\left(\hat{\delta}_R - \delta_R\right) \pm \epsilon_{MAX}\sqrt{\left|\mp HMR/HMR_{MAX}\right|}\right. \tag{8}$$

where K_V is the forward loop gain (set at unity for this study)

ϵ_{Max} is the minimum rudder error at which maximum hydraulic flow occurs (valve wide open), deg.

HMR_{Max} is the maximum available rudder hinge moment = 6000 ft-lb.

FIGURE 5-1 (Continued)

(9) $$HMR = \bar{q} S_R \bar{c}_R \left\{ \left(C_{h_\beta}\right)_R \beta + \left(C_{h_{\delta_A}}\right)_R \delta_A + \left(C_{h_{\delta_R}}\right)_R \delta_R + \frac{b}{2U_o}\left[\left(C_{h_p}\right)_R \dot{\phi} + \left(C_{h_r}\right)_R \dot{\psi}\right]\right\}$$

(10) $$|HMR| \leq HMR_{MAX}$$

(11) $$|\delta_R| \leq \delta_{R_{MAX}}$$

where $\delta_{R_{Max}}$, maximum rudder delfection, is a function of q_c

(12) $$\delta_A = K_\phi \phi + K_{\dot{\phi}} \dot{\phi}$$

where K_ϕ and $K_{\dot{\phi}}$ are roll and roll rate gains representing the pilot's attempt to maintain lateral control.

(13) $$|\delta_A| \leq \delta_{A_{MAX}}$$

where $\delta_{A_{Max}}$ is the aileron limit = 7 degrees

(14) $$F_e = F_{AB}\left(1-\frac{t}{0.2}\right) + F_M\left(1-e^{-t/\tau_e}\right), \qquad F_{AB}\left(1-\frac{t}{0.2}\right) \geq 0$$

where F_e is total thrust change on the RH engine lbs.

F_{AB} is the afterburner thrust on the RH engine, lbs.

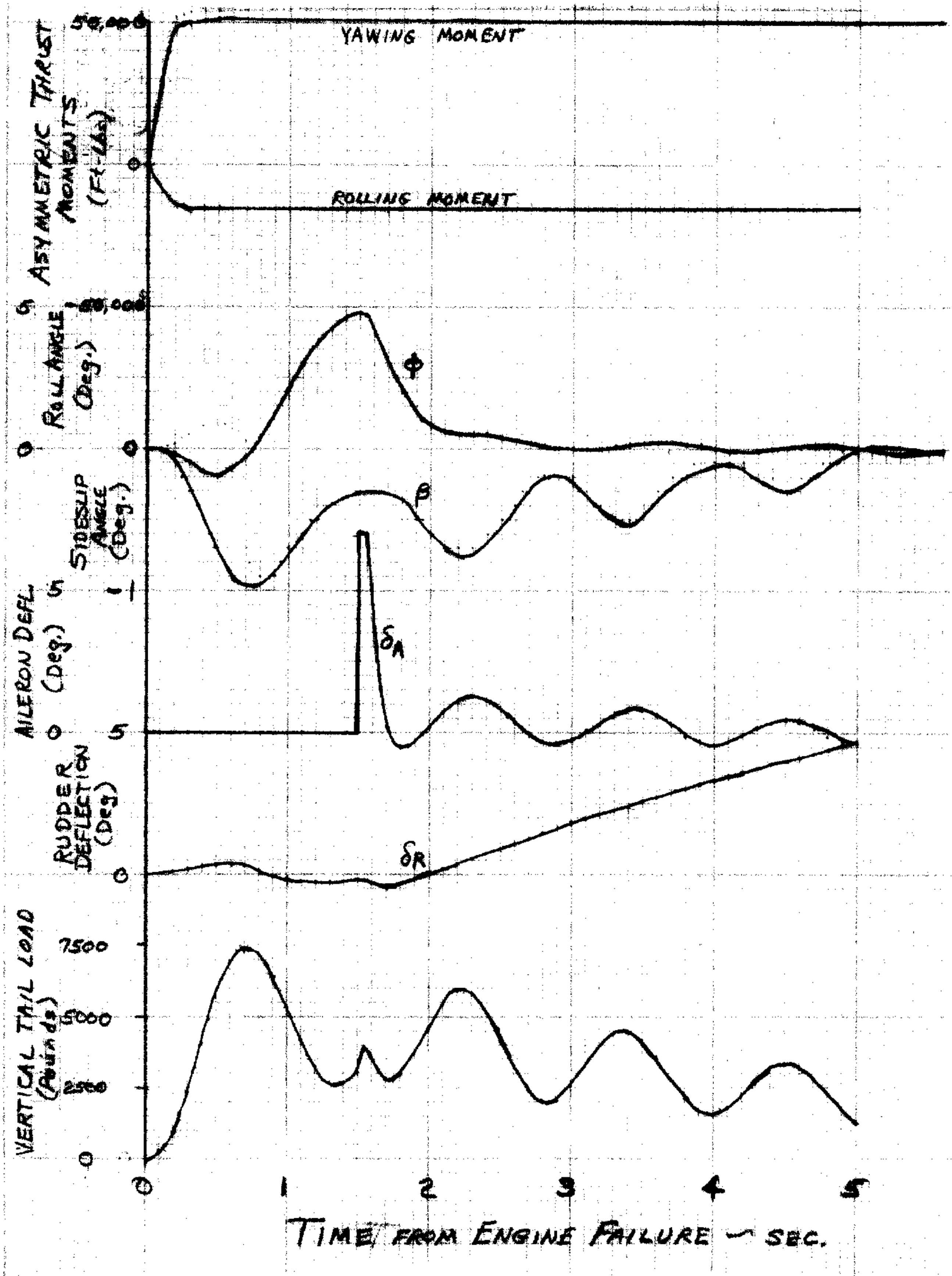
F_M is the military thrust on the RH engine, lbs.

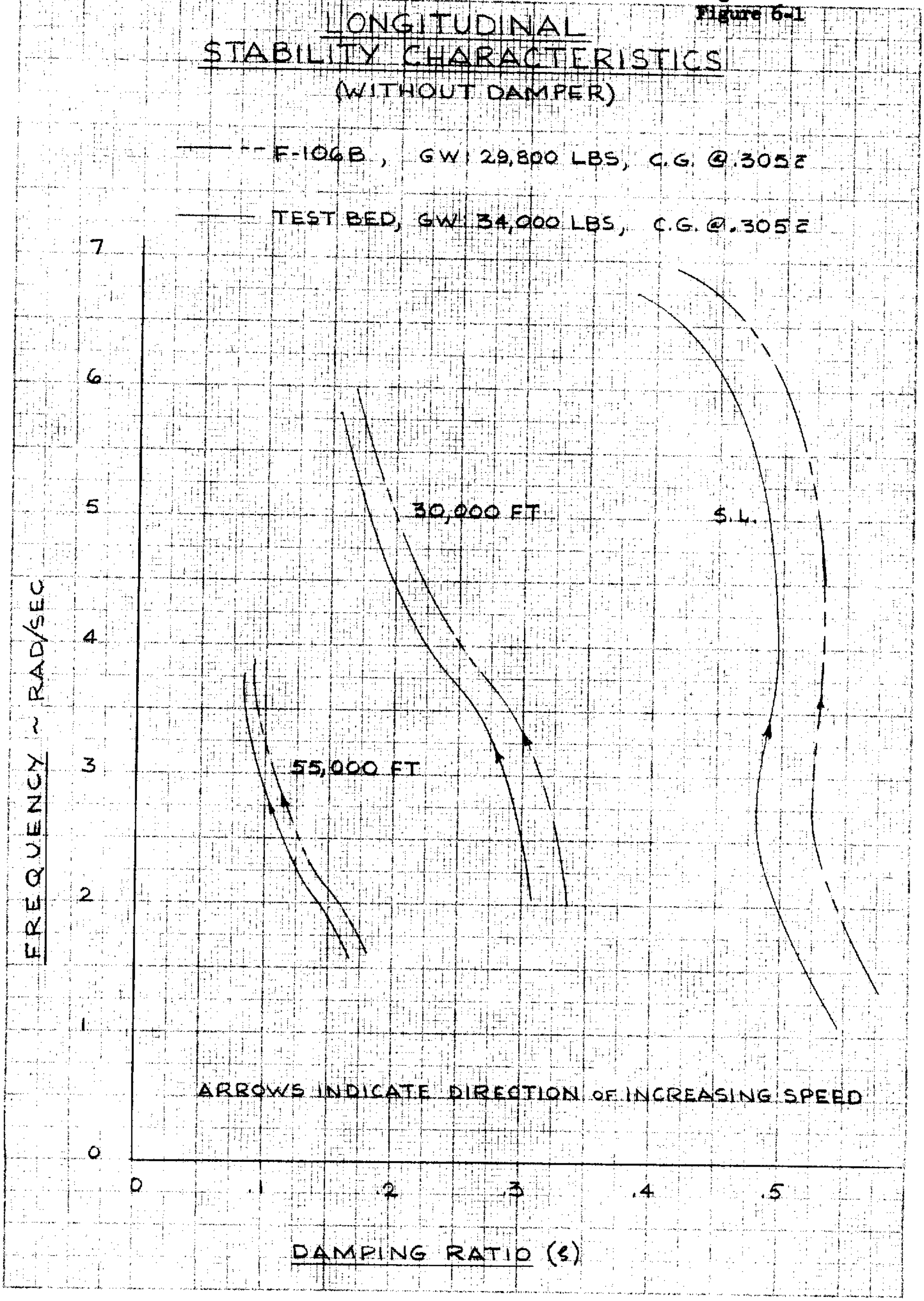
t is the time from start of engine failure, sec.

τ_e is the time constant for thrust decay, sec.

F-106B TEST BED ENGINE-OUT RESPONSE

MACH NO. = 2.0 AT 30,000 FT.



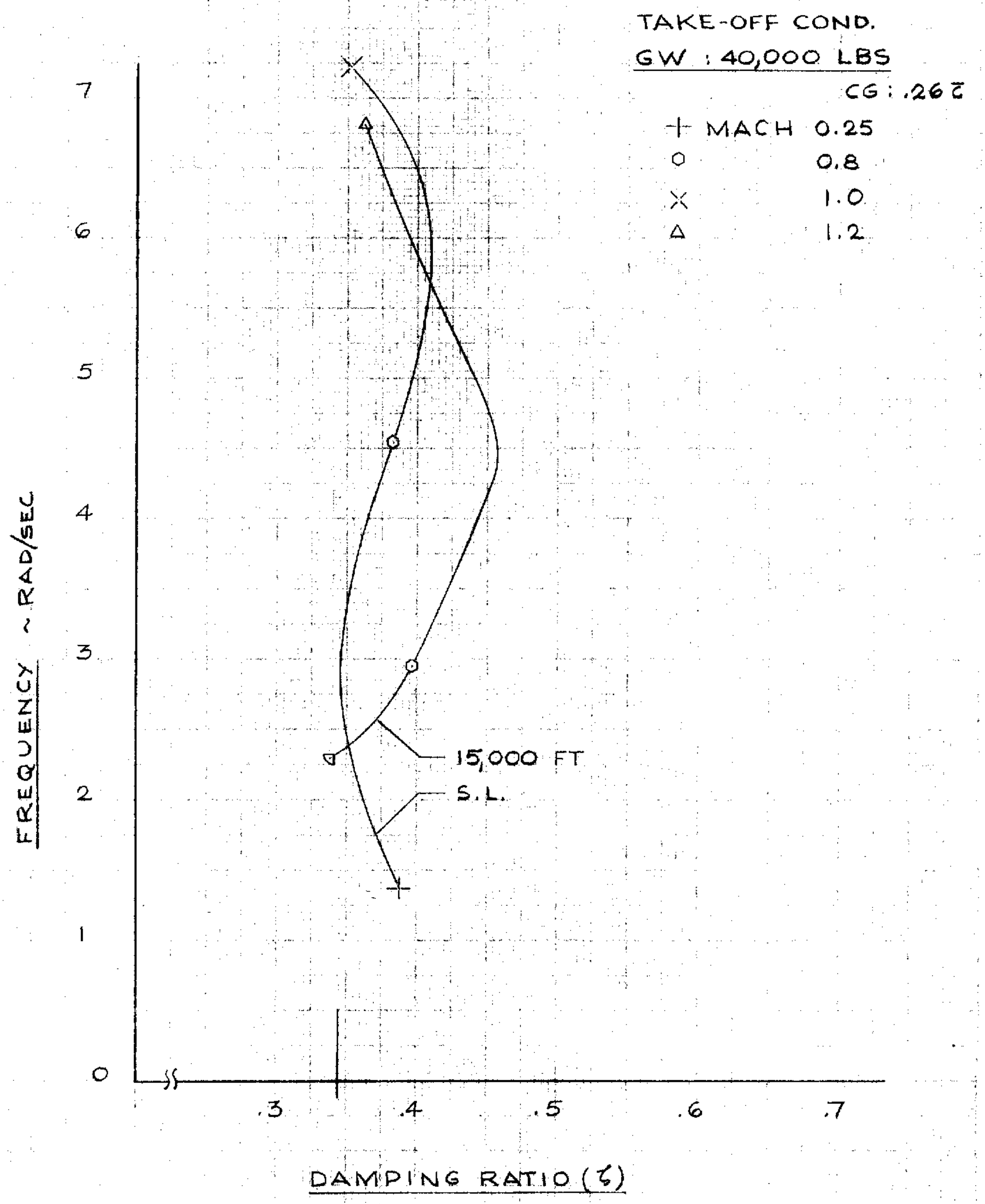


LONGITUDINAL STABILITY CHARACTERISTICS (WITHOUT DAMPER)

ARROWS INDICATE DIRECTION OF INCREASING SPEED

F-106B SST TEST BED PITCH CHARACTERISTICS

(WITH F-106B DAMPER)



F-106B SST TEST BED PITCH CHARACTERISTICS

(WITH F-106B DAMPER)

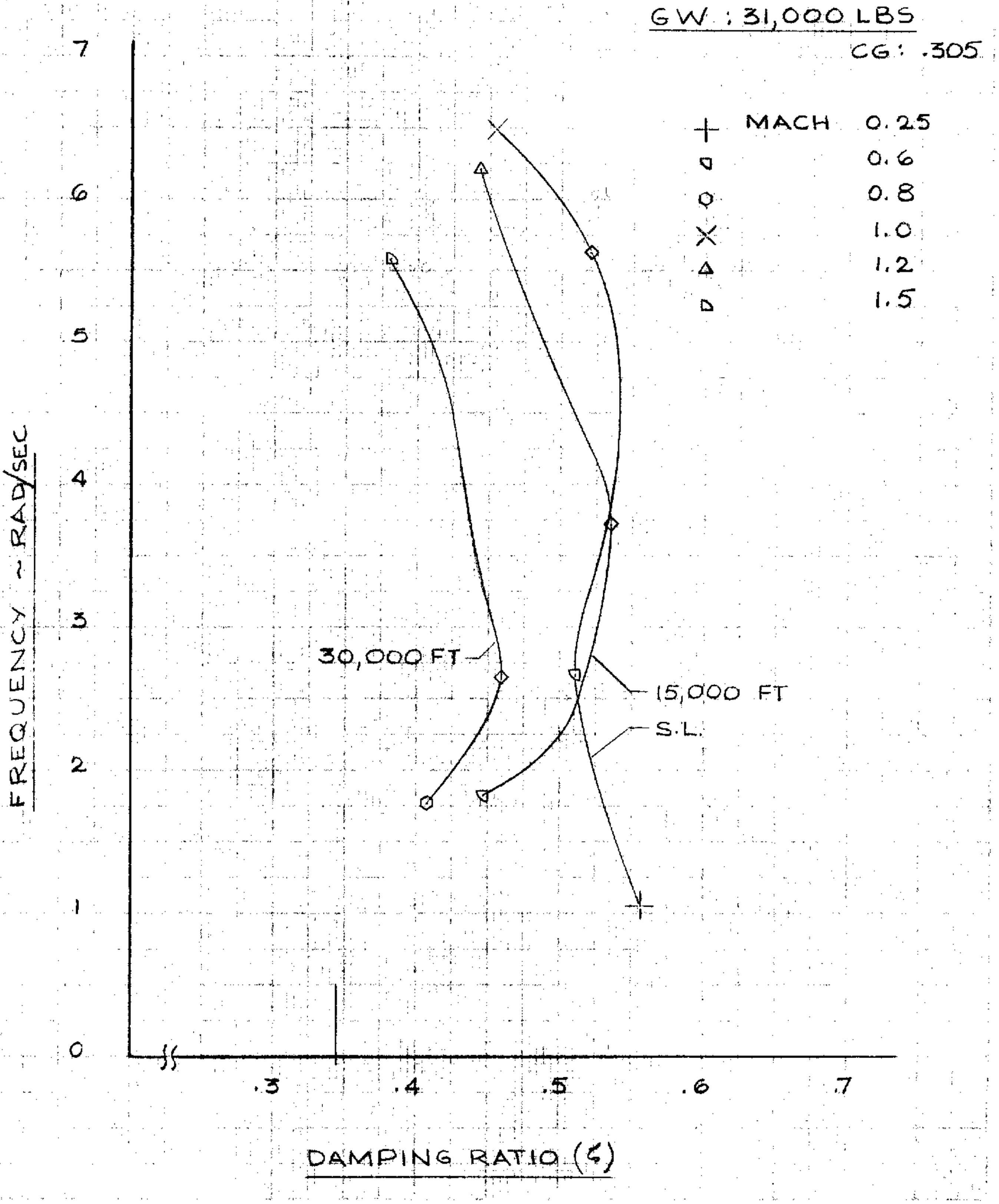
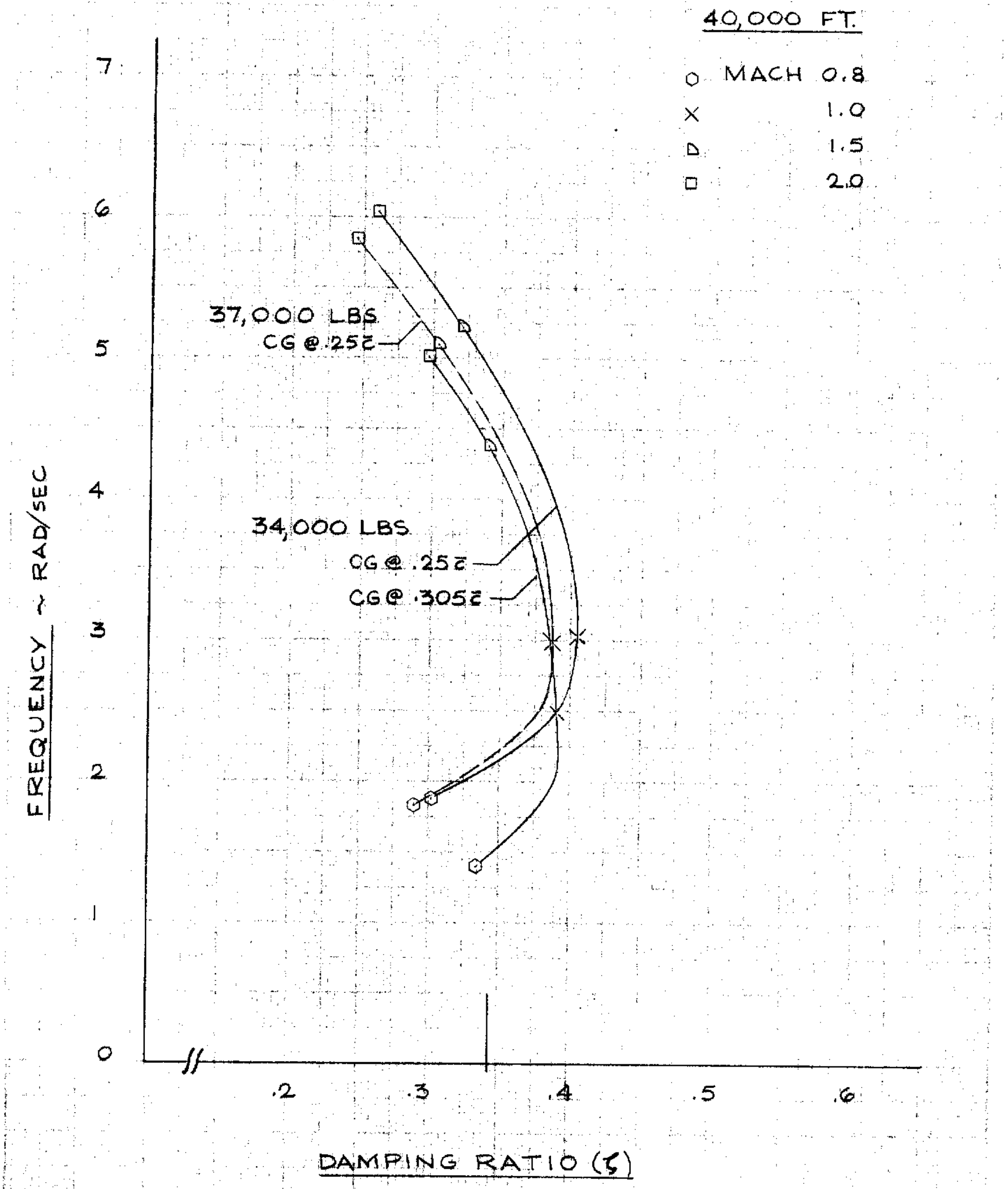


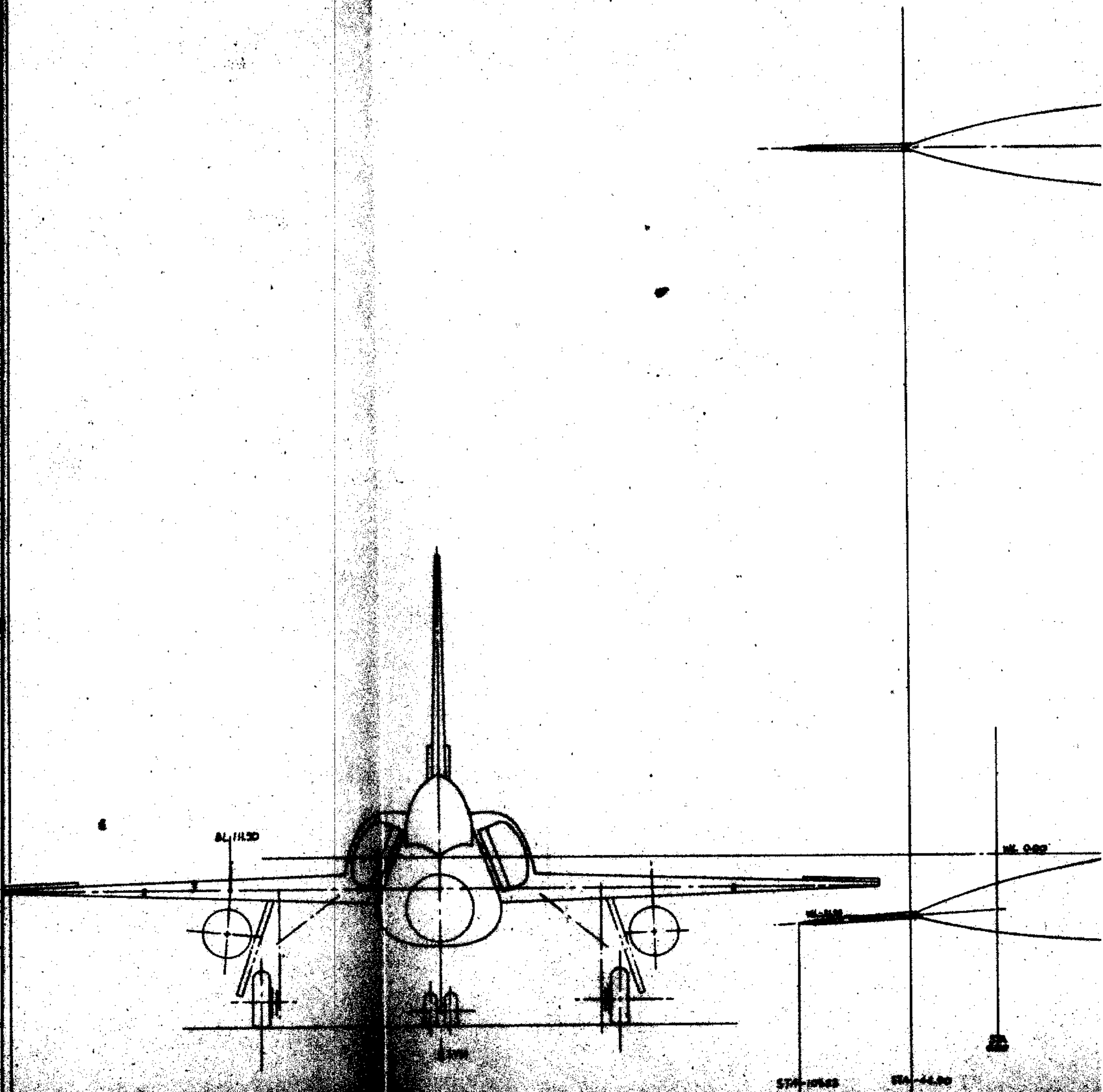
Figure 6-4

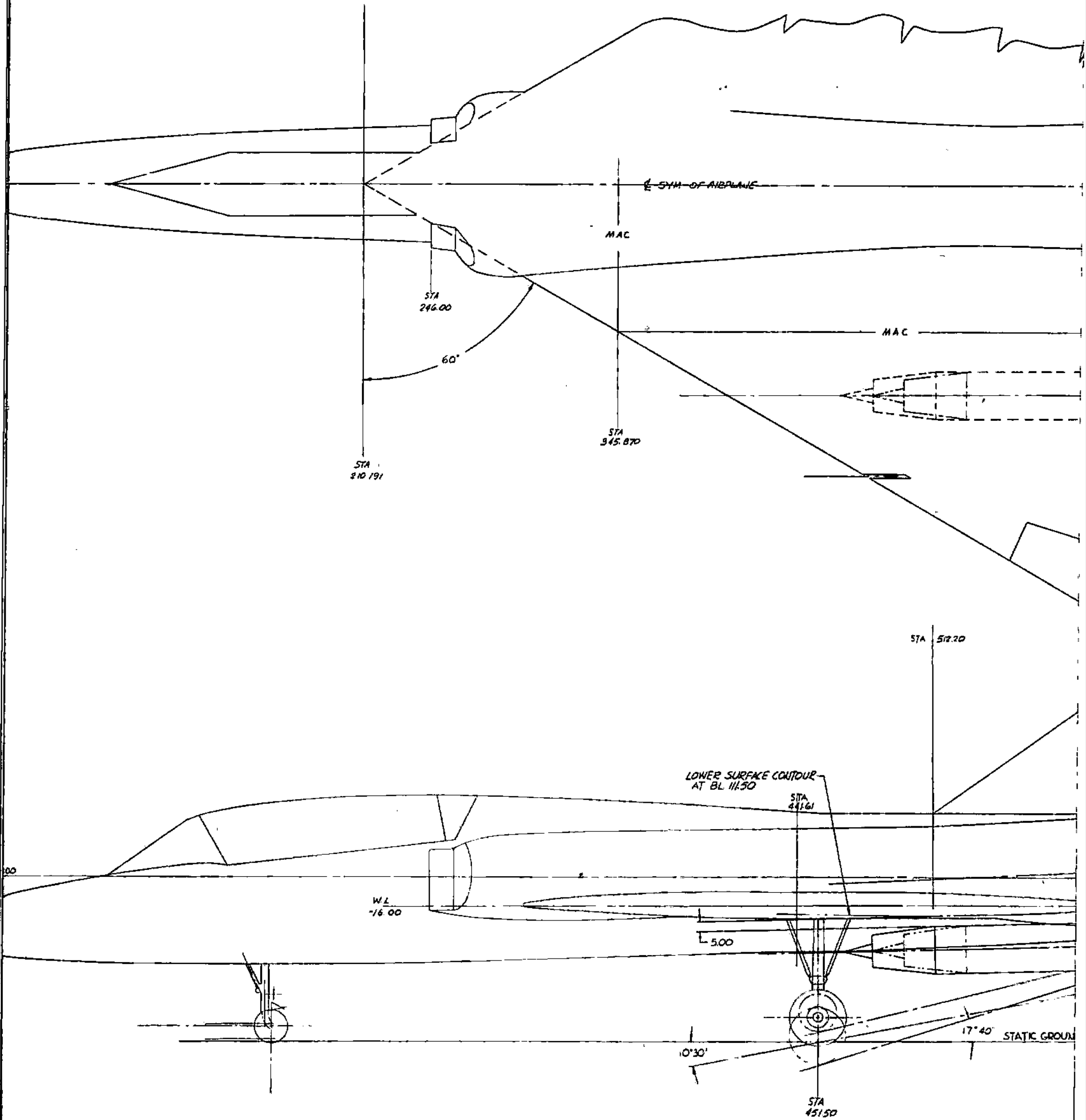
F-106B SST TEST BED
PITCH CHARACTERICTICS

(WITH F-106B DAMPER)



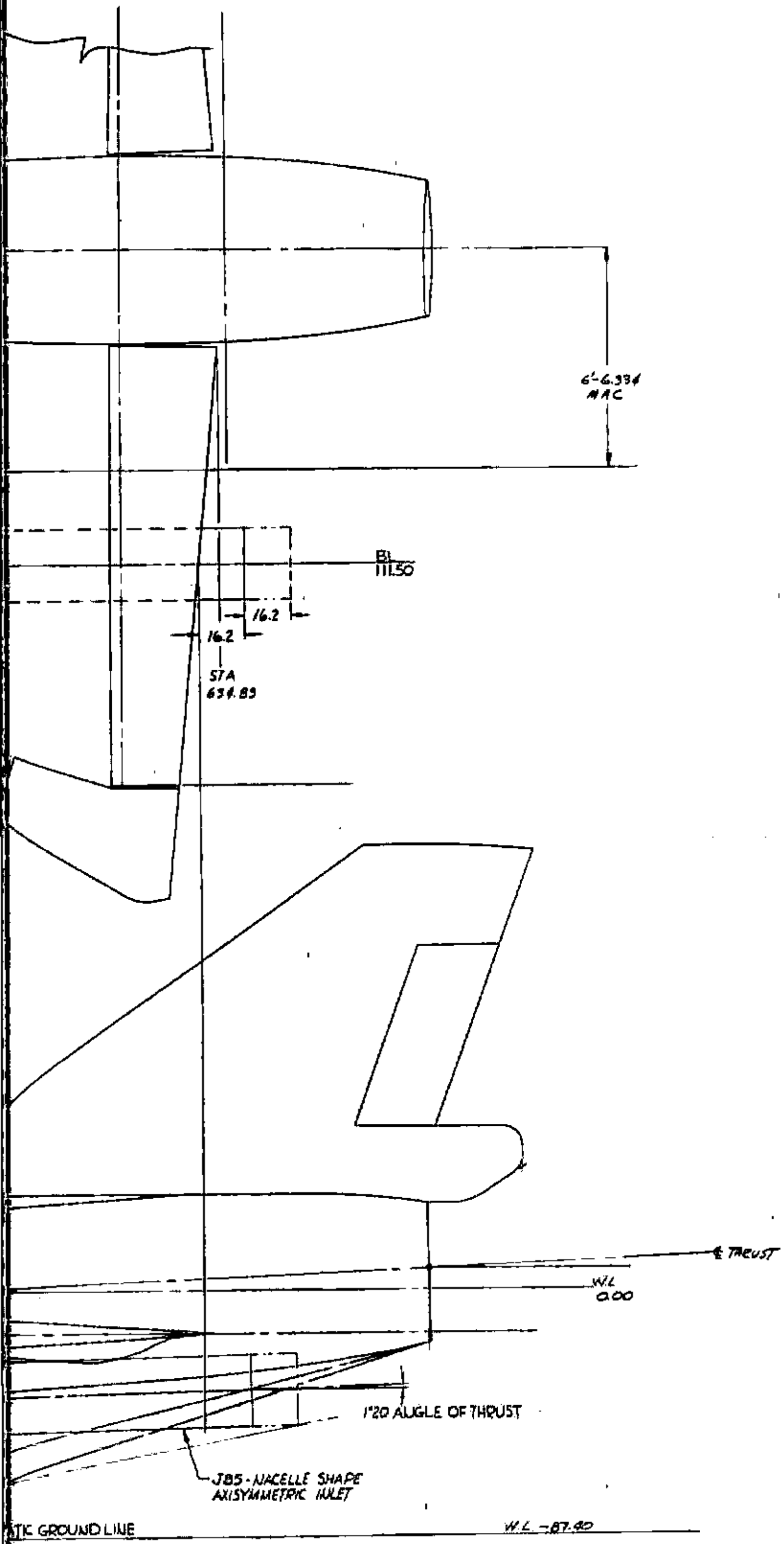
2-10


BL 111.30
WL 0.00
STA -44.00



℄ SYM OF AIRPLANE
MAC
STA 246.00
60°
MAC
STA 345.870
STA 210.191
STA 512.20
LOWER SURFACE CONTOUR AT BL 111.50
STA 441.61
00
W.L -16.00
5.00
10°30'
17°40'
STATIC GROUN
STA 451.50

GDC-66-062
ADDENDUM



QUANTITY REQUIRED
ITEM NO.
PART OR IDENT NO.
CODE IDENT NO
DASH NO.
OPP
SHN
NOTE NO.
I-R
NOM OR DESCR
MATERIAL
STOCK SIZE
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LIST OF MATERIALS (☐ FOR PARTS LIST DATA SEE DOCUMENT SAME NUMBER PREFIXED PL)

UNLESS OTHERWISE SPECIFIED
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APPROVED
SUBCONTRACT DESIGN ACTIVITY
CODE IDENT NO.
CONVAIR APPROVAL
CONTRACT NUMBER

GENERAL DYNAMICS | CONVAIR
SAN DIEGO 12, CALIFORNIA

F106B SST TEST BED-NASA

CODE IDENT NO.
14170
SIZE
SD66-05001
SCALE 1/20
RELEASED
SHEET OF

ZP-8-062

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[redacted]

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP 8-062
DATE August 1958
MODEL F102/F106

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by: Dept. Date
130-1 4-4-67

TITLE

F102-F106 INTERCEPTOR
FIGHTER-BOMBERS

PREPARED BY T.M. Hemphill
T. M. Hemphill

GROUP

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert
Chief of Pre Design

NO. OF PAGES

NO. OF DIAGRAMS

REVISIONS

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FORM 1812 A-4

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INTRODUCTION AND SUMMARY

The Convair F102A, the most effective All Weather Supersonic Interceptor that has been produced in quantity, is serving in squadrons of the Air Defense Command of the United States Air Force defending the United States against attack.

F106A Interceptors are being tested by Convair and by the USAF. Quantities are being produced at the Convair San Diego Division of General Dynamics Corporation, to supplement the F102A Interceptors already in service.

These Convair aircraft, because they are of delta-wing design, are fundamentally superior to the more conventional types of aircraft for interceptor and fighter missions. They are simpler, safer, and more rugged. Their superiority and safety is being demonstrated daily.

The size and arrangement of the F102A and F106A aircraft make them adaptable to various types of armament and equipment. Thus they have great potential not only as interceptors but for other uses as well.

This report shows these aircraft adapted to combined Interceptor Fighter-Bomber missions. It is divided into three main sections. The first describes the inherent advantages of Delta Wing aircraft. It explains why Delta Wing aircraft are superior to other aircraft for Fighter missions.

The second section shows how by simple modifications F102A's can be given an air-to-ground Fighter-Bomber capability while retaining an effective air-to-air capability. The modifications envolve adding a simple optical bombing system and modifying the aft armament compartment to carry bombs and additional fuel. Provision for carrying missiles in the forward armament compartments and rockets in the armament compartment doors remain unchanged.

In the third section similar changes to F106A type aircraft, to add air-to-ground bombing, are described. The modifications are to the rear half of the armament compartment to permit carrying a variety of bombs while missiles are carried in the forward half. Thus an air-to-air capability is retained.

Included in a final section is a discussion of the F102/F106 aircraft as All Weather Fighter-Bombers attainable because of their tremendous growth potential.

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ADVANTAGES OF F102/F106 DELTA WING AIRCRAFT

Delta Wing aircraft are inherently superior to other aircraft for fighter missions. This is because using the Delta Wing makes it possible to obtain more of the essential major characteristics needed in fighter aircraft than is possible with other wing shapes. There are four major characteristics of an aircraft which should be considered in determining its true combat value. They are:

1. Speed that can be made good in the large variety of missions that will be encountered in actual combat. This speed approaches the cruising speed more closely than the high speed. However, high (maximum) speed is such a spectacular characteristic it tends to obscure others and as a consequence is often given much more weight than is justified. In fact, in some cases, it appears to be used to the exclusion of other characteristics in judging aircraft.

2. The military load that can be carried. The armament must be sufficient in kind and amount to be effective against the target. The weapon control system must be sufficient in kind and amount to ascertain that the weapons will reach the target. These two items comprise the military load. It should be noted that military load rather than useful load is used as one of the determining criteria. Useful load is an arbitrary term including fuel and other items which in no sense can be considered "pay load". Fuel is a means to an end not an end in itself. The less fuel needed to accomplish the mission, the better. Useful load includes armament but excludes the fire control system. The reason for a fighter is to get the armament and its guidance system to the battle zone, therefore the fire control system is a part of military load.

3. Combat Radius - Radius of Action. The combat radius and/or endurance must be sufficient for the aircraft to reach the target zone and conduct combat.

4. Weight - The above must be accomplished at minimum weight. A poor aircraft arrangement or selection of major components which leads to excess weight diminishes the ability of the airplane to accomplish one or more of the above items of its task.

Military threats have been advancing so fast that the speed, military load and combat radius have been desired in unlimited quantities which, of course, cannot be provided. Therefore, the best combat aircraft is that aircraft whiah provides these factors in the greatest quantities when they are provided in proper proportions, and does so at minimum weight.

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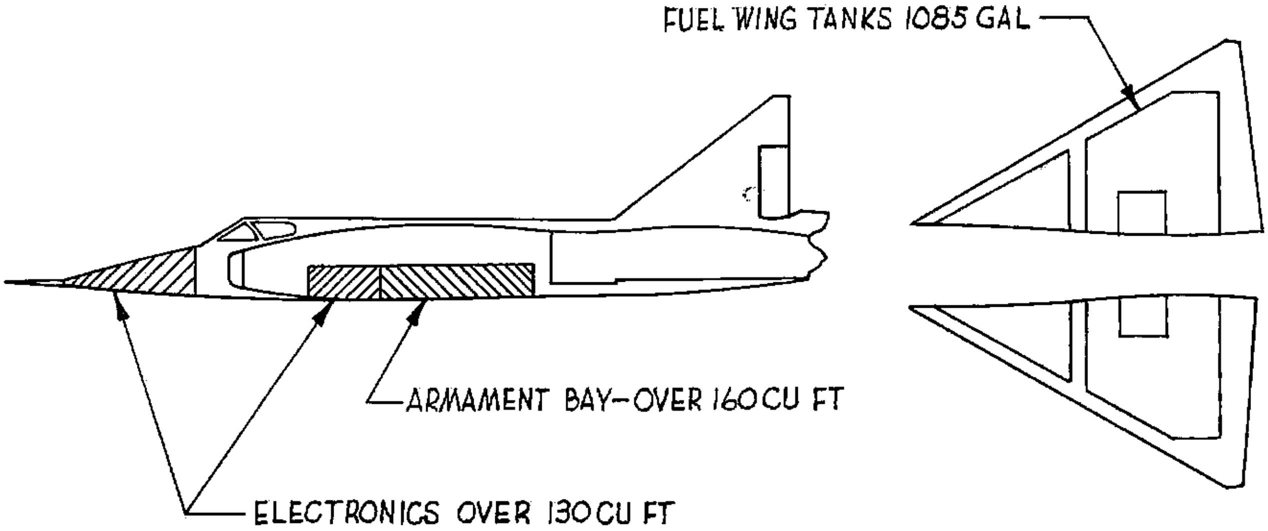
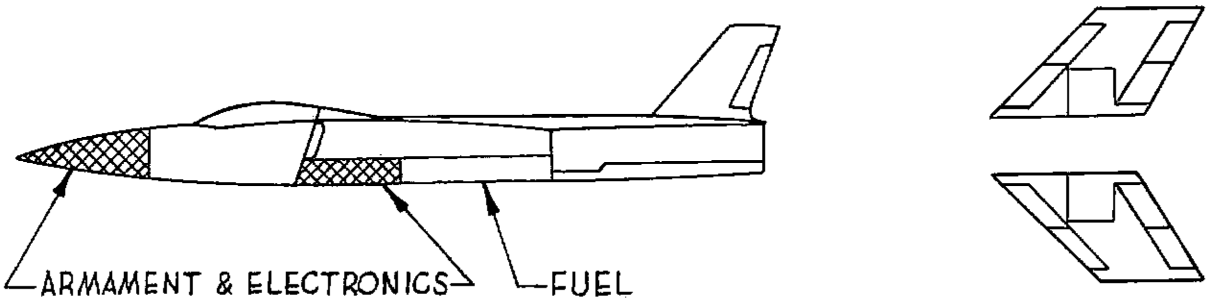
DELTA CAPACITY



F 102 DELTA CONFIGURATION



CONVENTIONAL AIRCRAFT CONFIGURATION

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Delta Wing aircraft have high cruising speeds and good economy (high L/D) throughout the speed range. The high sweepback angle (60°) minimizes the drag rise and causes it to occur at a high critical Mach number. The absence of a horizontal tail eliminates its drag and its associated interference drag. These same factors contribute to high rate of climb, high ceiling and to minimizing fuel used during climb.

The delta wing design has far greater military load carrying capacity per pound of weight than more conventionally arranged aircraft. This is largely due to the fact that in the delta wing design, where the horizontal tail is eliminated, a large wing area (low wing loading) is required. The delta shape, because of its long root chord, provides a thick wing section even through the percentage thickness (thickness/chord ratio) is small. The net result is that there is a large amount of useable space in the wing for minimum cost in weight and drag. This wing space is used for carrying most of the fuel which frees the fuselage space for much needed armament and weapons control equipment.

Present day fire control systems, that provide adequate solutions to the problems imposed upon them, are necessarily complicated and volumnous. Adequate space, properly arranged is required for their installation and servicing. The F102-F106 delta wing aircraft provide such space. They also provide for the ready installation of constantly improving equipment.

A most important result of using a large delta wing rather than a smaller wing with lift augmenting devices is that it provides internal space for most of the fuel. When most of the fuel is carried internally the range is maximized.

In addition to the space afforded by the wing the delta shape provides the most strength where it is most needed, that is, at the wing root. At this section the bending and torsional loads are a maximum. Carrying the fuel load in the wing distributes it better over the lifting surfaces than it would be if carried concentrated in the fuselage with other load items. The net result is that the delta wing design is lighter than other designs.

These characteristics of the delta wing design are inherent and cannot be matched by other designs. As mentioned above they determine the ability of the aircraft to perform its function in the large variety of missions encountered in actual combat. Other things being equal the delta wing aircraft is bound to perform better.

In contrast conventionally arranged aircraft have little or no space in their wings for fuel and must utilize valuable fuselage space for it. The resultant crowding of the fuselage space makes servicing of the equipment in it difficult and makes it necessary to carry armament externally. This degrades performance.

With the exception of one, which is twice the gross weight of the F102, all of the non-delta wing century series aircraft are deficient in one or more of the following:

1. Fire control system seriously limited.
2. Fire power seriously limited.
3. Inadequate space for internal carriage of weapons.
4. Inadequate space for internal carriage of fuel.

In addition to high combat effectiveness the other inherent advantages of F102/F106 Delta Wing aircraft are:

1. Simplicity
2. Ruggedness
3. Good Control & Stability
4. Good Performance other than Speed and Endurance
5. Safety

Simplicity

The sketch on page 6 shows why F102/F106 delta wing aircraft are relatively simple. One set of elevons are used for both longitudinal and lateral control. High lift is obtained by simply using large wing area. No lift augmenting devices are used.

In contrast the conventional type of airplane uses movable ailerons, elevators and stabilizers for longitudinal and lateral control; and flaps perhaps slats also for lift augmentation. The fewer movable control surfaces and their actuating mechanisms lend rigidity and ease of maintenance to F102/F106 Delta Wing aircraft.

Pilots have more freedom with fewer controls to actuate; hence find the delta wing aircraft safer. For example, consider a landing. In F102/F106 aircraft besides the throttle, stick and trim controls the pilot has to actuate only one control, the landing gear. In other aircraft there may be as many as four; the landing gear, flaps, slots and the boundary layer control.

Ruggedness

Supersonic, even high subsonic, aircraft benefit aerodynamically by high wing leading edge sweep-back and low wing thickness ratios. The delta wing shape provides these desirable aerodynamic characteristics and at the same time closely approaches the ideal shape for structural strength and stiffness. High stiffness is desirable to avoid flutter and diminished control effectiveness.

Conventionally arranged aircraft must have strength and stiffness provided through the use of excess structural weight, inadequate wing area, excess thickness ratio or some combination of these.

Control and Stability

The characteristics which give an aircraft good combat effectiveness have been enumerated above. Combat effectiveness can be nullified however unless an aircraft also has good control and stability characteristics. On the basis of data in the pilot's handbooks for U.S. built fighter aircraft, any fair comparison with respect to the following will show the F102/F106 aircraft to be far in the lead.

1. Low speed handling
2. Stall characteristics
3. Immunity from buffeting
4. Stability
5. Immunity from pitch-up
6. Immunity from transonic trim changes

Other Performance Factors

While the speed, military load and combat radius are the principle factors determining the combat effectivenss of aircraft, these must not be achieved at the expense of other important characteristics. The F102/F106 airplanes rate high in:

1. Rate of climb
2. Low landing speed
3. Short take-off and landing distances
4. Power-off glide distance

Safety

Air Force accident records for fighter types show the F102A airplane to be the safest yet built. The major accident rate per thousand flying hours after the first year and half in operational service for F102 airplanes is far below (approximately 2/3) the next best airplane in a like service period. Further the fatalities per thousand hours are far below those of the next best competitive airplane. In addition, the major accident rate forecast for the year 1957-1958 shows F102 aircraft to be below any other of the century series. The exact figures are available in Air Force quarterly accident records.

This demonstrates, at least in part, the inherent safety of the delta wing design. Simplicity yields safety in several ways. First the pilot has fewer controls to operate and think about. His job is easier and he is likely to make fewer mistakes. The delta wing airplane does not "stall" in the normal sense of the word. As a result the few accidents that have occurred on take-off and landing have been less damaging. The inherent ruggedness of the airplane has played a part in reducing fatalities and damage.

The F102/F106 Delta Wing aircraft have more useable space which means better access for inspection and servicing. Units which require it get better service because servicing is easier.

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F102 INTERCEPTOR-FIGHTER BOMBER - DESCRIPTION

In the preceeding pages it was shown that F102/F106 Delta Wing aircraft have inherently high combat effectiveness as compared with more conventionally arranged aircraft. This inherent superiority applies not just to F102/F106 Delta Wing airplanes on interceptor missions but to these aircraft on any fighter mission. The allocation of internal fuselage space, the nose section to the radar and fire control system and the mid-fuselage section to armament is the ideal arrangement for fighter bombers. Accordingly the F102 can be converted to an outstanding Fighter Bomber.

This study shows that the F102 aircraft by relatively simple modifications can be given an air-to-ground bombing capability while retaining an air-to-air capability. Briefly the modifications envolve adding a simple optical bombing system and modifying the aft armament compartment to carry bombs in the center bay and additional fuel in the side bays. The forward armament bays will remain unchanged. Three GAR missiles and the 24 FFAR rockets will be retained.

Although a slight reduction in air-to-air capability would accompany the modifications, improvements in weapons already scheduled for the near future (GAR-Y type) would restore the single pass air-to-air kill capability.

The addition of the air-to-ground capability can be achieved at a small (approximately 4%) increase in combat weight thus allowing a negligible reduction in interceptor strength factors.

The missions of the modified F102 Interceptor-Fighter Bomber would be to intercept and destroy enemy bombers or fighter bombers under all conditions of weather and visibility and make optical air-to-ground attacks on enemy installations in the toss or over-the-shoulder modes of bomb release. For air-to-air attacks it is preferable that operation be in conjunction with ground-alert and ground vectoring equipment but F102s are capable of free lance attacks when warned by any means.

The configuration of the airplane is characterized by a 60° delta wing with trailing edge elevons and a highly swept vertical tail. There is no horizontal tail. The fuselage is featured by a pointed nose radome, "V" shaped windshield, side located engine air inlet ducts, and a waisted midsection.

Data.

Gross Weight	30132	Length	68 ft., 4.68 in.
Wing Area	695 sq.ft.	Span	38 ft., 1.6 in.
Vertical Tail Area	95 sq.ft.	Height	21 ft., 2.53 in.

For air-to-air attacks the airplane has a Hughes fire control system. This system controls the airplane in the pre-attack, attack, armament launching and post attack phases of operation. The radar and fire control system are located in the nose section of the fuselage while the armament is stowed internally in forward bays in the lower mid-fuselage section. The armament bays are equipped with quick-opening doors and displacing mechanisms. 2.75" folding fin rockets are stored in tubes in the armament bay doors.

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For air-to-ground attacks it is proposed to install optical sighting (Labs) equipment. The sight head would displace the present optical sight used with the Hughes equipment. Provisions for armament (bombs) would be made in the aft armament bays in the lower mid-fuselage section.

A Pratt and Whitney J-57 turbojet engine powers the airplane. The engine has an axial flow dual-spool compressor and is equipped with an afterburner. In the normal configuration all the fuel is carried internally. Two 230 gallon dropaway pylon mounted wing tanks can be used to extend the range of the airplane. While carrying these tanks the airplane cruises at high subsonic speeds.

F102A development features include an area rule fuselage with a pressurized cockpit; a fully powered flight-control system which utilizes elevons to provide both aileron and elevator action from conventional cockpit controls; and a tricycle landing gear equipped with an electrically controlled and hydraulically powered nose wheel steering unit.

Air-to-Air System

The F102A airplane, the Hughes fire-control system, and the Falcon and Folding Fin Rocket armament have been integrated into an all weather combat weapon system designed for use in the interception of aerial targets. It is automatic in operation and is capable of attacking airborne threats from any aspect. There are plans to give the system diversity in the form of an infrared mode for supplementing its radar operation during effective countermeasures or under radar silence conditions.

The fire control system represents the most modern operational equipment available for countering light-to-heavy bomber threats. It is all-weather all-aspect equipment designed to search out, detect, identify, and track hostile aircraft and from a lead collision or lead pursuit course launch radar and infrared Falcon missiles and standard 2.75 inch FFAR rockets.

This weapon control system provides the following principal functions:

(a) Utilization of ground-controlled interception (GCI) facilities for directing the interceptor to the target area.

(b) Detection and tracking of targets under all conditions of visibility and weather.

(c) Automatic flight control.

(d) Automatic firing of guided missiles or rockets under conditions leading to a high probability of Kill.

(e) Complete communication, navigation, landing, and identification features.

(f) Broadcast control and free lance operational capabilities.

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In normal operation the interceptor is directed to the target zone by ground control. The target is detected by the interceptor system, selected and locked on by the pilot, and the interceptor is flown automatically to weapon release. After automatic launch of weapons the pilot performs a pull-out maneuver and is returned to base.

The radar subsystem employs a conical scan search and acquisition radar and in conjunction with other subsystems, comprises a complete airborne system which can operate independently of ground control when such facilities are not available or are in-operative. Under these conditions the pilot, warned of the approximate position of enemy aircraft by any means whatever, initiates search for the target and upon acquisition, the interceptor is flown automatically to completion of attack.

Additional weapon system features are anti-chaff tracking circuits, and a pilot tunable radar transmitter frequency. Automatic instrument landing approach system (ailas) functions are accomplished in conjunction with existing Instrument Landing System facilities. The system also includes beacon and ground mapping for use as navigational aids.

Air-to-Ground System

There are a number of optical (Labs) bombing systems available, anyone of which could be utilized in F102 aircraft to provide means of delivering an air-to-ground attack. Most of these systems weigh about the same and utilize the same inputs. Generally they consist of a bombing computor, a simple optical sight and suitable indicators for the pilot. Likewise the various systems compute the same modes of bomb delivery including the toss and over-the-shoulder modes.

For this study the installation of a typical Labs system has been assumed, and the estimates made are based on one of the low altitude types of bomb delivery.

Radio navigational aids already installed in the airplane are the most modern available. In addition the radar ground mapping mode provides excellent support to the other navigational aids during bombing missions.

In normal operation the bombing mission is preplanned. An initial point is selected, the range determined and set into the computor along with trajectory information to match the armament chosen. On approaching the target, the pilot utilizes the optical sight to set up a collision course with target. On passing the initial point, he initiates the bombing computation. At the appropriate time, he is warned to initiate a pull-up and presented a roll angle to hold as a means of making the proper cross wind connection. At the proper pitch angle the bomb is automatically released and an indicator so informs the pilot. The pilot continues on performing an escape maneuver.

In the event of a target of opportunity that shows up on the scope in the ground mapping mode the pilot can set range into the computor corresponding to a range circle on the scope. He then proceeds as above initiating the attack computation when the target crosses the range circle.

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PERFORMANCE
F102 INTERCEPTOR FIGHTER BOMBER

The performance of the F102 Interceptor is summarized on the following pages. The design of the F102A has been changed several times during the course of its development and production program to take advantage of component improvements. Recently however the J-57 engine has been undergoing improvements to greatly increase its thrust. These improvements came too late to permit incorporating them into F102A airplanes as their production was being displaced by F106A airplane production.

Convair has proposed the installation of the JT3C-22 (advanced J57) engine in F102 airplanes. This engine will be in production in early 1961 and can be installed on a modification basis. At the same time it has been proposed that F102 wing leading edges be reworked to achieve a drag reduction. Column II in the Summary Table below shows the performance available through these modifications.

ITEM		F102 J57-P23 ENGINE	F102 JT3C-22 ENGINE
Max Rated Engine Thrust @ Sea Level	LB.	16,000	18,000
Take-Off Wt	LB.	30,132	29,366
Combat Wt	LB.	25,750	25,100
Fuel-Internal (1285 Gal.)	LB.	8,353	8,353
V-Max A/B Power @ 35,000 Ft	MACH.	1.22	1.60
Absolute Ceiling	FT.	55,000	55,000
Time to Climb A/B Power to 40,000 Ft.	MIN.	4.4	3.6
Take-Off Over 50' Obstacle	FT.	4,150	3,340
Land Over 50' Obstacle with Chute	FT.	3,750	3,650
Combat Radius (Area Intercept Mission)	N.Mi.	585	575

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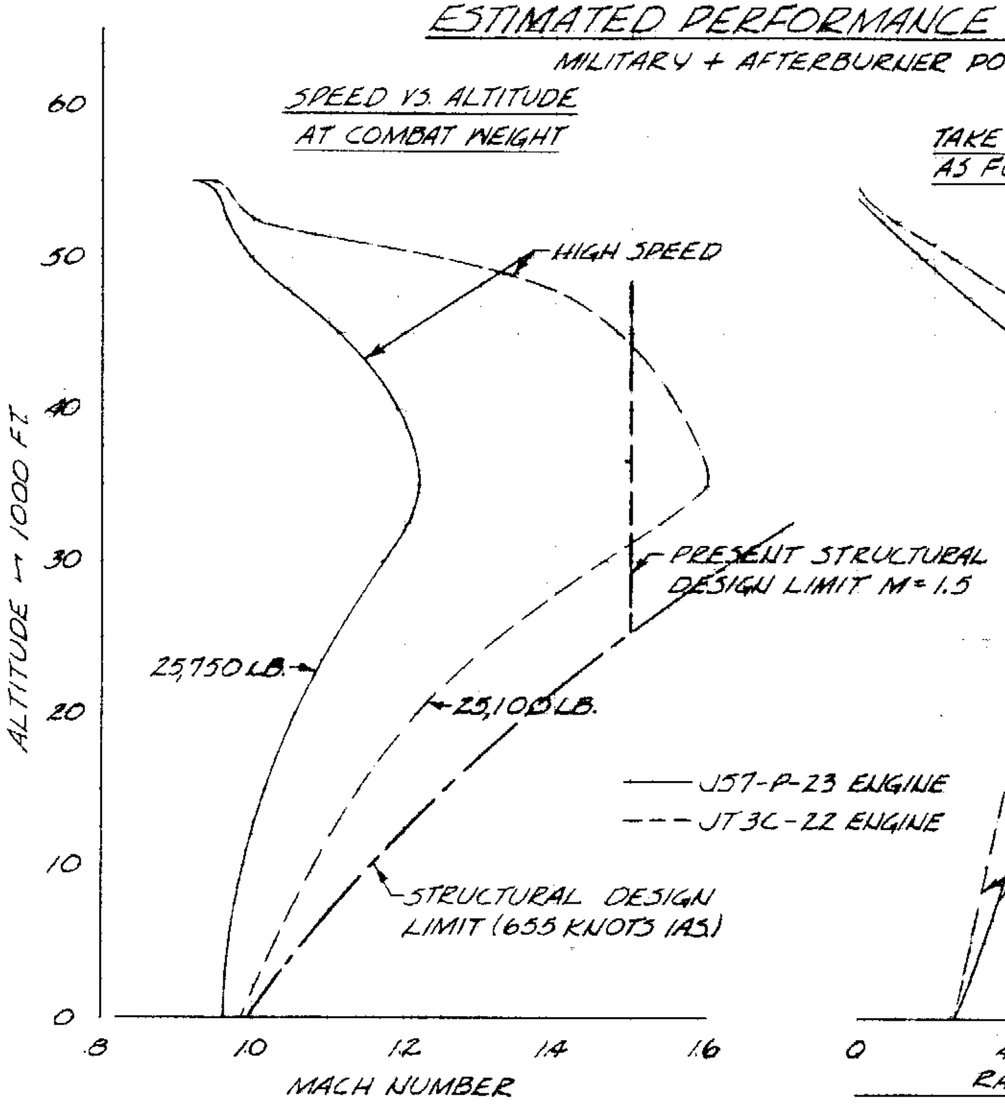
F-102 AIRPLANE

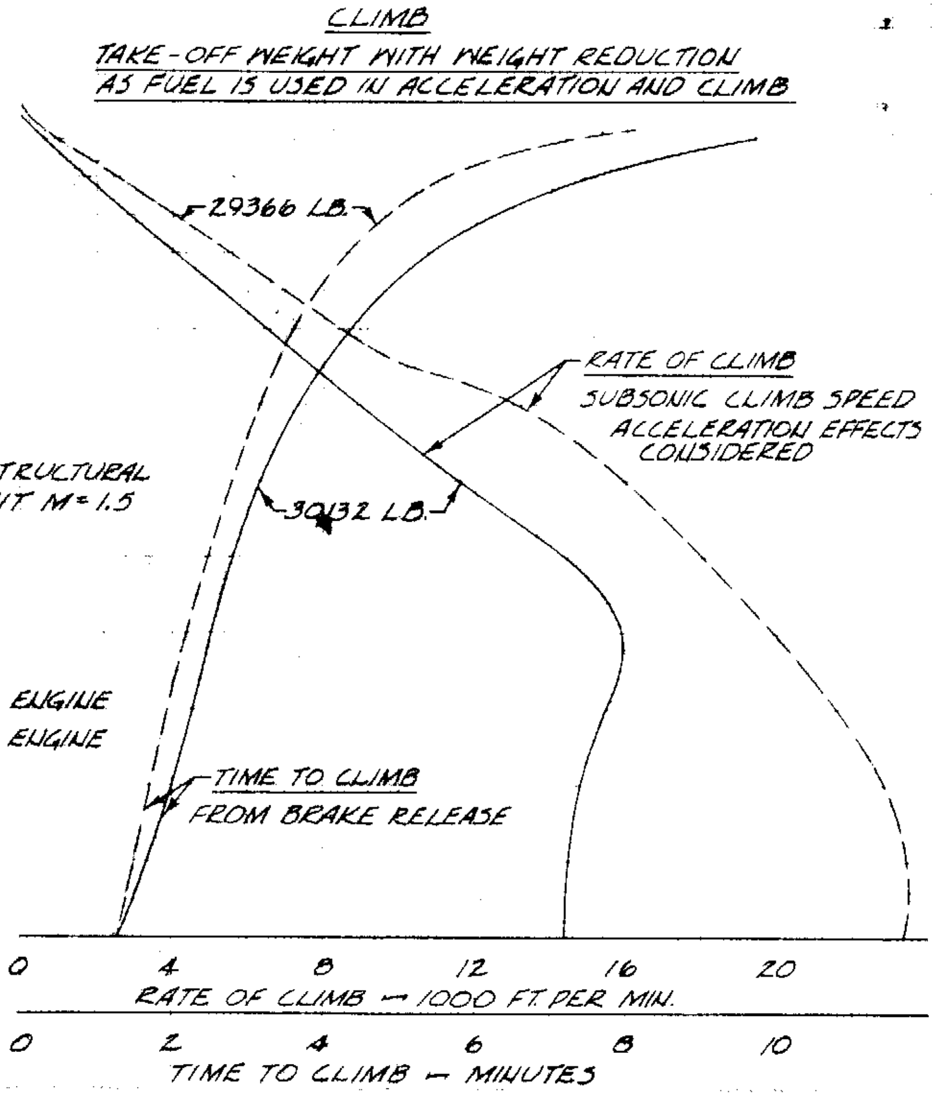
FIGHTER BOMBER

ESTIMATED PERFORMANCE SUMMARY

MILITARY + AFTERBURNER POWER



SPEED VS. ALTITUDE
AT COMBAT WEIGHT



CLIMB

TAKE-OFF WEIGHT WITH WEIGHT REDUCTION
AS FUEL IS USED IN ACCELERATION AND CLIMB

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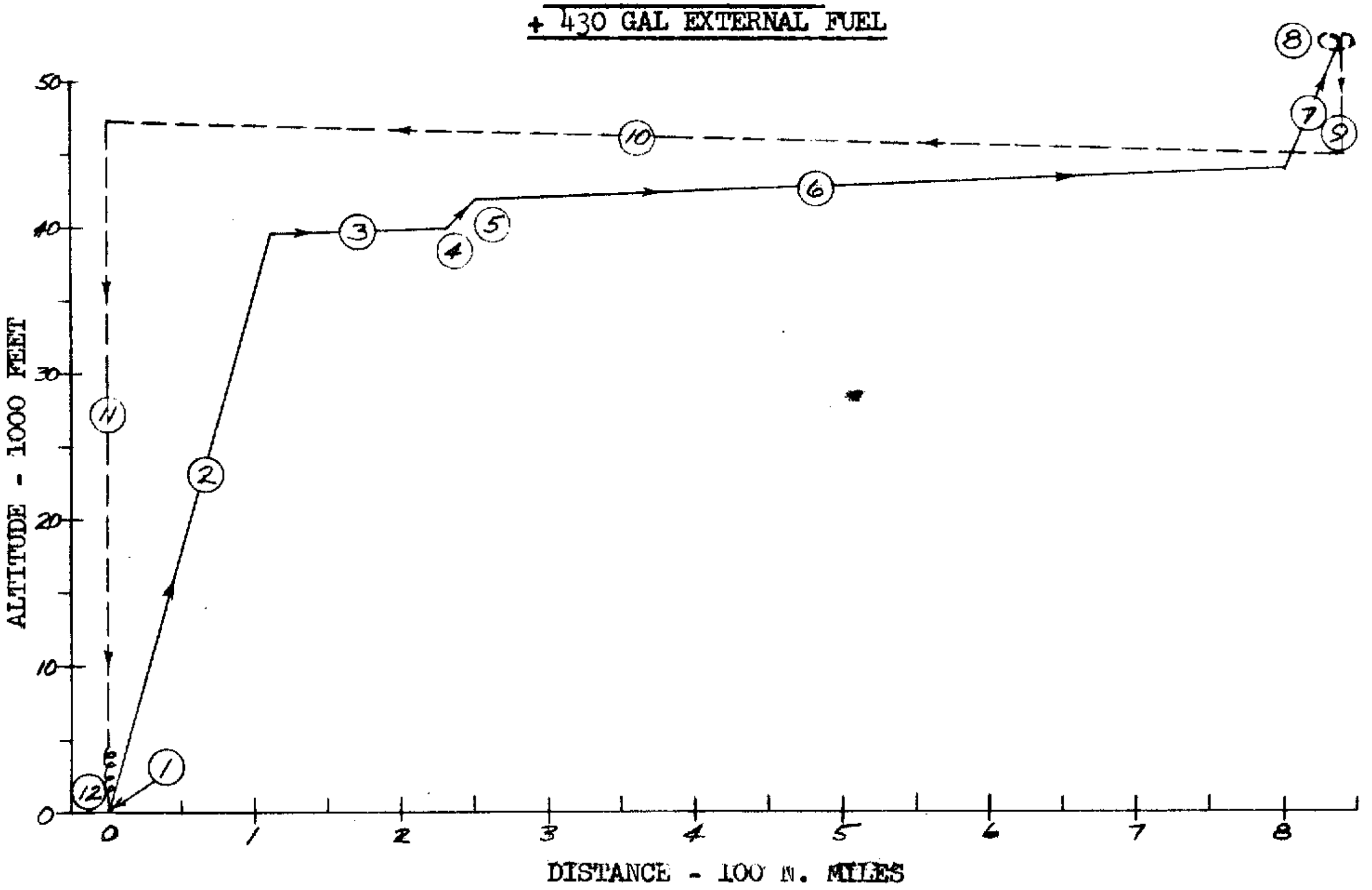
F-102
MIL-C-5011A AREA INTERCEPT MISSION
ALTITUDE VS. DISTANCE
INTERNAL FUEL LOAD
+ 430 GAL EXTERNAL FUEL



(1) Warm-up, take-off and accelerate to climb speed at sea level (Fuel allowance: 2 minutes normal rated power, plus 1 minute maximum power).
(2) Military power climb to optimum cruise altitude with external fuel tanks.
(3) Optimum cruise with external fuel tanks (M = 0.9).
(4) Drop external fuel tanks.
(5) Military power climb to clean airplane optimum cruise altitude.
(6) Optimum cruise, clean (M = 0.9).
(7) Maximum power climb to combat ceiling. +
(8) Combat ++
(9) Descend to optimum cruise altitude (No distance credit).
(10) Optimum cruise home (M = 0.9).
(11) Descend to sea level (No distance credit).
(12) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

+ Combat ceiling is altitude corresponding to R/C = 500 ft/min with maximum power.

++ Combat fuel based on 5 minutes maximum power acceleration at 50,000 ft altitude.

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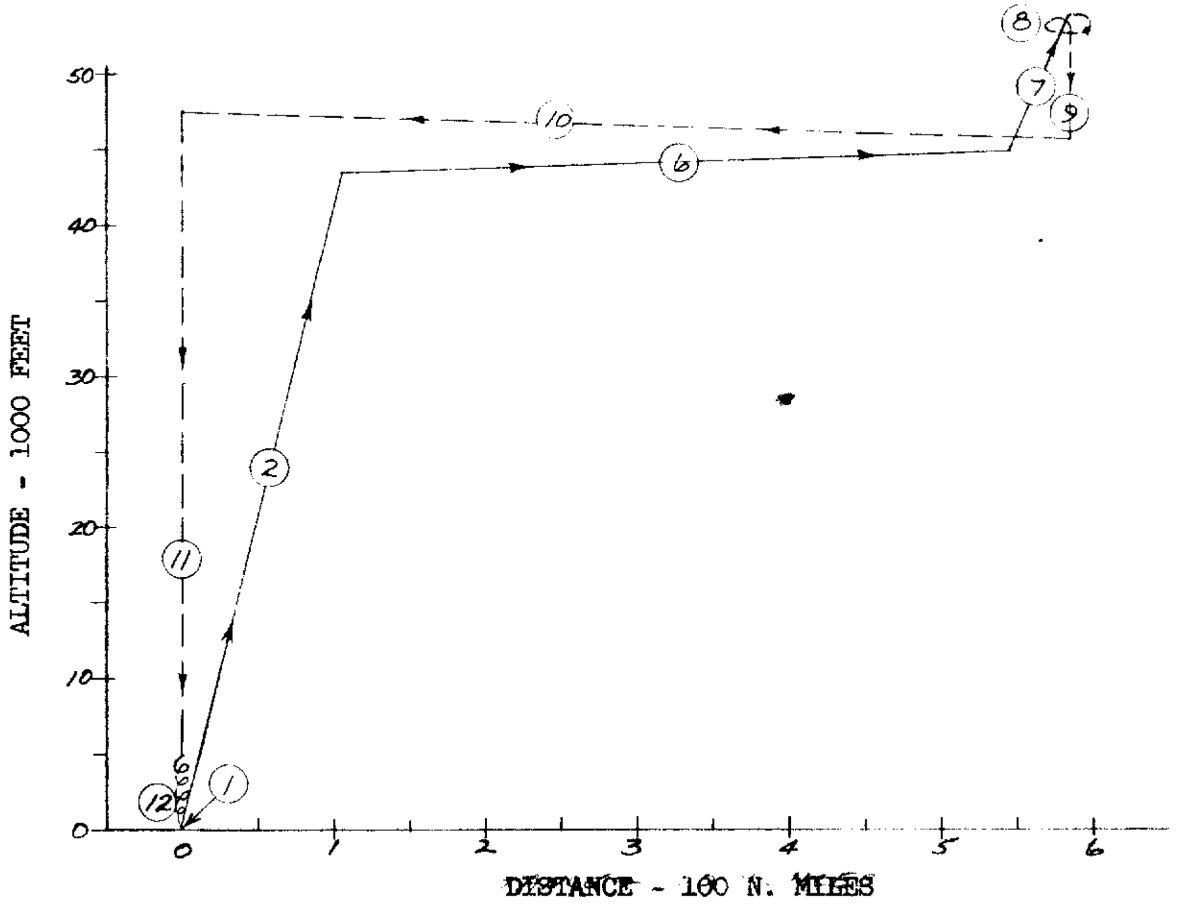
F-102
MIL-C-5011A AREA INTERCEPT MISSION
ALTITUDE VS. DISTANCE
INTERNAL FUEL LOAD



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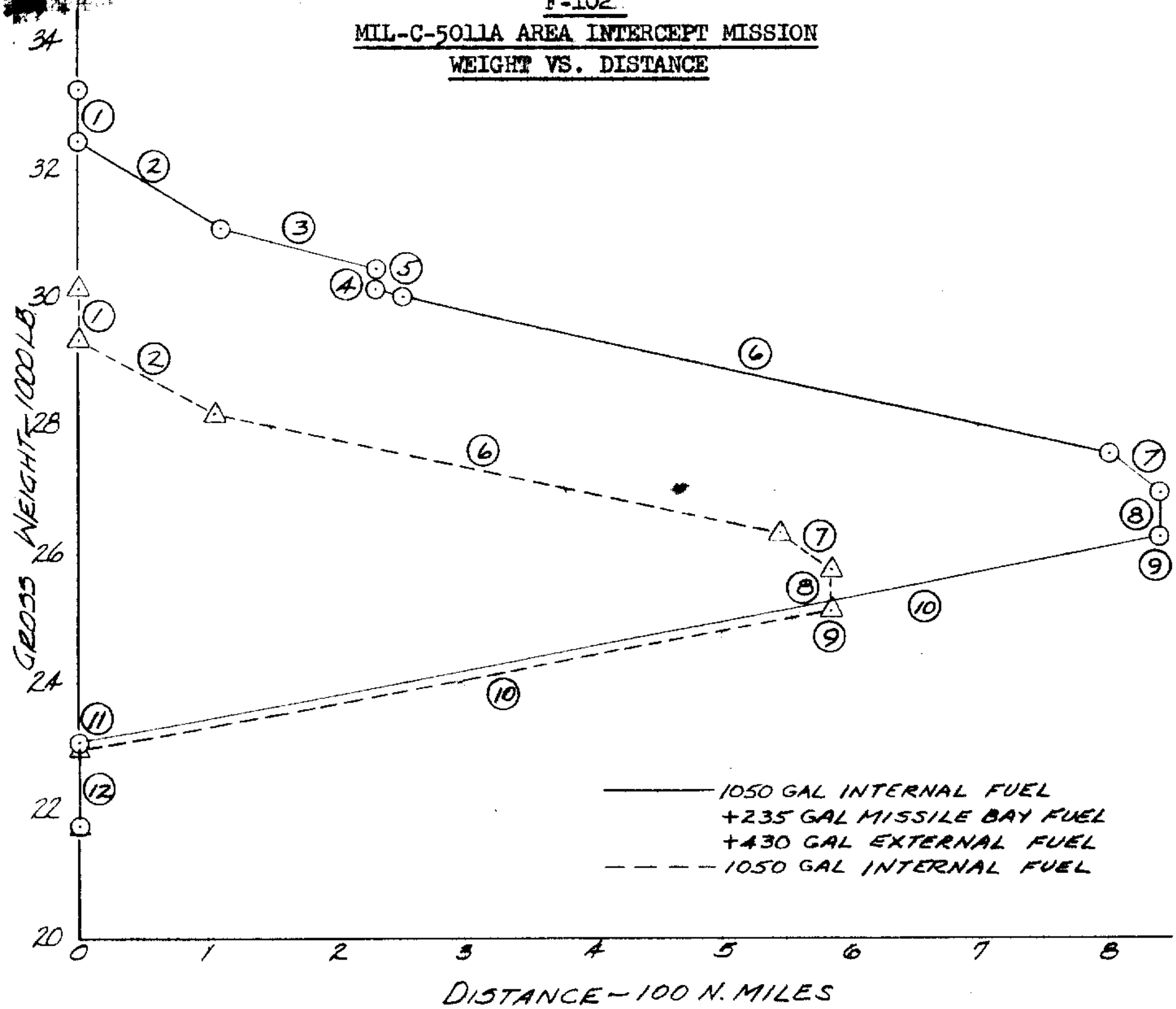
(12) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

+ Combat ceiling is altitude corresponding to R/C = 500 ft/min with maximum power.

++ Combat fuel based on 5 minutes maximum power acceleration at 50,000 ft altitude.

F-102

MIL-C-5011A AREA INTERCEPT MISSION
WEIGHT VS. DISTANCE



(1) Warm-up, take-off and accelerate to climb speed at sea level (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power).
(2) Military power climb to optimum cruise altitude with external fuel tanks.
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(5) Military power climb to clean airplane optimum cruise altitude.
(6) Optimum cruise, clean (M = 0.9).
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(8) Combat ++
(9) Descend to optimum cruise altitude (No distance credit).
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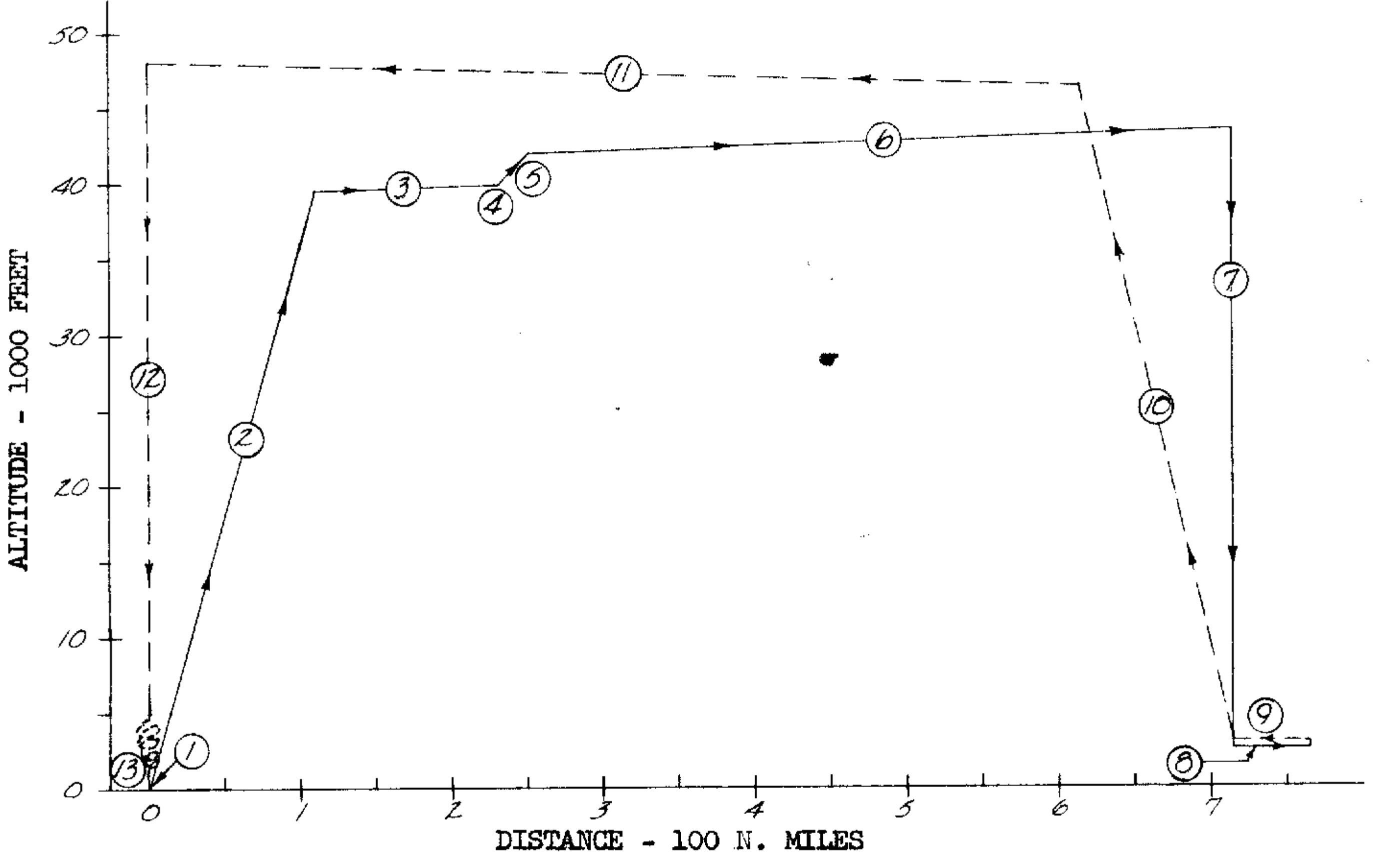
(12) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

+ Combat ceiling is altitude corresponding to R/C = 500 ft/min with maximum power.

++ Combat fuel based on 5 minutes maximum power acceleration at 50,000 ft altitude.

F-102 FIGHTER-BOMBER MISSION
LOW ALTITUDE RUN-IN
ALTITUDE VS. DISTANCE
INTERNAL FUEL LOAD
+ 430 GAL EXTERNAL FUEL



(1) Warm-up, take-off and accelerate to climb speed at sea level (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power).
(2) Military power climb to optimum cruise altitude.
(3) Optimum cruise with external fuel tanks (M = .90).
(4) Drop external fuel tanks.
(5) Climb to clean airplane optimum cruise altitude.
(6) Optimum cruise (M = .90).
(7) Descend to sea level, no distance credit.
(8) 50 nautical mile military power run-in (M = .88). Drop weapon.
(9) 50 nautical mile military power run-out (M = .88).
(10) Military power climb to optimum cruise altitude.
(11) Optimum cruise home (M = .90).
(12) Descend to sea level, no distance credit.
(13) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

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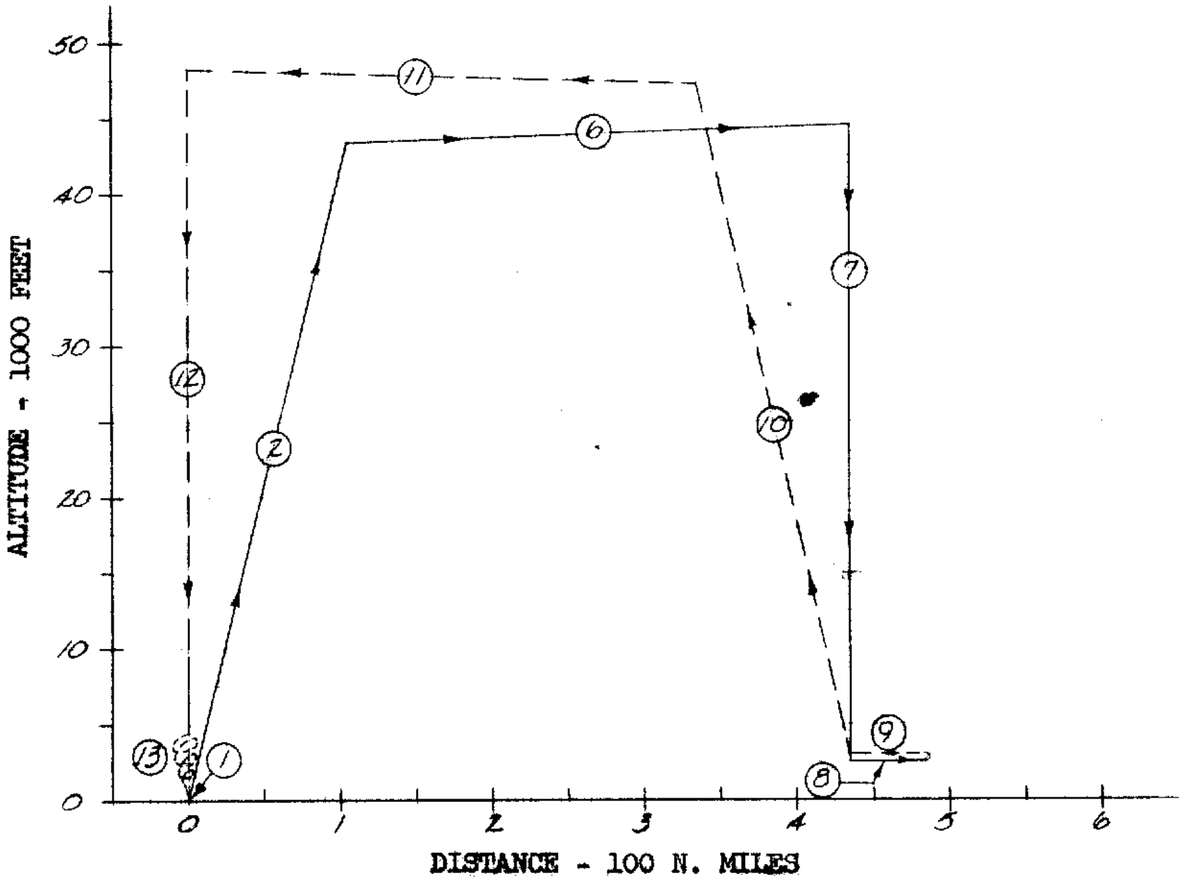
F-102 FIGHTER-BOMBER MISSION
LOW ALTITUDE RUN-IN
ALTITUDE VS. DISTANCE
INTERNAL FUEL LOAD



(1) Warm-up, take-off and accelerate to climb speed at sea level (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power).
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(9) 50 nautical mile military power run-out (M = .88).
(10) Military power climb to optimum cruise altitude.
(11) Optimum cruise home (M = .90).
(12) Descend to sea level, no distance credit.
(13) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

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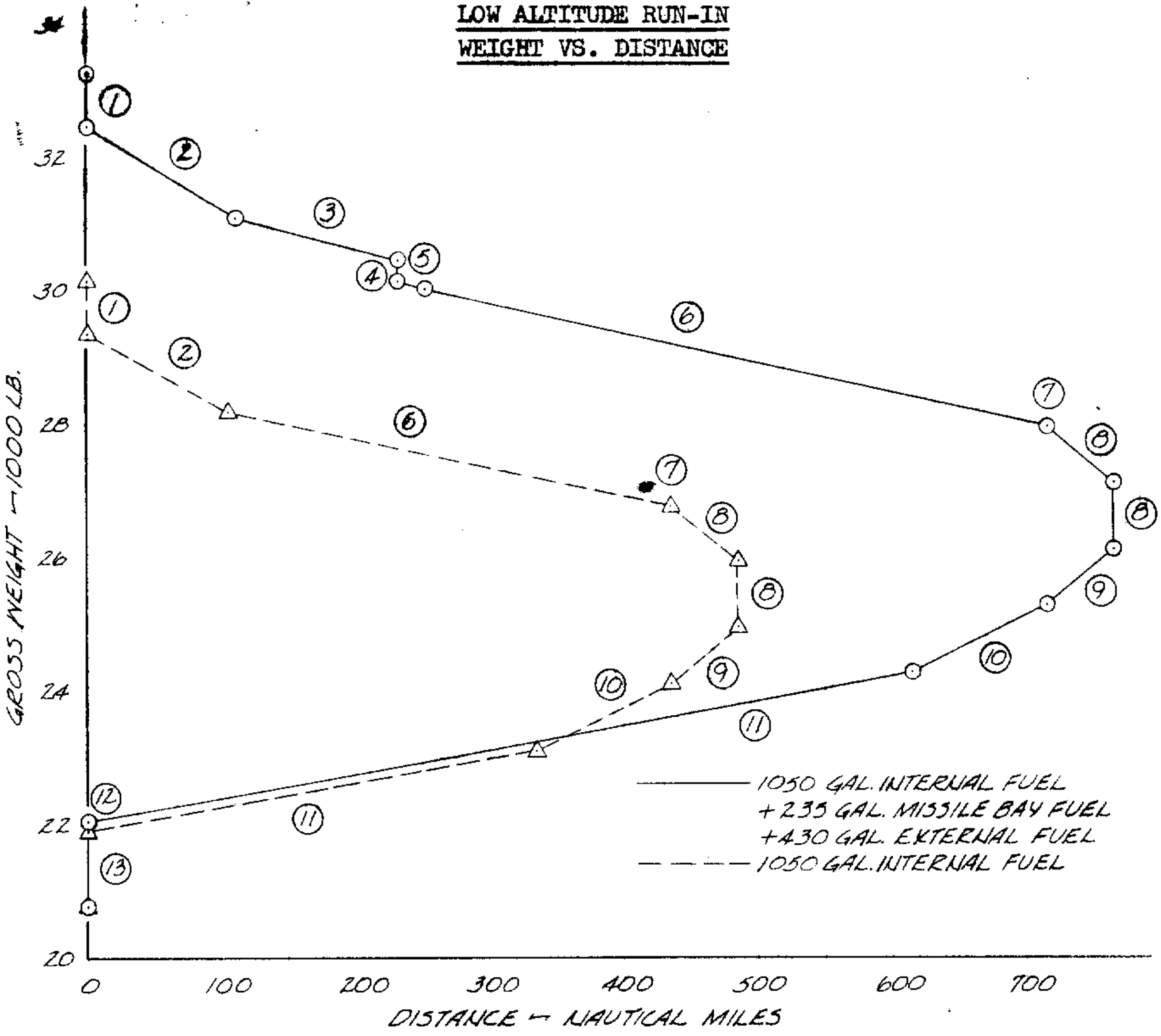
F-102 FIGHTER-BOMBER MISSION
LOW ALTITUDE RUN-IN
WEIGHT VS. DISTANCE



(1) Warm-up, take-off and accelerate to climb speed at sea level (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power.)
(2) Military power climb to optimum cruise altitude.
(3) Optimum cruise with external fuel tanks (M = .90).
(4) Drop external fuel tanks.
(5) Climb to clean airplane optimum cruise altitude.
(6) Optimum cruise (M = .90).
(7) Descend to sea level, no distance credit.
(8) 50 nautical mile military power run-in (M = .88). Drop weapon.
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(11) Optimum cruise home (M = .90).
(12) Descend to sea level, no distance credit.
(13) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

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ALTERNATE ARMAMENT

Typical loadings of alternate armament that can be carried internally are shown on the following drawing.

Although not shown, the MB-1 air-to-air rocket has been ejected from both the F102 and F106 airplanes. This rocket when released at the proper angle at high altitude has a range of well over one hundred miles. Such performance offers possibilities for air-to-ground action not yet evaluated.

However, it is suggested that the warhead only in the MB-1 rocket, or another of approximately the same size could be repackaged and fused to make an excellent weapon for air-to-ground action. It is further suggested that giving such a weapon a bluff shape geometrically similar to the TX 28 would give it equivalent stability so that it could be mounted internally in place of the standard 1000 pound general purpose bomb.

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F-102 INTERCEPTOR FIGHTER-BOMBER

WEIGHT SUMMARY

Item	Lbs.
Structure	7,757
Landing Gear	1,075
Propulsion	6,451
Instruments	149
Hydraulic & Pneumatic	285
Electrical	583
Electronics	2,073
Armament Provisions	508
Furnishings and Equipment	310
Air Conditioning and Anti-Icing	202
WEIGHT EMPTY	19,393
Crew (1)	230
Fuel	8,358
Bombs (1)	1,000
Missiles (3)	381
Rockets (24) 2.75"	432
Other	338
GROSS WEIGHT	30,132
External Tanks and Fuel	3,120
GROSS WEIGHT - LONG RANGE	33,252

Based on ZD8-003A including current change notices.

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F106 INTERCEPTOR - FIGHTER BOMBER - DESCRIPTION

The fact that F102/F106 Delta Wing aircraft are inherently more effective than more conventionally arranged aircraft for fighter missions has been brought out in Section I. This inherent superiority applies not just to interceptor missions but to any fighter mission. The allocation of internal fuselage space, the nose section to the radar and fire control system and the mid-fuselage section to armament is the ideal arrangement for fighter bombers. Accordingly the F106 can be converted to an outstanding Fighter-Bomber.

A first step in such a conversion could be relatively simple modifications to give the F106 an air-to-ground bombing capability while simultaneously retaining an air-to-air capability. The modifications would envolve adding a simple optical bombing system and modifying the armament compartment to carry air-to-air missiles in the forward half while carrying air-to-ground armament(bombs) in the rear half.

The slight reduction in air-to-air capability caused by removal of air-to-air armament would be restored by improvements in weapons already planned for the near future.

A small (approximately 5%) increase in combat weight would accompany the addition of the air-to-ground capability. Thus Interceptor strength factors would be retained with negligible reduction.

With the modifications mentioned the F106 would be capable of intercepting and destroying enemy bombers or fighter bombers under all conditions of weather and visibility and making optical air-to-ground attacks on enemy installations in the toss or over-the-shoulder modes of bomb release. For air-to-air attacks it would be preferable to operate in conjunction with ground alert and ground vectoring equipment but F106s are capable of free lance attacks from any kind of ground alert warning.

In designing the F106 airplane, advantage has been taken of developments in its components to improve almost every feature and phase of its operations over those of the F102.

The Pratt & Whitney J75, a larger jet engine, affords improved performance. More fuel is provided to increase range. The Hughes MA-1 electronic equipment provides improved radar range, countermeasures, navigation and better (digital) computations.

The F106, like the F102 has a 60° delta wing with trailing edge elevons and a highly swept vertical tail. There is no horizontal tail. The fuselage has a pointed nose radome, and "V" shaped windshield. The side located engine air inlet ducts start further back on the waisted midsection than those on the F102.

Data

Gross Weight	36656	Length	70 ft. 8.8 in.
Wing Area	697.8 sq. ft.	Span	38 ft. 1.6 in.
Vertical Tail Area	105.0 sq. ft.	Height	20 ft. 3.3 in.

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The MA-1 radar antenna, transmitter-receiver unit and most of the fire control system components are located in the nose section of the fuselage while most of the landing, communication and navigation components are located in an electronic compartment aft of the cockpit.

The air-to-air armament (missiles) are stowed internally in the forward armament bay in the mid-fuselage section. The armament bays are equipped with quick opening doors and displacing mechanism.

For air-to-ground attacks it is proposed to install optical sighting (Labs) equipment. The sight head would displace the optical sight head used in the Interceptor with the Hughes equipment. Bombs would be carried in the aft armament bay in the lower mid-fuselage section.

A Pratt & Whitney J75 turbojet engine powers the F106 airplane. The engine has an axial flow dual spool compressor and is equipped with an afterburner. The use of this engine provides a ceiling and maximum speed for the airplane greater than those of the F102.

The F106 normally carries its fuel internally. But like the F102 it has two 230 gallon droppable pylon mounted wing tanks to extend the range. The airplane cruises at high subsonic speeds when carrying external fuel tanks.

F106 features include "area rule" fuselage, with a pressurized cockpit; a fully powered flight control system which utilizes elevons to provide both aileron and elevator action from conventional cockpit controls; controlled engine air inlets; and a tricycle landing gear equipped with an electrically controlled and hydraulic powered nose wheel steering unit.

Air-to-Air System

The Hughes MA-1 electronic system and Falcons missiles (FAR 3 and 4) have been integrated into the F106 airplane with even more refinement than the corresponding items were integrated into the F102. The resulting all Weather combat weapon system is superior in every respect. It is automatic in operation and capable of attacking airborne threats from any aspect. The system can be given diversity in the form of an infra red mode for supplementing its radar operation during effective countermeasures or under radar silence conditions. When used for air-to-air attacks, it controls the airplane in the pre-attack, attack, armament launching and post attack phases of operation.

Like the Hughes MG-10 electronic equipment in the F102, the MA-1 system is all weather all-aspect equipment designed to search out, detect, identify and track hostile aircraft. It automatically launches radar and infra red missiles from a lead collision or lead pursuit course.

Its radar has increased power and an added moving target indication feature which improves its search and track capability. Its broadcast control and free lance operations are improved by an added inertial navigator. Improved computations are afforded by the use of a digital computor as opposed to an analogue

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computor in the MG-10.

The principal functions provided by the weapon control system are:

(a) Directs the interceptor to the target area utilizing ground control intercept facilities (GCI).

(b) Detection and tracking of targets under all conditions of visibility and weather.

(c) Automatic flight control.

(d) Automatic firing of weapons under conditions leading to a high probability of Kill.

(e) Complete communications, navigation, landing and identification features.

(f) Broadcast control and free lance operational capabilities.

Normally the F106 is directed to the target area by ground control. The target is detected by the weapon control system, selected and locked on by the pilot. The airplane is then flown automatically to weapon release. The pilot performs a pull-out maneuver and returns to base after automatic launch of weapons.

The radar subsystem in conjunction with other subsystems, comprises a complete airborne system which can operate independently of ground control when such facilities are not available or are inoperative. Warned of the approximate position of enemy aircraft by any means whatever, the pilot initiates search for the target, and upon acquisition, locks-on where-upon the interceptor is flown automatically to completion of the attack.

For counter counter measures the weapons control system has anti-chaff tracking circuits, a pilot tunable radar frequency and angle tracking of continuous wave jammers. Other functions included in the system are automatic instrument landing approach and beacon and ground mapping radar navigational modes.

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PERFORMANCE F106 INTERCEPTOR FIGHTER-BOMBER

The performance of the F106 airplane as an Interceptor Fighter-Bomber is summarized in the following table and on following pages.

Item		
Max Rated Engine Thrust @ Sea Level	LB.	24,500
Take-Off Weight	LB.	36,656
Combat Weight	LB.	31,750
Fuel, Internal (1530 Gal.)	LB.	9,945
V-Max A/B Power 35000 Ft	MACH.	2.0*
Absolute Ceiling	FT.	53,500
Time to Climb, A/B Power, to 40000 Ft	MIN.	3.1
Take-Off Over 50' Obstacle	FT.	4,600
Land Over 50' Obstacle with Chute	FT.	4,900
Combat Radius (Area Intercept Mission)	N.M.	455

*Structural Limit Speed

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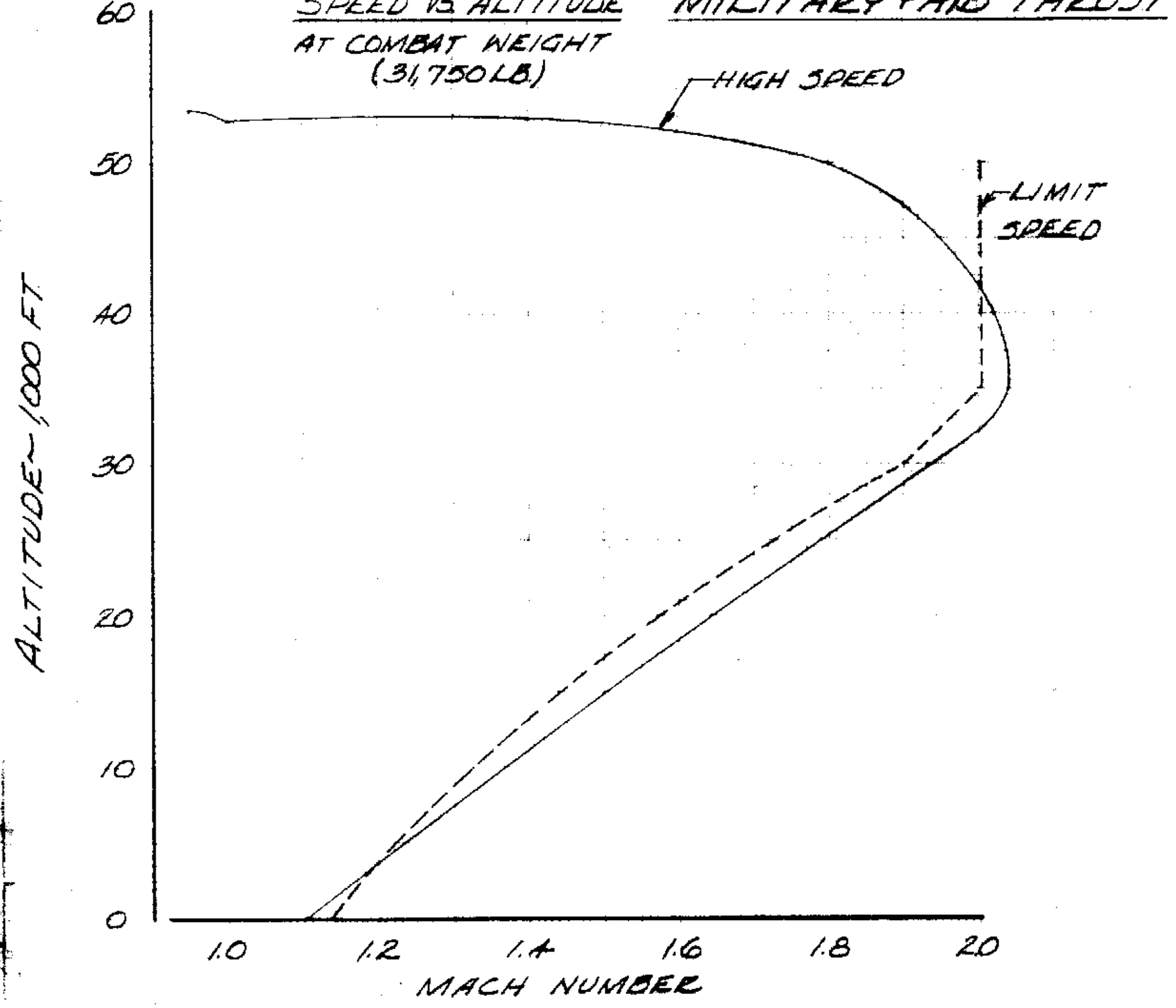
F-106 AIRPLANE

FIGHTER BOMBER

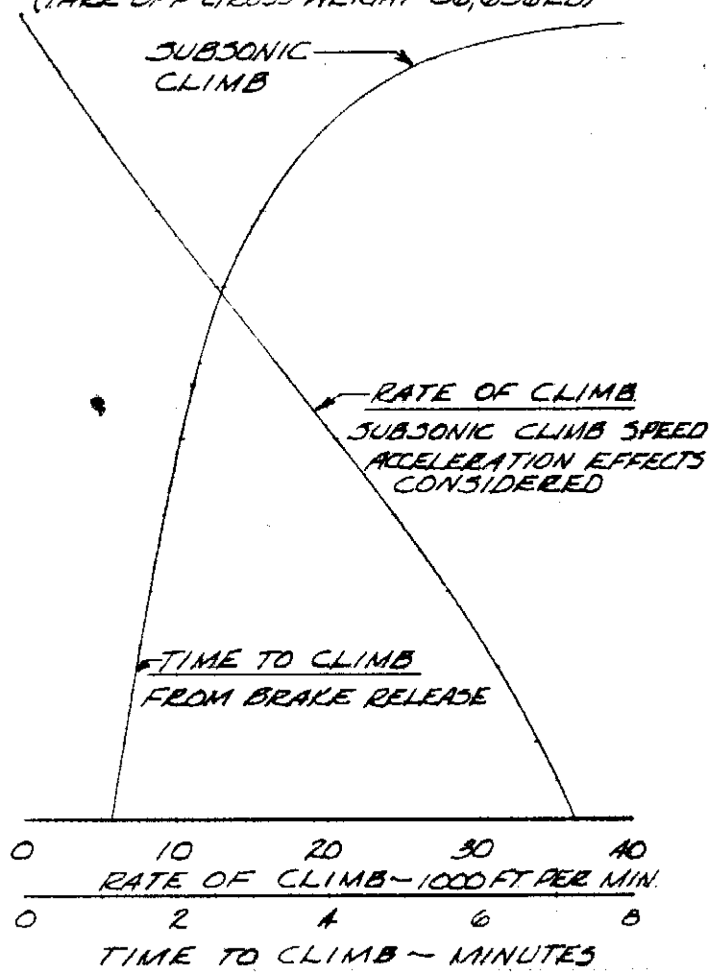
ESTIMATED PERFORMANCE SUMMARY

MILITARY + A/B THRUST

SPEED VS ALTITUDE
AT COMBAT WEIGHT
(31,750 LB.)



CLIMB
TAKE-OFF WEIGHT WITH WEIGHT REDUCTION AS FUEL IS USED IN ACCELERATION AND CLIMB.
(TAKE-OFF GROSS WEIGHT 36,656 LB)



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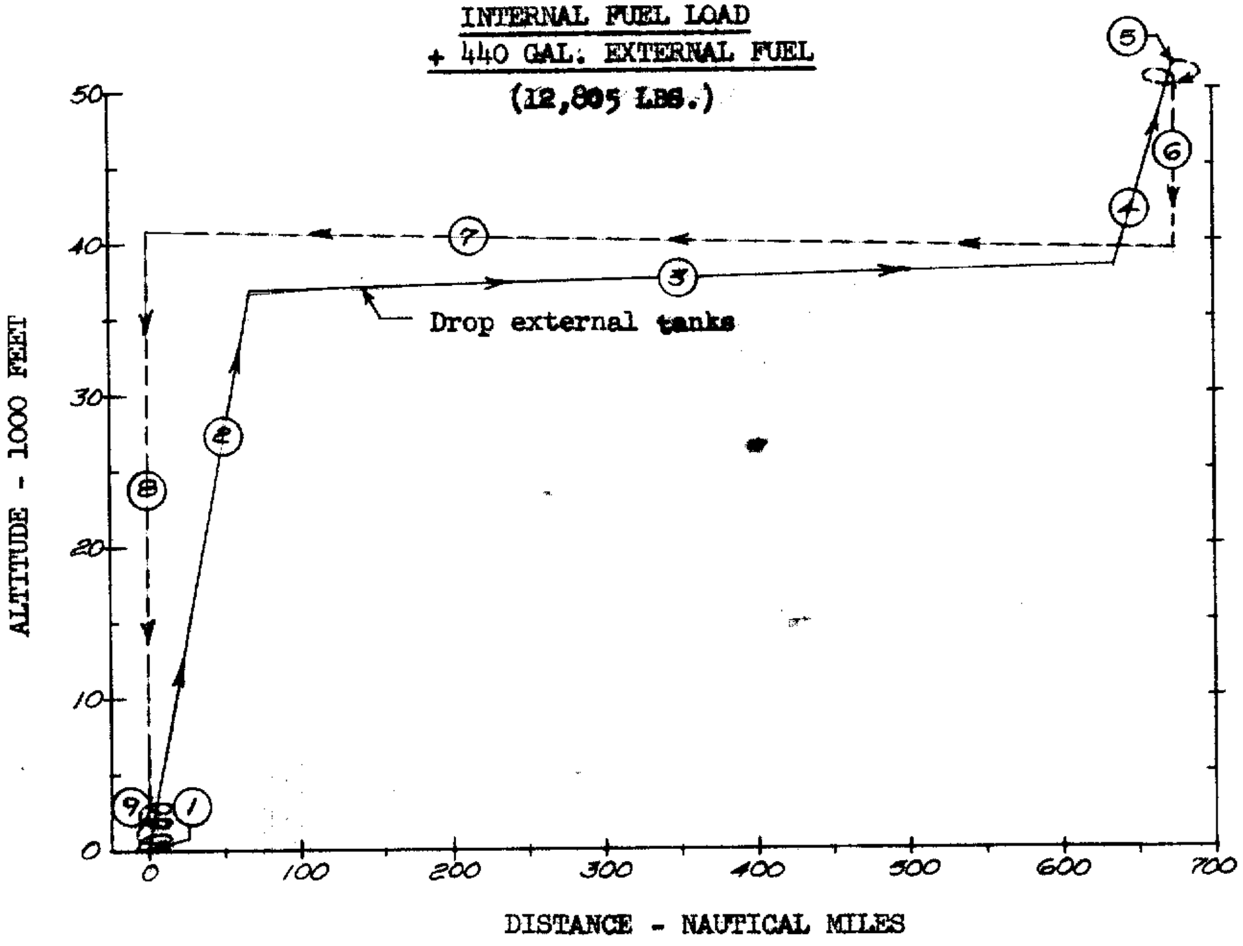
F-106
MIL-C-5011A AREA INTERCEPT MISSION

INTERNAL FUEL LOAD
+ 440 GAL. EXTERNAL FUEL
(12,805 LBS.)



(1) Warm-up, take-off and accelerate to climb speed at sea level (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power).
(2) Military power climb to optimum cruise altitude.
(3) Optimum cruise (M = .925).
(4) Maximum power climb to combat ceiling. +
(5) Combat ++
(6) Descend to optimum cruise altitude (No distance credit).
(7) Optimum cruise home (M = .925).
(8) Descend to sea level (No distance credit).
(9) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

+ Combat ceiling is altitude corresponding to R/C = 500 ft/min with maximum power.

++ Combat fuel based on 5 minutes maximum power acceleration at 50,000 ft altitude.

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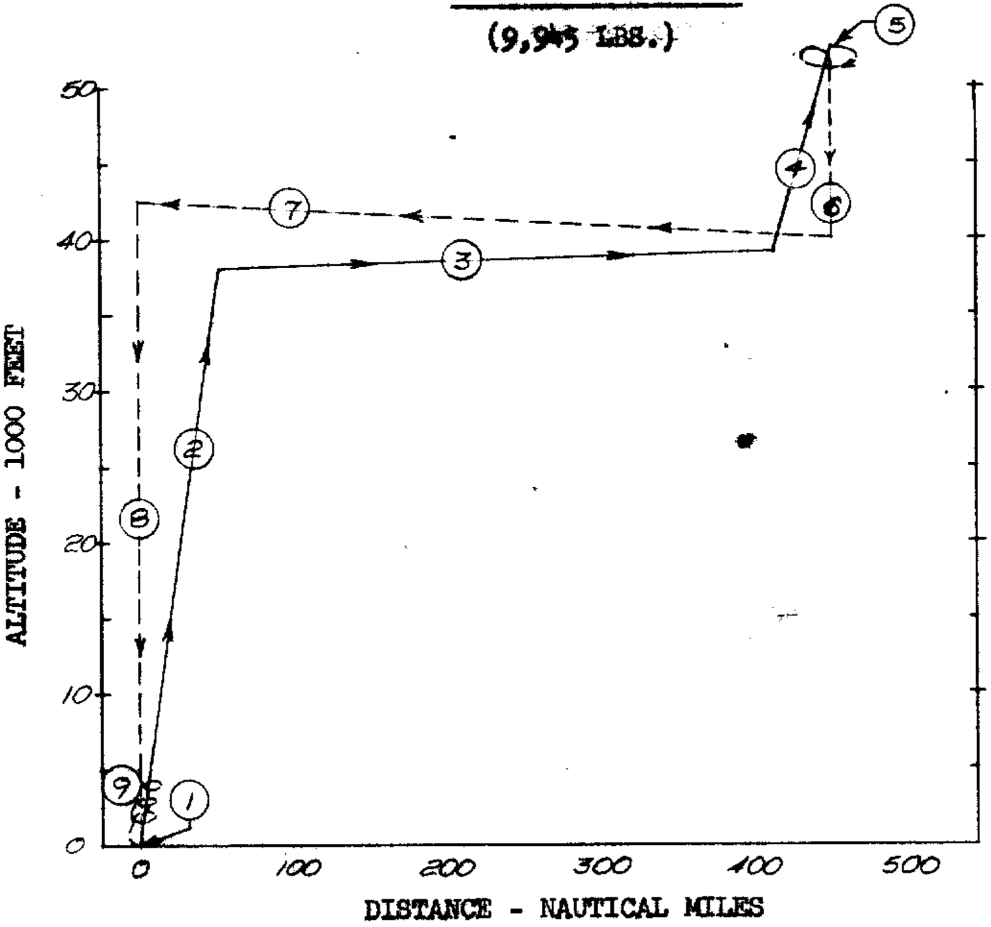
F-106
MIL-C-5011A AREA INTERCEPT MISSION

INTERNAL FUEL LOAD
(9,945 LBS.)



(1) Warm-up, take-off and accelerate to climb speed at sea level (Fuel allowance: 2 minutes normal rated power, plus 1 minute maximum power).
(2) Military power climb to optimum cruise altitude.
(3) Optimum cruise (M = .925).
(4) Maximum power climb to combat ceiling. +
(5) Combat ++
(6) Descend to optimum cruise altitude (No distance credit).
(7) Optimum cruise home (M = .925).
(8) Descend to sea level (No distance credit).
(9) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

+ Combat ceiling is altitude corresponding to R/C = 500 ft/min with maximum power.

++ Combat fuel based on 5 minutes maximum power acceleration at 50,000 ft altitude.

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F-106
MIL-C-5011A AREA INTERCEPT MISSION
WEIGHT VS. DISTANCE

(1) Warm-up, take-off and accelerate to climb speed at sea level (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power).

(2) Military power climb to optimum cruise altitude.

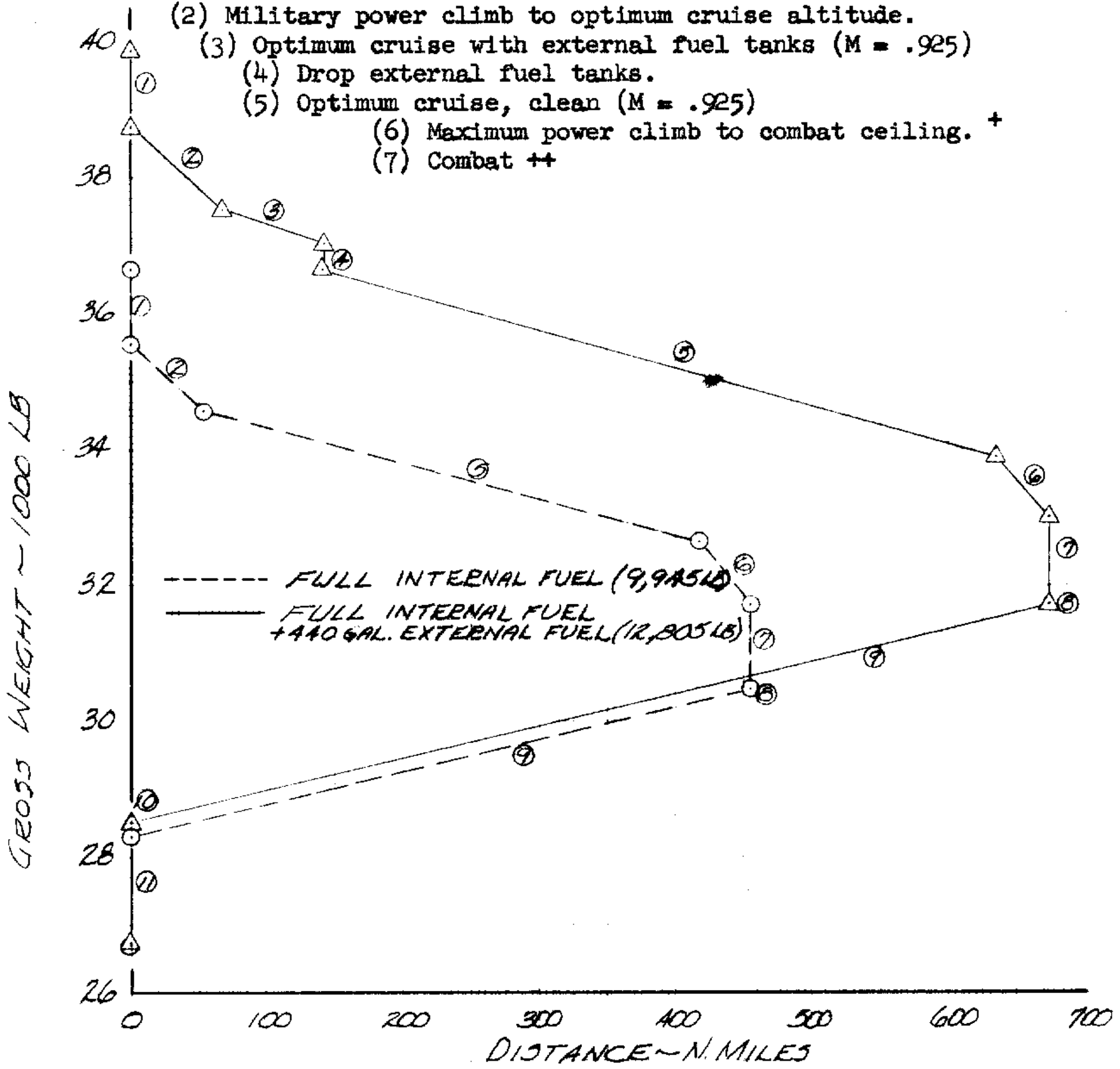
(3) Optimum cruise with external fuel tanks (M = .925)

(4) Drop external fuel tanks.

(5) Optimum cruise, clean (M = .925)

(6) Maximum power climb to combat ceiling. +

(7) Combat ++



(8) Descend to optimum cruise altitude (No distance credit).

(9) Optimum cruise home (M = .925).

(10) Descend to sea level (No distance credit).

(11) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

+ Combat ceiling is altitude corresponding to R/C = 500 ft/min with maximum power.

++ Combat fuel based on 5 minutes maximum power acceleration at 50,000 ft. altitude.

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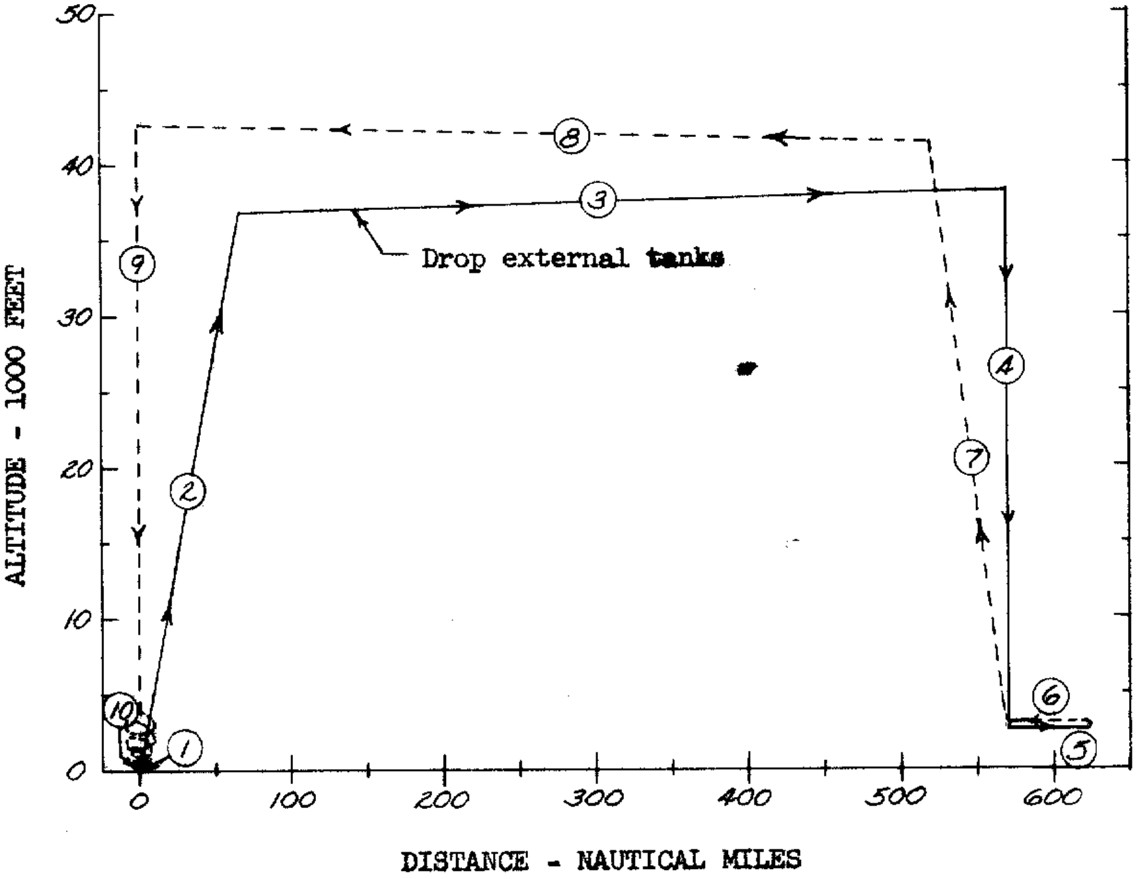
F-106 FIGHTER-BOMBER MISSION
LOW ALTITUDE RUN-IN

INTERNAL FUEL LOAD
+ 440 GAL. EXTERNAL FUEL
(12,805 LBS.)



(1) Warm-up, take-off and accelerate to climb speed at sea level. (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power).
(2) Military power climb to optimum cruise altitude (average velocity = 500 knots).
(3) Optimum cruise (M = .925).
(4) Descend to sea level, no distance credit.
(5) 50 nautical mile military power run-in (M = .95). Drop weapon.
(6) 50 nautical mile military power run-out (M = .95).
(7) Military power climb to optimum cruise altitude.
(8) Optimum cruise home (M = .925).
(9) Descend to sea level, no distance credit.
(10) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

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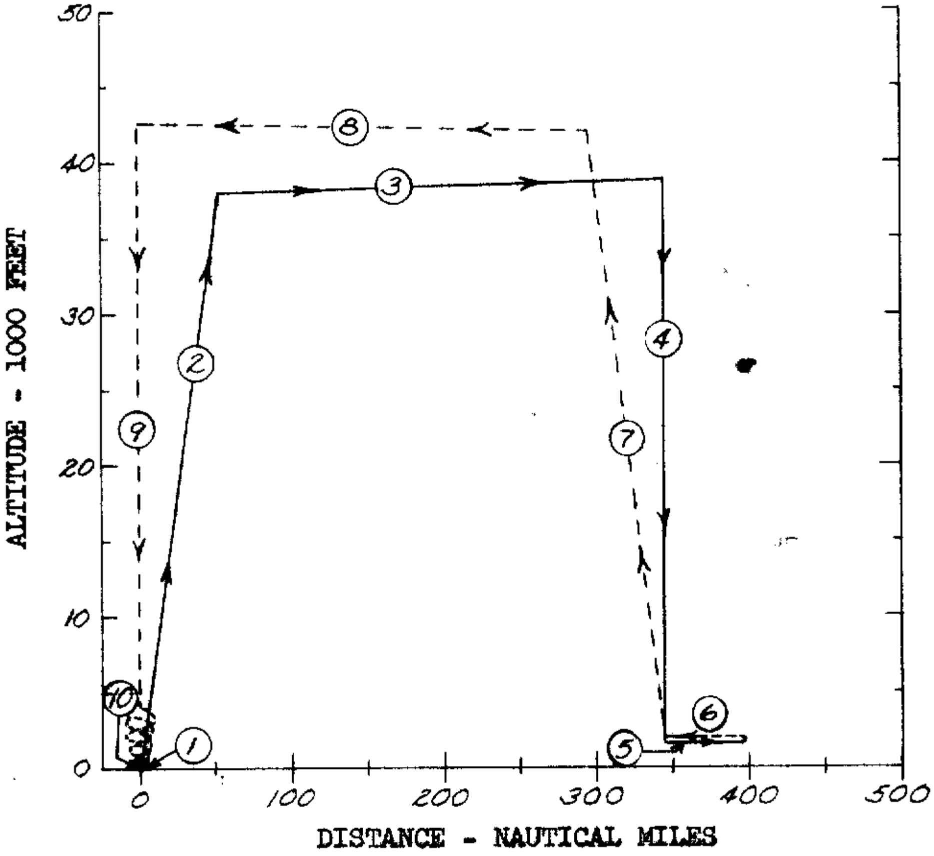
F-106 FIGHTER-BOMBER MISSION
LOW ALTITUDE RUN-IN
INTERNAL FUEL LOAD
(9,945 LBS.)



(1) Warm-up, take-off and accelerate to climb speed at sea level. (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power).
(2) Military power to climb to optimum cruise altitude (average velocity = 500 knots)
(3) Optimum cruise (M = .925).
(4) Descend to sea level, no distance credit.
(5) 50 nautical mile military power run-in (M = .95). Drop weapon.
(6) 50 nautical mile military power run-out (M = .95).
(7) Military power climb to optimum cruise altitude.
(8) Optimum cruise home (M = .925).
(9) Descend to sea level, no distance credit.
(10) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).

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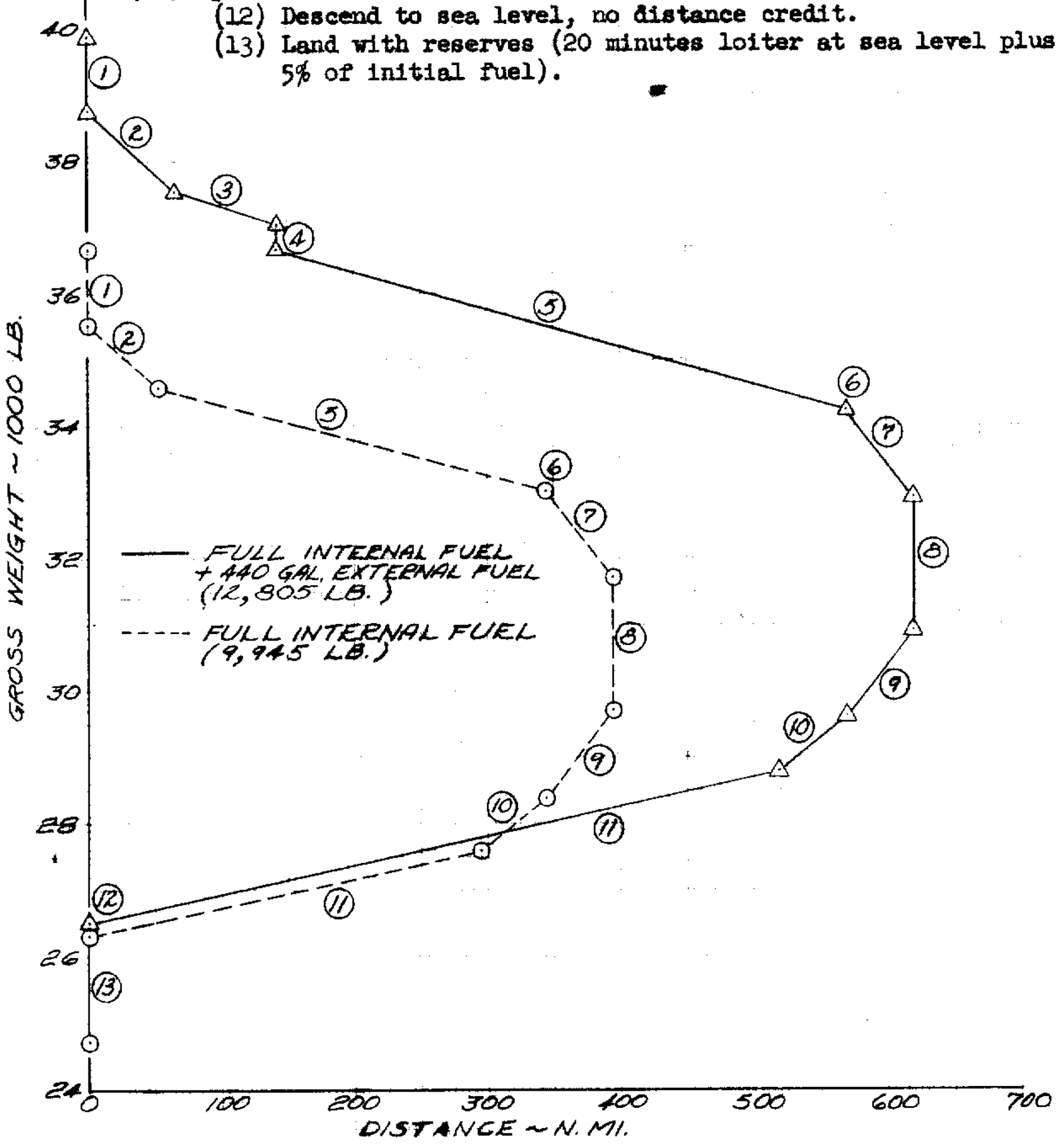
F-106 FIGHTER-BOMBER MISSION
LOW ALTITUDE RUN-IN
WEIGHT VS. DISTANCE

(1) Warm-up, take-off and accelerate to climb speed at sea level. (Fuel allowance: 2 minutes normal rated power plus 1 minute maximum power).
(2) Military power climb to optimum cruise altitude (average velocity = 500 knots).
(3) Optimum cruise with external fuel tanks (M = .925).
(4) Drop external fuel tanks.
(5) Optimum cruise, clean (M = .925).
(6) Descend to sea level, no distance credit.
(7) 50 nautical mile military power run-in (M = .95).
(8) Drop weapon.
(9) 50 nautical mile military power run-out (M = .95).
(10) Military power climb to optimum cruise altitude.
(11) Optimum cruise home (M = .925)
(12) Descend to sea level, no distance credit.
(13) Land with reserves (20 minutes loiter at sea level plus 5% of initial fuel).



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F-106 INTERCEPTOR FIGHTER-BOMBER

WEIGHT SUMMARY

Item	Lbs.
Structure	9,028
Landing Gear	1,224
Propulsion	8,148
Instruments	187
Hydraulic and Pneumatic	389
Electrical	606
Electronics	2,776
Armament Provisions	479
Furnishings and Equipment	525
Air Conditioning and Anti-Icing	397
WEIGHT EMPTY	23,759
Crew	261
Fuel	10,158
Missiles	418
Bombs	2,000
Other	60
GROSS WEIGHT	36,656
External Tanks and Fuel	3,229
GROSS WEIGHT - LONG RANGE	39,885

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Based on Status Report dated 8 Aug. 1958.

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ALL WEATHER CAPABILITY - GROWTH POTENTIAL

Modification of the F102/F106 aircraft need not stop with the quick provision of a simple air-to-ground capability as heretofore described. Through subsequent modifications an all weather bombing capability can be evolved. There are at least two methods by which this can be accomplished.

The Air Force already has the E-30 bombing computer developed by the General Electric Company. Likewise the Air Force has the Sartack, Ku band tactical radar equipment. Both of these pieces of equipment are performing in an outstanding manner. General Electric and Convair have proposed an integrated all weather system utilizing the principles and know-how in these two pieces of developed equipment. On several occasions in the past this proposal when presented was received favorably in that it was stated that such an integrated system was badly needed by the Air Force for tactical use.

The need for a All-Weather bombing system has not diminished and it is believed that the shortest route to it is through the combination of the above subsystems as previously proposed by Convair and General Electric. The E-30 computer and the Sartack radar are already designed to perform essential functions of an All-Weather bombing system. The development required to "marry" these subsystems and incorporate the other essential functions should be a minimum. It is not intended to imply that these subsystems can be used "as is". The only way to evolve a satisfactory bombing system in its entirety is to correctly visualize and plan it, including assigning functions to subsystems of the proper kind, scope and accuracy. This is what General Electric and Convair propose to do to evolve an All-Weather Bombing System. It is proposed that the system be evolved as an integral part of the F102 or F106 aircraft. Convair report ZP8-051A shows such a system integrated into the Convair F102 aircraft.

Building up a system composed of adapted existing components not designed as subsystems to a particular major system can never be as satisfactory.

The second method by which an All-Weather system can be evolved is by taking an All-Weather system designed for a different but similar purpose and modifying the components so as to accomplish the desired end. The Hughes MA-1 fire control system has a very flexible digital computer and most of the functions required for air-to-ground attacks. Convair report ZP28-001 describes a modified MA-1 system integrated into the F106 airplane to provide a Fighter Bomber fire control system.

In either of the above developments an air-to-air capability is retained but it becomes secondary to the air-to-ground capability. As a consequence and as illustrated in report ZP8-051A consideration is given to reducing air-to-air armament in favor of more fuel and air-to-ground armament. Similar measures can be taken with respect to the F106 airplane. In either case the combat radius would be increased.

The inherent physical features of F102/F106 aircraft permit them to be repeatedly modified to accommodate constantly improving armament and equipment and thus remain superior weapons.

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EXTERNAL STORES

In designing F102/F106 aircraft the philosophy has been to produce an airplane with sufficient internal space to accommodate all the necessary equipment and armament. The object, of course, is to have an aircraft whose performance is not degraded when the armament, which on competitor's aircraft is external, is carried.

However, even after having followed this design philosophy, it is still possible to make provisions for carrying a number of types of armament externally slung beneath the wing and comparatively speaking, the performance is still excellent.

The carriage of external armament is illustrated in the associated reports ZP8-051A and ZP28-001.

TRAINING

Convair is prepared to conduct training programs to familiarize ground crews and pilots with its aircraft and their modes of operation to develop their skills to the levels demanded to successfully operate the airplane and its allied equipment.

SERVICE ORGANIZATION

To back up its training program and provide for unforeseen problems arising in the field, Convair maintains a world wide service organization. Convair is capable of providing service anywhere in the world.

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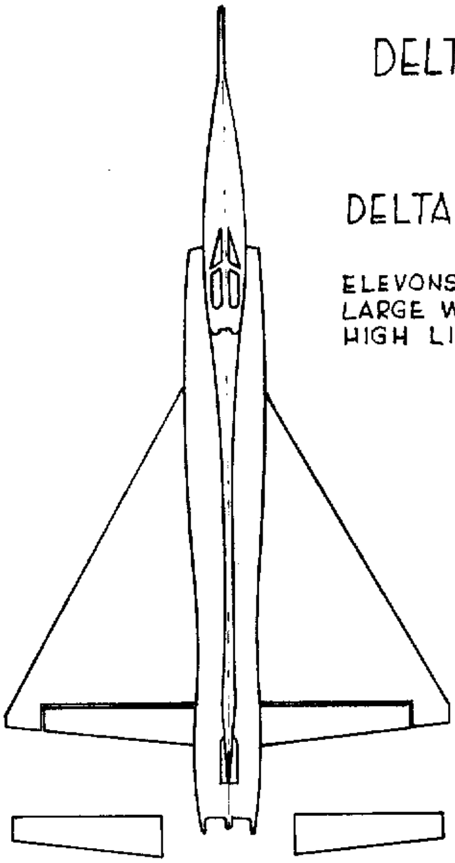
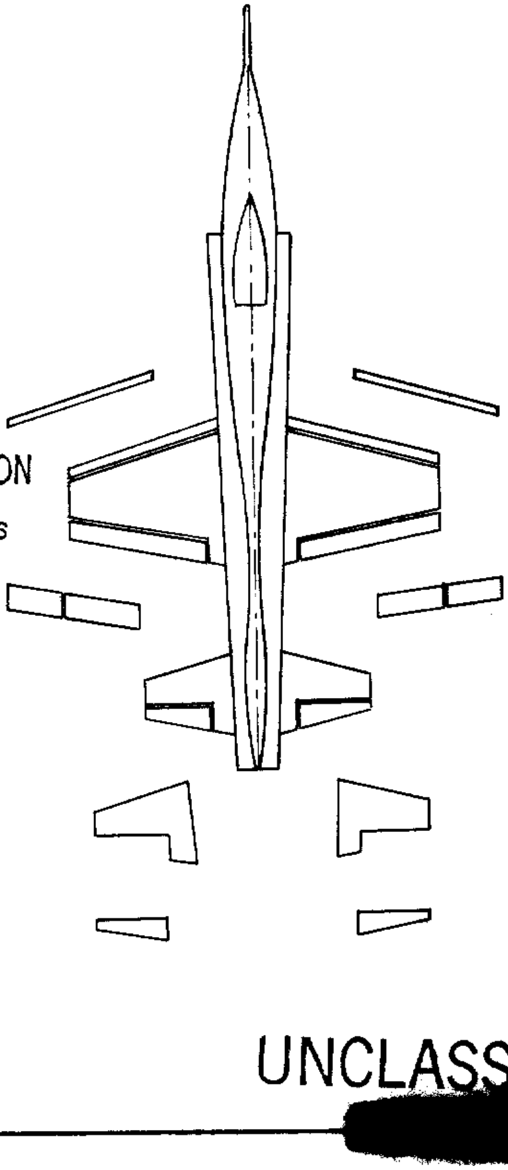
DELTA SIMPLICITY



DELTA CONFIGURATION

ELEVONS FOR CONTROL
LARGE WING AREA FOR
HIGH LIFT.



CONVENTIONAL AIRCRAFT CONFIGURATION

STABILIZER, ELEVATORS, AND AILERONS FOR CONTROL, SLOTS AND FLAPS FOR HIGH LIFT PRESENT OPERATION AND MAINTENANCE PROBLEMS.

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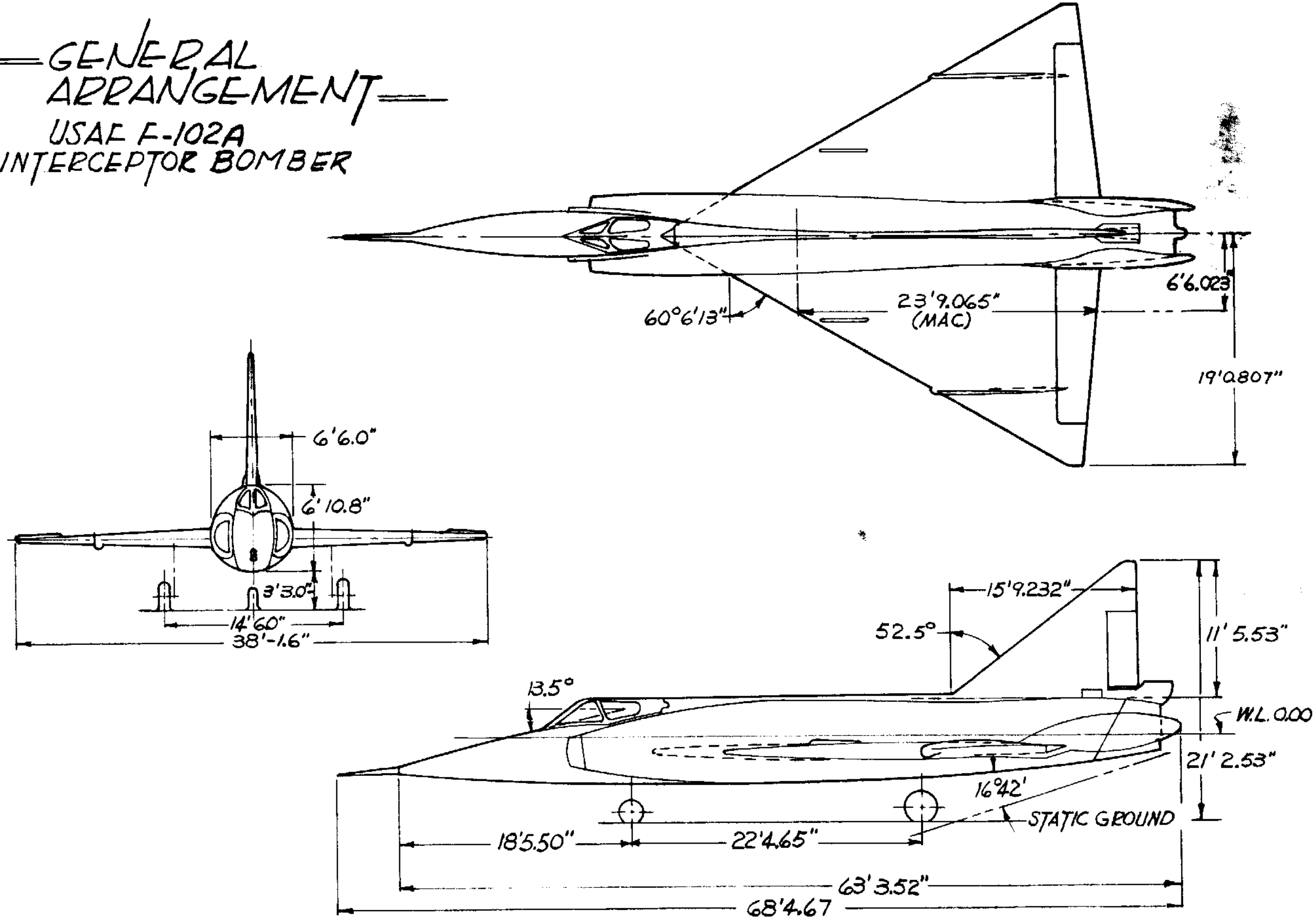
GENERAL ARRANGEMENT

USAF F-102A
INTERCEPTOR BOMBER



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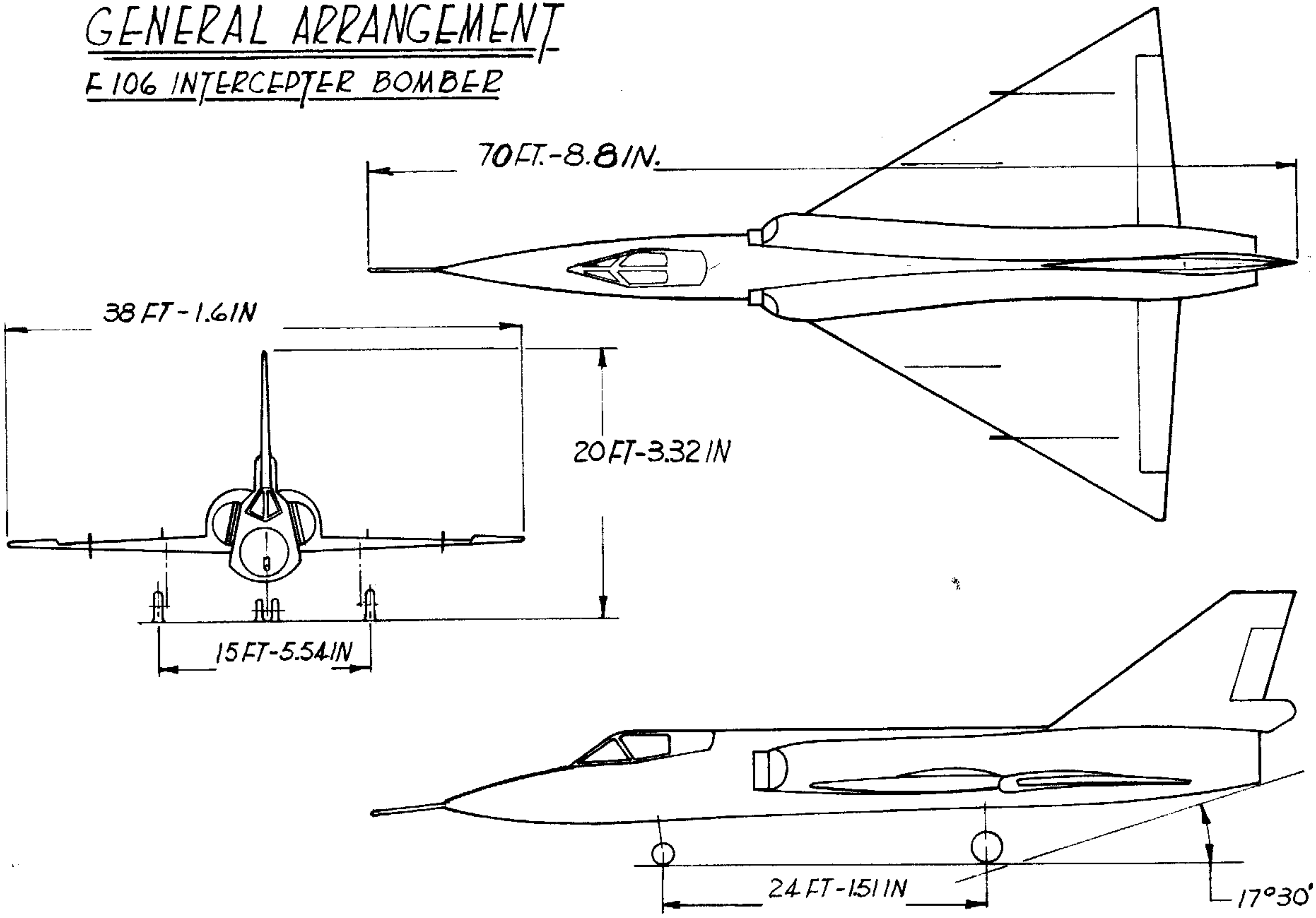
GENERAL ARRANGEMENT

F 106 INTERCEPTER BOMBER



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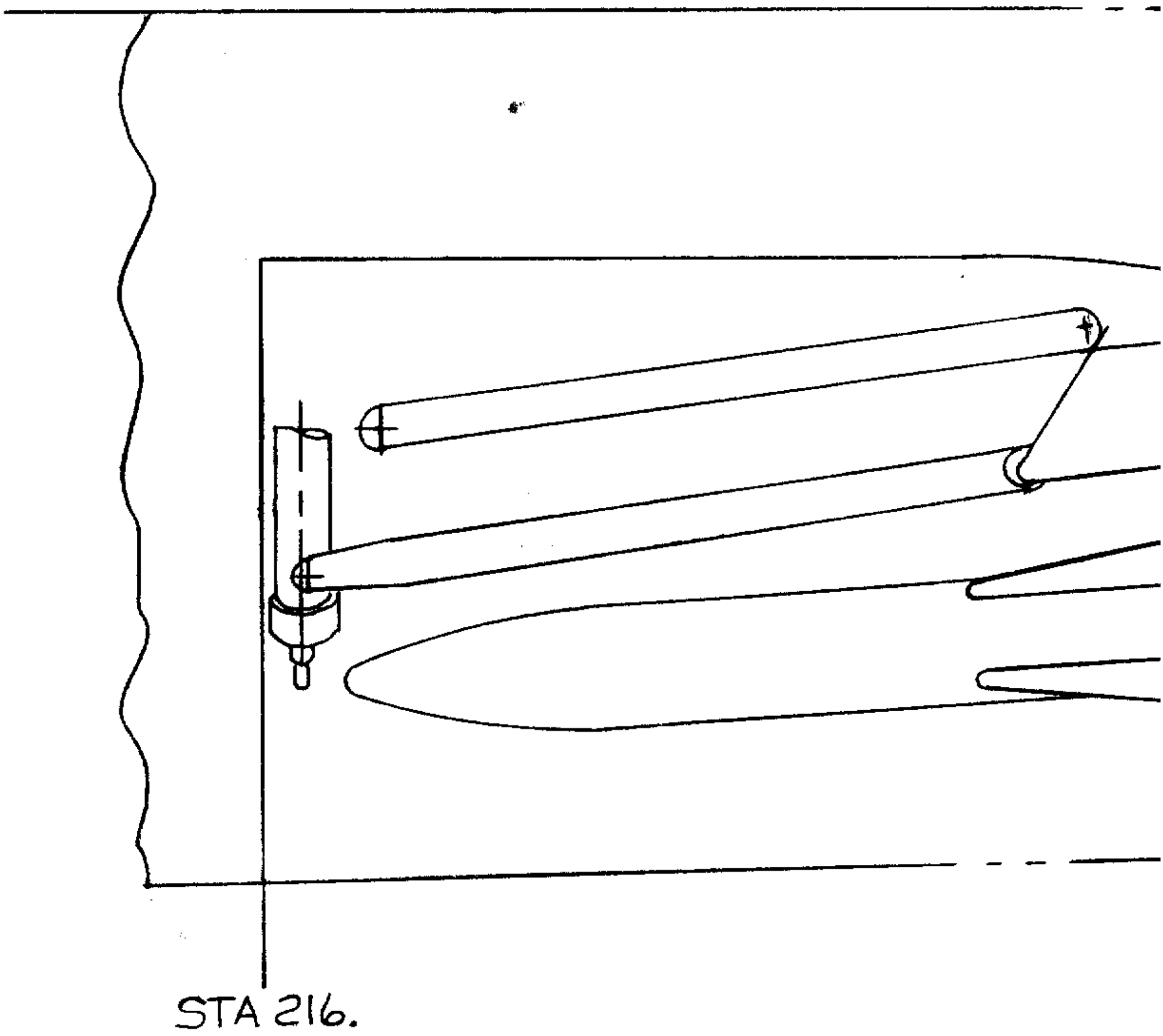
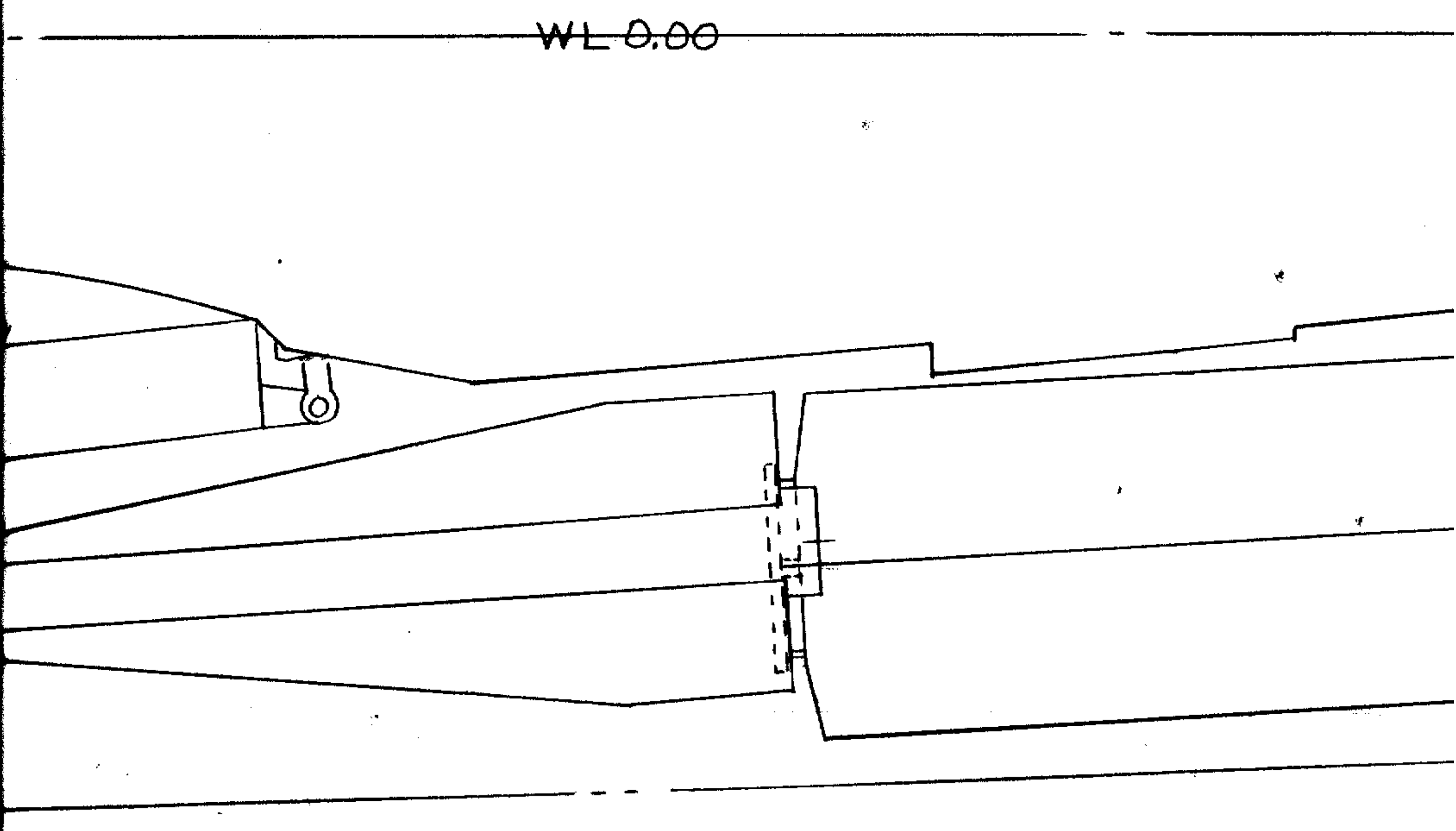
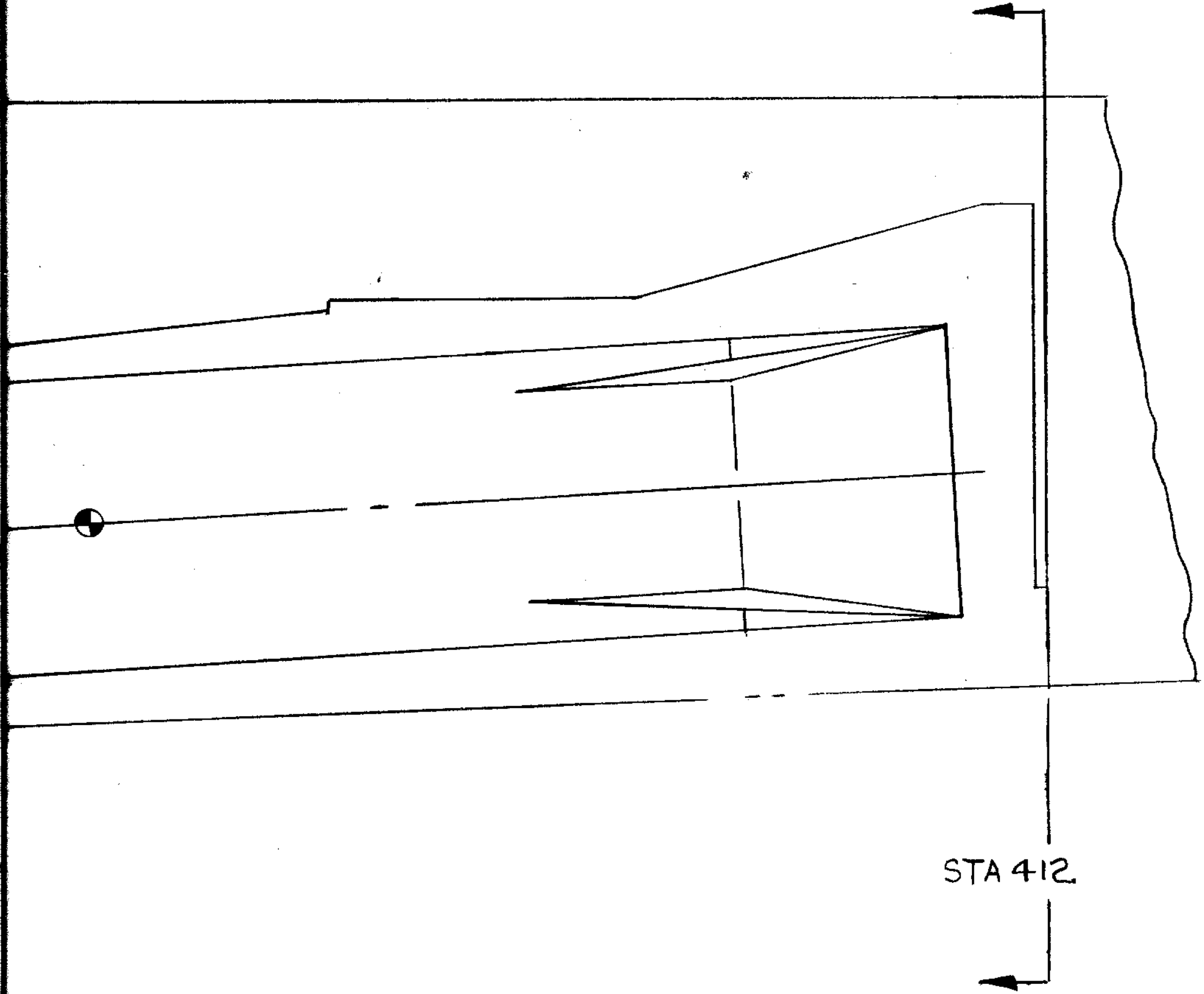
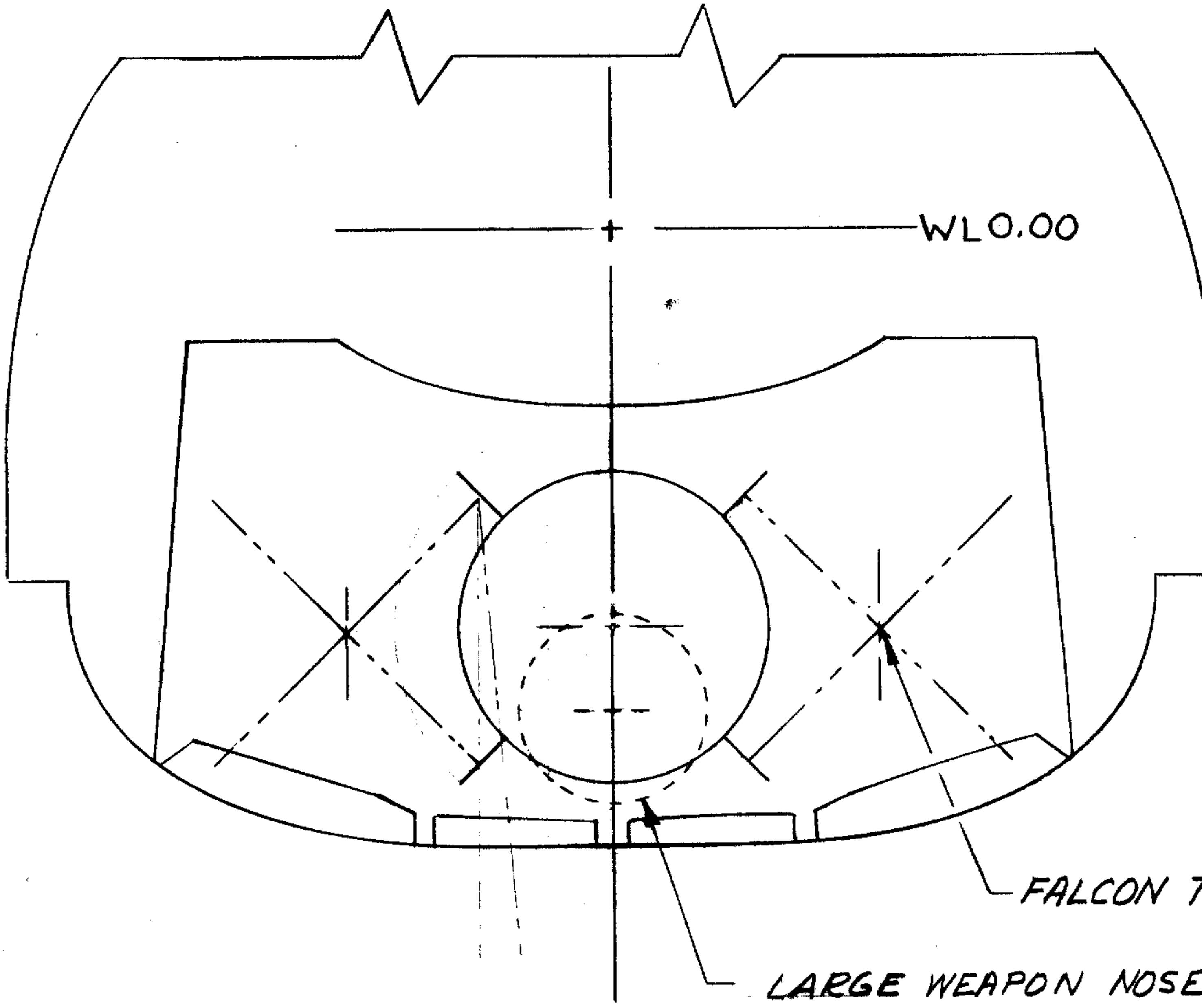


R.E. Dub 7/1/8
BY J.C. WALKER



(2) FALCONS PLUS LARGE WEAPON


STA 412



STA 412 LKG FWD

CONVAI

ALTERNATE ARM

SHEET	SD 58-0
3	

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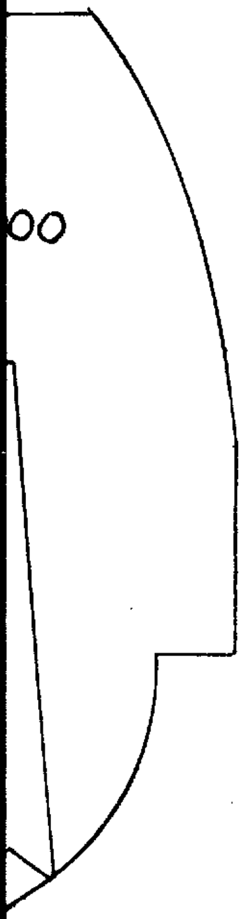

FALCON TAIL & FIN

ON NOSE

NVAIR

NATE ARMT INTERNAL

D 58-08008

ZP-8-027

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A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

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REPORT ZP-8-027
DATE 16 Sept. 1955
MODEL 8
Revised 19 Sept. 1955

TITLE

ADVANCED F-102A AIRPLANE
J57 - P35 ENGINE
CASE XIV CAMBER

ZP-8-027

16 Sept. 1955

PREPARED BY Dick Lingley
R. C. Lingley

GROUP Preliminary Design and Engineering Groups

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert

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INTRODUCTION

This study has been made to determine the task involved on the F-102A (8-10) airplane when installing the J57 - P35 engine and changing the wing from the Case X camber to the Case XIV camber.

It is assumed that current problems on the airplane, such as the speed brakes and duct buzz will be resolved by the time this proposed version is in production; hence, they are not discussed in this report.

For the purpose of this study, the large fin now being designed for the F-102A is neglected since it will be either incorporated or deleted prior to this version. Data showing C. G. effect, however, is included.

SUMMARY

Installation of the Pratt & Whitney J57-P35 engine and rework of the wing to install the Case XIV camber instead of the present Case X camber, increases the performance of the airplane.

High speed is increased from M = 1.18 to M = 1.35. Subsonic ceiling is increased from 53,200 feet to 54,300 feet. The combat radius of 375 nautical miles will be maintained pending further aerodynamic investigation.

The physical changes are confined to the engine change, revised wing camber and extended tail cone. The extended tail cone decreases the tail touchdown angle from 18° to 17°45'. This is considered negligible.

The "q" limitations and load factors of the 8-10 airplane will be maintained.

For weight comparison, the existing F-102A take-off gross weight has been corrected to include the category I horizon items and 36 2.00" rockets.

The engine and camber change decreases this gross weight 246 lbs. with no ballast required.

If the large fin is included with the engine and camber change, the gross weight is decreased 31 lbs. including 43 lbs. ballast. This ballast could be eliminated by an impractical wing shift of 0.6"; or, more reasonably, by a forward nose section movement of 4.5 inches.

[redacted]

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[redacted]

Note: This report is based upon using titanium in the J57-P35 engine. Weights shown were obtained verbally from the Pratt & Whitney representative. In the event the engine is changed to steel with a subsequent weight increase of 388 lbs., it would be necessary to shift the wing 2.5 inches to maintain the C.G. within limits (no ballast) or add 185 lbs. ballast in the nose.

[redacted]

DISCUSSION

I. Propulsion Group

The J-57 - P-35 engine is 388 pounds lighter and 5 inches longer than the J-57 - P-23 and has an aft C.G. shift of approximately 7 inches, thus, resulting in a slightly forward shift of the airplane C.G. The following changes are necessary due to differences in engine configuration:

1. Main Fuel Connection

This connection has moved outboard approximately 4 inches, increased in I.D. from 1.781 to 2.281 and changed from a 4 stud to a 6 stud connection.

A larger connecting line must be provided to match larger flange and I.D. This new line must be so routed as to clear the A.C. generator. (Clearance problem caused by connection moving outboard.) This rerouting may affect the fuel lines back to the flow equalizer.

2. Engine Shroud

A new shroud will be necessary because:

A. The contour of the fireseal has changed.
B. The routing of the A/B fuel and actuator air lines interfere with the shroud baffle.
C. The engine has increased in length approximately 5.12 inches in the region of the A/B.

3. Ignition Unit Cooling

Excitors are to be mounted opposite the oil tank. Temperature limits are to be similar to the J-75. If cooling is not sufficient in this area, additional cooling will have to be added.

4. C. S. D. Oil Lines

It will be necessary to reroute the C.S.D. oil system to agree with relocated oil system components. Fuel pressure switch and C.S.D. disconnect line mounting pads are gone.

[redacted]

5. Power Control - Power lever movement at power control differs from previous engines. There are no take-off locks. Until we are told differently, Pratt & Whitney recommends that we assume travels the same as on the P-23.

6. Inlet Duct Strut - The strut in the inlet duct is no longer needed as a passage is provided through engine for drains from N1 gear box. The present strut type duct may be used without alterations if needed by instrumentation.

7. Fuel System - A slight decrease in fuel tank capacities due to the rearrangement of the wing structure can easily be made up by adjustment of the automatic shutoff.

8. Changes Required from other Groups

   A. Control Group - If power lever travel changes, they will possibly have to rework the quadrant.

   B. Body Group - The increased engine length will necessitate changes to the tailcone.

II. Wing Group

The wing change is confined mainly to the leading edge area. The leading edge spar is relocated to accommodate the new cambered leading edge. Increase in wing bending causes a change to spars #4 and #5. The following wing changes must be made:

1. New leading edges - camber similar to F-102B
2. New wing tip - similar to F-102B.
3. New leading edge spar.
4. Four new gas-tight corners per side.
5. Modify 5 wing skins per side - change at leading edge attachment.
6. Change 25 ribs per side.
7. Change forward fuselage tie (Sta. 298.05) to accommodate revised leading edge spar and gas-tight corner.

[redacted]

8. Change outboard region of spar #2.

9. Change outboard end of spar #3 to accommodate changed leading edge spar.

10. Change outboard end of spar #4 to accommodate changed leading edge spar. Also increase inboard fuselage attach lug width and bolt size.

11. Change spar #5 at outboard end and increase flange thickness throughout. Also increase inboard fuselage attach lug width and bolt size.

12. Change spar #6 at outboard end and increase flange thickness throughout. Revise elevon hinge fittings.

13. Change inboard and outboard fence to agree with modified camber.

14. Revise inboard fuel tight rib at leading edge.

15. Change drag angle installation at leading edge.

16. Change 10 wing access doors per side. Change door installations also.

17. Change outboard trailing edge access door and installation.

18. Change top assembly drawing to reflect foregoing changes and initiate common releases.

III. Fuselage Group

The fuselage changes are as follows:

1. Increase tail cone length 5-1/8 inches due to longer engine.

2. Relocate C.S.D. oil filter access door to agree with engine change.

3. Redesign fuselage attachment to wing at spars #4 and #5 due to increased bending of spars due to camber.

[redacted]

IV. Hydraulics and High Pressure Pneumatics

Revision of tubing to accommodate revisions in power plant installation.

V. Structures Group

1. Wing root bending moment will increase from 5% to 10% due to wing shape changes and minor weight changes.

2. Wing internal load distributions will change due to new airload distributions.

3. "q" limitations and load factors of the 8-10 airplane will be maintained.

4. There will be minor shear and moment changes on fuselage due to new airload distributions.

5. It is assumed that the present specifications and methods of compliance will be retained.

[redacted]

VI. WEIGHT AND BALANCE

	WEIGHT	HORIZONTAL ARM	MOMENT
Basis: T. O. Gross Weight*	27,566	420.0	11,577,484
Plus Changes:			
P-35 engine in lieu of P-23	(-318)	461	(-146,647)
Delete P-23 engine	-5,058	541	-2,736,378
Delete bleed governor drain strut	-2	450	-900
Extend fuselage 5"	+8	702	+5,616
Extend shroud 5"	+5	700	+3,500
**Add P-35 engine	+4,720	546	+2,577,120
Flex hose fuel line vs. tube	+2	520	+1,040
C.S.D. oil line rerouted	+3	505	+1,515
Cooling ducts for G.L.A. boxes	+4	460	+1,840
Case XIV Wing in Lieu of Case X	(+72)		+39,758
Wing increase	+52	579	30,114
Fuselage increase	+20	482	9,644
Sub-Total - Condition 1	27,320	420.0	11,470,595
Plus Additional Changes:			
95.1 Sq. Ft. tail in lieu of 68.3 Sq. Ft.	+172	591	+101,684
Add ballast = 43 lbs.	+43	217	+9,331
Sub-Total - Condition 2	27,535	420.6	11,581.610

Note: Without ballast, wing would move aft 0.6 inches or move nose section forward 4.5 inches from Sta. 122.

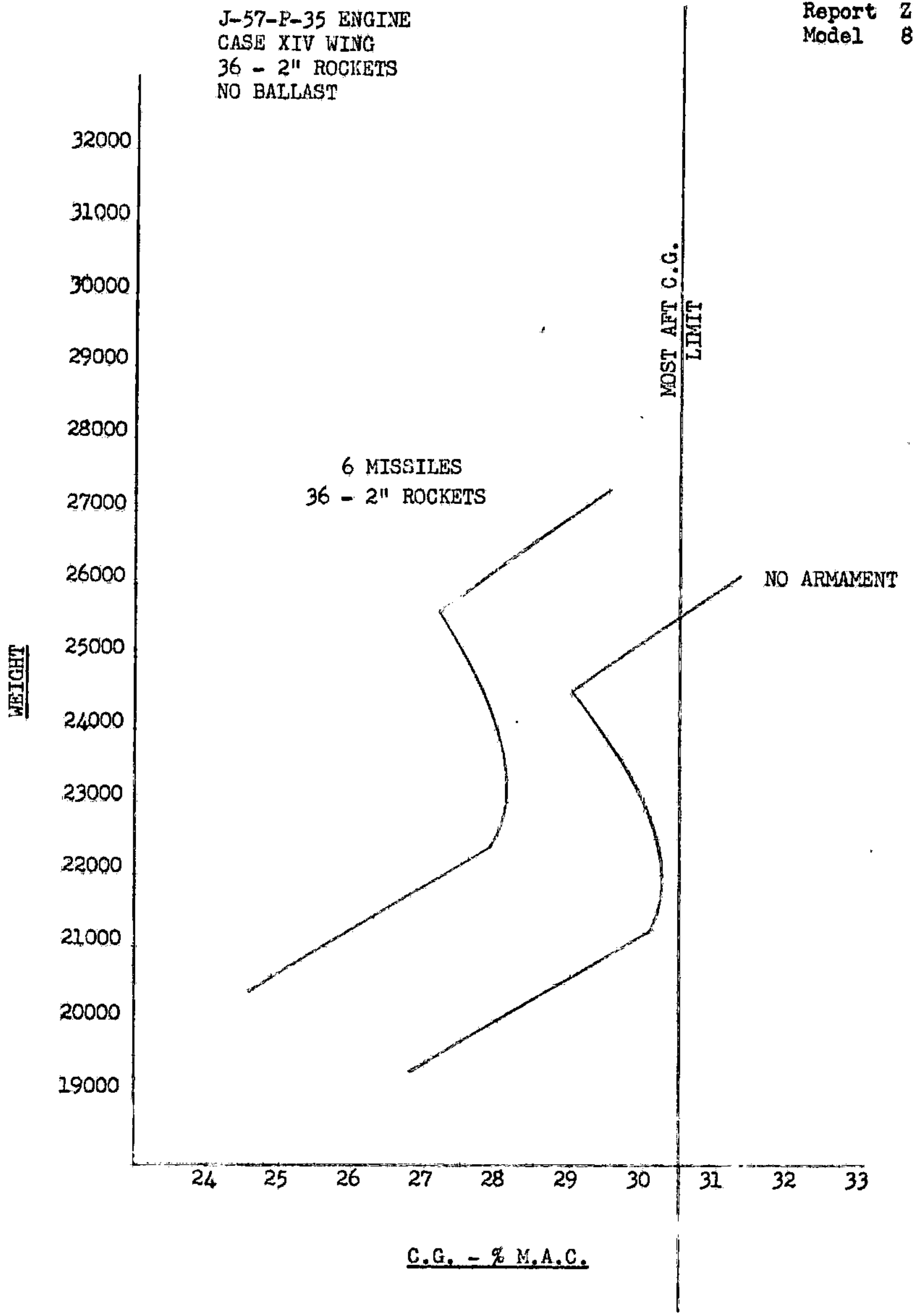
Notes: 1. * Take-off gross weight of existing F-102A (Ship #5) plus 36 2.00" rockets and category I horizon items.

2. Conditions based upon maintaining the most aft C.G. limit of 30.5%

**3. P-35 engine weight based on verbal information from P&W rep. and based on a titanium engine. A steel engine would be approx. 388# more and would require 185# of add'l. ballast. (388+185=+573 # gross weight) without this additional [redacted] ballast, the wing would have to move aft an additional 2.5".

[redacted] 102A - IMPROVED VERSION

CONDITION 1


J-57-P-35 ENGINE
CASE XIV WING
36 - 2" ROCKETS
NO BALLAST

[redacted] R. L. Benson

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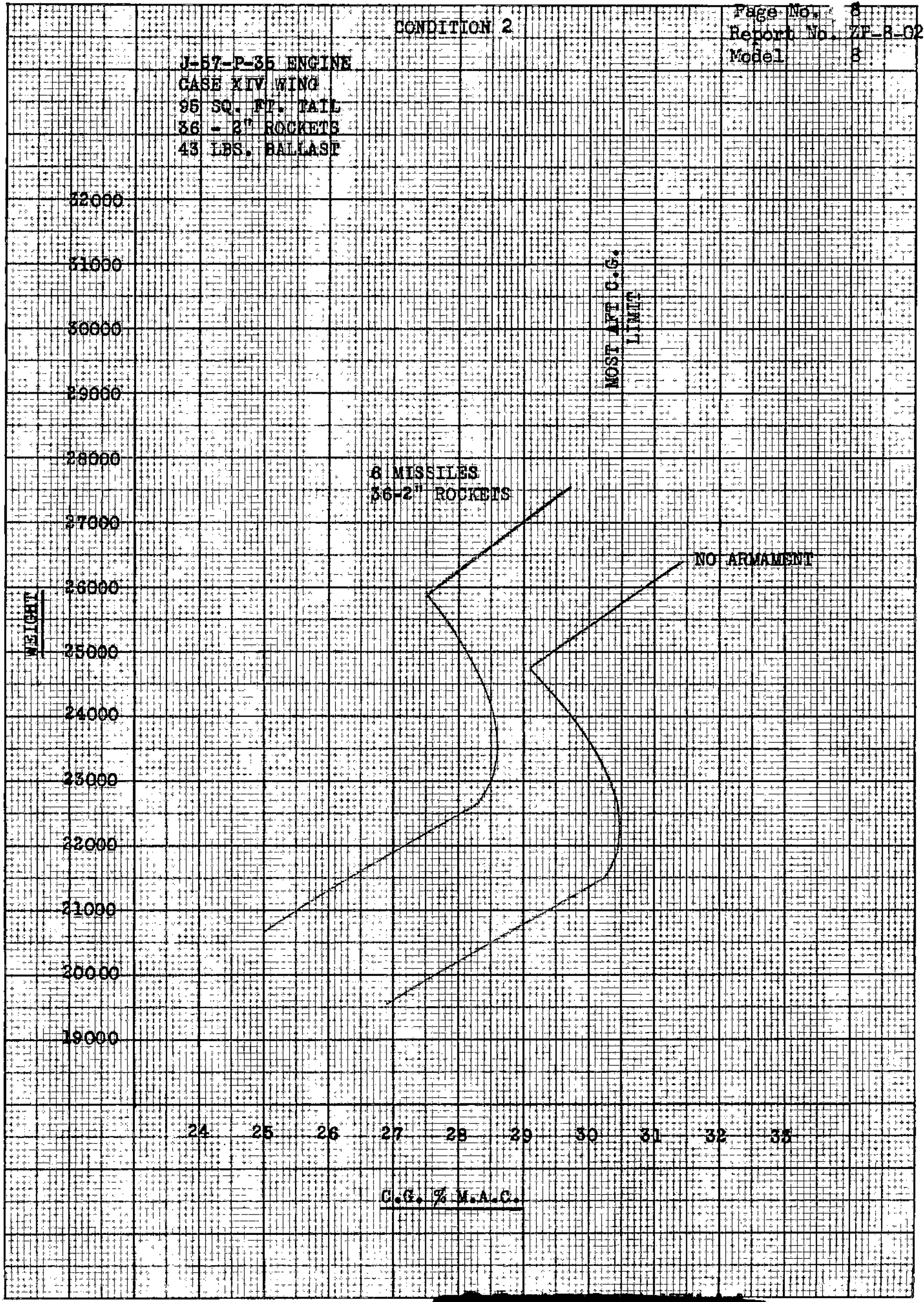
F-102A - IMPROVED VERSION

9-15-55

Page No. 8
Report No. ZP-8-027
Model 8

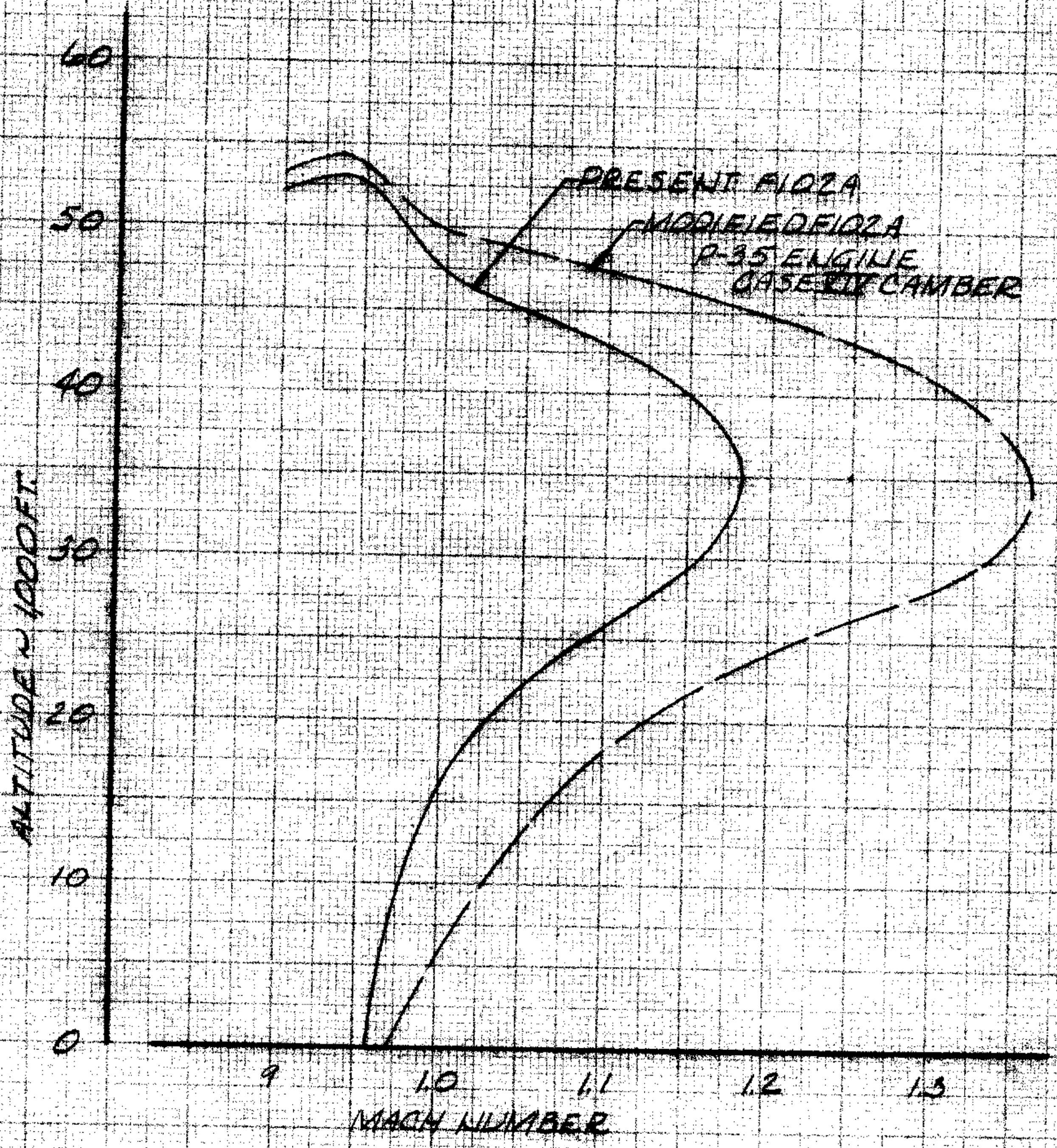


10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO.

[redacted] L. Benson

PERFORMANCE COMPARISON

F102A VERSUS MODIFIED F102A AIRPLANE



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APPENDIX "A"

INTRODUCTION:

This study explores the question of what to do if the advanced airplane has been scheduled into production and the J57-P35 engine is unexpectedly delayed.

SUMMARY:

The existing J57-P23 engine could be installed in the advanced airplane with extremely minor changes.

A performance gain is still realized over the present F-102A airplane due to the Case XIV camber change.

No ballast would be required due to the engine and camber changes; however, if the large fin has been incorporated, 44 pounds of additional ballast will be required at Station 217.

When the advanced engine becomes available, it could be installed with equally minor changes.

DISCUSSION:

These changes would be necessary to install the existing J57-P23 engine in the advanced version of the airplane.

1. Use the existing J57-P23 fuel line in place of the approximately 3-ft. of line that will have been revised for the advanced engine.
2. If ignition cooling is necessary for the advanced engine, the source of the cooling air must be capped.
3. If differences in the power lever movement are incorporated in the advanced engine, they must be accounted for at the power level quadrant.
4. Use existing tail cone.
5. Differences in engine shroud, C.S.D. oil lines and inlet duct strut will be accomplished by change of engine.

[redacted]

Appendix "A"

ANALYSIS
PREPARED BY Dick [illegible]
CHECKED BY
REVISED BY

CONVAIR
SAN [illegible]

PAGE 2A
REPORT NO. ZP-8-027
MODEL 8
DATE 23 Sept. 1955

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WEIGHT & BALANCE COMPARISON:

	Advanced F-102A		
	J57 - P35	J57 - P23	Difference
*Cond. 1. T.O. Gross Wt. No ballast required	27,320	27,638	+318
*Cond. 2. T.O. Gross Wt.	27,535	27,897	+362
87 Lb. ballast required at Station 217 (was 43 lb.) or move the nose forward 9 inches.			

*See Page 6 of Report ZP-8-027

PERFORMANCE COMPARISON:

	J57 - P35	J57 - P23	Difference
High Speed	M = 1.35	M = 1.24	-.11 Mach
Subsonic Ceiling	54,300 Ft.	53,500 Ft.	-800 Ft.

375 nautical mile combat radius will be maintained.

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Negative in Misc

tubes –

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-023
DATE 14 July 1955
MODEL 8

TITLE

PRELIMINARY EVALUATION OF THE EFFECTS OF EXTERNALLY MOUNTED DING DONGS ON F-102A AIRPLANE

ZP-8-023

14 July 1955

PREPARED BY H. E. Dickard
H. E. Dickard

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert

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Weapon System Effectiveness	3
Performance	4 through 5
Weights Data	6 through 8
Chart - Major items investigated for external Ding Dong installation	9
Discussion:	10 through 14
Airplane Performance Handling Characteristics Structural Strength Flutter Missile Installation Electronics Fuselage Ballast Problem Areas	

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SUMMARY

This report presents a preliminary evaluation of the effects of installing two (2) Ding Dong air-to-air missiles in place of external fuel tanks on the wing of the F-102A airplane. The Ding Dongs are alternate to the basic internal Falcon and rocket armament.

The study is based on two weight configurations. The first is an absolute minimum change in which only armament is removed. The second involves removing basic internal armament, displacing gear and easily detachable related items. Both arrangements result in take-off weights greater than the basic F-102A and both require removable fuselage ballast to maintain C. G. limits.

The external Ding-Dong armament as studied in this report appears to be feasible provided that the following basic problems are resolved:

1. Effects of under-wing mounting position on missile dispersion as related to firing accuracy.
2. Effects of removable ballast on the fuselage structure and of the pylon mounted Ding D ongs on the wing local structure.
3. Effects of missile rocket jet pressures and temperatures on the wing structure and on handling characteristics of the airplane.
4. Extensive investigation of the dynamic characteristics of the airplane as affected by the mass distributions associated with the external Ding Dong arrangement.

It would appear that the design changes resulting from the above problems could be incorporated in production F-102A airplanes without need for a prototype airplane, provided sufficient data is obtained from presently scheduled tests to determine the effects of Item 3 above.

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WEAPON SYSTEM EFFECTIVENESS

Insofar as the bomber intercept mission is concerned, it would appear that the performance degradation, including a slight reduction of the interception ceiling, caused by external Ding Dongs is of minor consequence. Neglecting possible gross increases in missile dispersion due to launching from the under-wing position, the effectiveness of the external Ding Dongs would be comparable to that of the internally stored system. "Gross increase" is taken here to mean more than twice the launch errors now assumed attainable.

The predicted performance differences on which the above statement is based are as follows:

	(1) F-102A	(2) F-102A & Ding Dong
High Speed, Mach No.	1.18	1.10
Radius, N. Mi.	375	310
Time to Climb, Min.		
40,000 Feet	4.3	4.9
50,000 Feet	7.1	8.9
Combat Ceiling, Feet	52,500	50,800

(1) Based on a take-off weight of 27,500 Lbs.

(2) Based on a take-off weight of 28,940 Lbs.

[redacted]

F-102 EFFECT OF EXTERNAL DING DONG INSTALLATION

ESTIMATED TABULATED PERFORMANCE

[redacted]

			F-102A + Ding Dongs	
		F-102 A	(1)Cond. I.	(1)Cond. II
Rated Static Thrust at Sea Level (J-57 P-23)	Lb.	16,000	16,000	16,000
Take-Off Weight (Full Load)	Lb.	27,500 (2)	28,940	28,605
Combat Weight*	Lb.	24,867	26,080	25,765
Landing Weight **	Lb.	21,400	22,950	22,580
Fuel	Lb.	6,825	6,825	6,825
High Speed with Military plus Afterburner Power 35,000 feet	Mach No.	1.18	1.10	1.10
Combat Ceiling with Military plus Afterburner Power (500 Ft./Min. Rate of Climb)	Ft.	52,500	50,800	51,100
Time to Climb to Altitude with Military plus Afterburner Power (from start of take-off run)				
40,000 Ft.	Min.	4.3	4.9	4.8
50,000 Ft.	Min.	7.1	8.9	8.5
Take-Off Without Auxiliary Devices over 50' Obstacle with Military plus Afterburner Power (Take-Off Weight)	Ft.	2,900	3,200	3,100
Land Over 50' Obstacles (Landing Weight)				
Without Auxiliary Landing Devices	Ft.	4,300	4,500	4,400
With 14' Diameter Drogue Chute	Ft.	3,600	3,800	3,700
Combat Radius - Area Intercept Mission per MIL-C-5011A (Combat 5 Min. at Max. Power)	N. Mi.	375	310	320

* Weight at start of combat for Point Intercept Mission.

** Landing Weight corresponding to Point Intercept Mission. Includes full armament and 20 Min. reserve fuel at sea level (at V for maximum endurance).

(1) Ref. Weight Study on following pages.

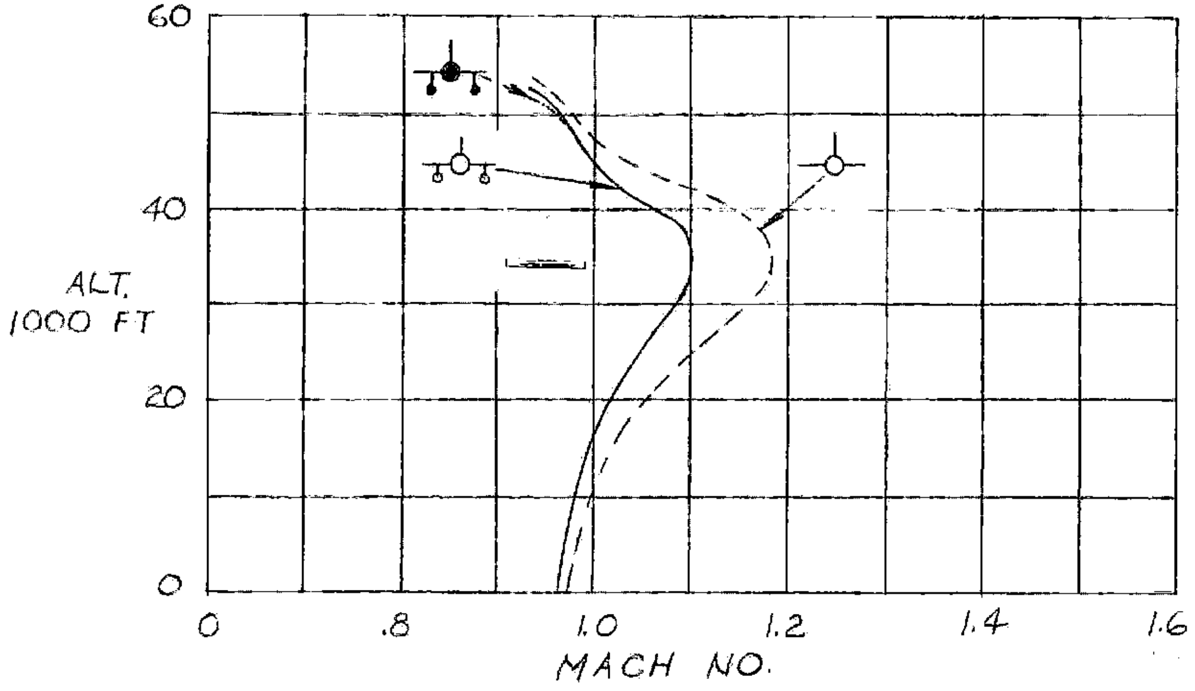
(2) IBM Wt. Status Report #23, 1 May 1955, shows 27,605 Lb. or .38% increase. This discrepancy is considered negligible insofar as this report is concerned.

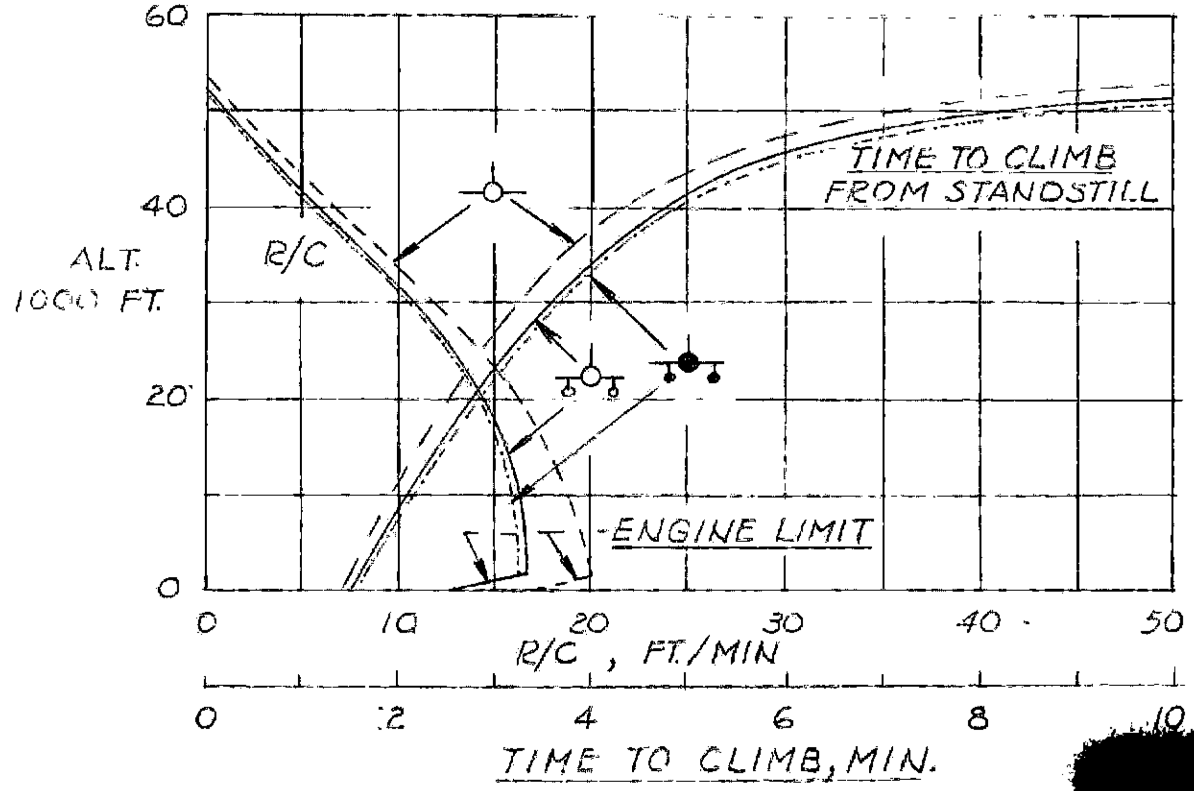
[redacted]

PREDICTED PERFORMANCE

F-102 A WITH INTERNAL ARMAMENT, 27500 LB, T.O WT.
F-102 A WITH (2) EXT. DING DONGS , 28605 LB, T.O. WT.
F-102 A WITH (2) EXT. DING DONGS , 28940 LB, T.O WT.

MIL + A/B POWER J-57P-23 ENGINE





[redacted]

WEIGHT BASIS

The two weight conditions considered for the external Ding Dong installation are shown in the following tabulation. In both cases, the (24) 2.75" rockets have been left in as part of the ballast required. Dummy rockets, or cylindrical ballast weights of equivalent size and weight may be used in place of the rockets.

Condition I

Includes complete Falcon provisions.

				Lb.
F-102A	Gross Weight per IBM Status Report #23, 5/1/55			27,605
Less:				
	Missiles (6) Falcons	-762		
	Launchers	-223	-	985
Plus:				
	Provisions for Ding Dongs			
	Rails	60		
	Pylons	84		
	Attaching Parts	6		
	Electrical Provisions	16		
	Frangible Fairing	12		
	Escape Maneuver Equipt.	35		
	Ding Dongs (2)	1630		
	*Ballast at Sta. 200	477	+	2,320
	T. O. Weight			28,940

*Additional ballast at Sta. 265 in the form of rockets or equivalent weight in the missile bay doors is implied by not removing 432 Lbs. for rockets.

Total Ballast = 477 + 432 = 909 Lbs.

[redacted]

WEIGHT BASIS

Condition II.

Removes all Falcon provisions which are easily detachable.

F-102A	Gross Weight per Status Report No. 23		Lb. 27,605
Less:			
	Missiles (6) Falcons	-762	
	Missile Launchers	-223	
	Missile Intervalometer	- 4	
	Displacement Gear, Fwd.	-111	
	Displacement Gear, Aft.	-110	
	Displacing Cylinders	- 49	
	Snubber Cylinders	- 23	
	Uplatch and Release	- 3	
	Uplatch Stops	- 37	
	Air Bottle	- 19	
	MG-3 Equip. for Falcons	-350	
			-1,691
Plus:			
	Provisions for Ding Dong	213	
	Ding Dongs (2)	1,630	
	*Ballast at Sta. 200	848	
			+2,691
	T. O. Weight		28,605

*Additional ballast at Sta. 265 in the form of rockets or equivalent weight in the missile bay doors is implied by not removing 432 lbs. for rockets. Total Ballast = 848 + 432 = 1,280 Lbs.

[redacted]

GROUP WEIGHT STATEMENT

			F-102A + Ding Dongs	
		F-102A (3)	Cond. I	Cond. II
Wing		3,086.6		
Tail		362.0	Figures are shown in these two columns only when they differ from the basic F-102A	
Fuselage		3,526.7		
Landing Gear		1,046.6		
Surface Controls		433.9		
Nacelle		36.5		
Propulsion		6,376.9		
Instruments		144.1		
Hydraulics & Pneumatic		259.0		
Electrical		564.3		
Electronics		2,068.5		1,718.5
Armament		574.4	703.4	347.4
Furnishings		196.8		
Air Conditioning & Anti-Icing		205.3		
Auxiliary Gear		53.4		
Ballast (At Sta. 200)		0.0	(1) 477.0	(2) 848.0
WEIGHT EMPTY		18,935.0	19,541.0	19,206.0
Pilot		230.0		
Unusable Fuel		130.0		
Unusable Oil		25.0		
Oil - Engine		25.0		
Fuel - (Wing)		6,825.0		
Armament		1,417.0	2,146.0	2,146.0
Rockets (24)	432.0		(1)	(2)
Falcons (6)	762.0			
Ding Dongs (2)	0.0		1,630.0	1,630.0
Launchers	223.0			
Pylons	0.0		84.0	84.0
Equipment		18.0		
USEFUL LOAD		8,670.0	9,399.0	9,399.0
T.O. WEIGHT		27,605.0	28,940.0	28,605.0

Note: (1) & (2) Total chargeable to ballast:
(1) 477 + 432 = 909 Lbs.
(2) 848 + 432 = 1,280 Lbs.
(3) IBM Wt. status #23, 5/1/55

[redacted]

CHART - [redacted] INVESTIGATED FOR EXTERNAL DING-DONG INSTALLATION

		ITEM AFFECTED				
	EFFECT	AIRPLANE PERFORMANCE	HANDLING CHARACTERISTICS	STRUCTURAL STRENGTH	FLUTTER	AIR PLANE BALANCE
1	Addition of two (2) Ding Dong Missiles Externally Mounted	Page 4		Page 11 - 12	Page 12	Page 6 & 7
2	Increase in Gross Weight	Page 4		↓		
3	Asymmetric Aerodynamic Configuration (1 Missile Only)		Page 10	↓		
4	Yawing Effect Due to Momentary Restraint of Missile in Launching Rack After Firing		Page 11	↓		
5	Transient Pressure and Temperature Effects Due to Jet Blast of Missile		Page 11	↓		

[redacted]

DISCUSSION

The chart on the preceding page points out the areas of investigation covered in this study. The work done in connection with the various phases is discussed as follows:

1. Airplane Performance

The comparative tabulated data and performance curves of Page 5 describe the airplane performance. The differences in predicted performance between the basic F-102A and the F-102A with externally mounted Ding Dongs, is due to the increase in gross weight and the incremental drag of the missiles. The magnitude of the drag increment due to the missiles is estimated as follows:

Mach No.	C_D
.6	.0010
.8	.0011
1.0	.0021
1.2	.0025

2. Handling Characteristics

The handling characteristics of the airplane are affected by:

2.1 Asymmetric aerodynamic configuration with one (1) missile in place.

2.2 Yawing effect due to momentary restraint of missile in launching rack after firing.

2.3 Transient wake pressure and temperature effects due to the jet blast of missile in proximity to the wing.

2.4 Change in mass distribution with external missiles and fuselage ballast.

[redacted]

The effect of item 2.2 is considered negligible in view of the comparatively low value of restraint (2500 lbs.) and the time interval involved (.08 sec). The effects of item 2.3 are unknown. Trim conditions associated with item 2.1 above are tabulated as follows:

CONTROL SURFACE DISPLACEMENTS FOR
TRIM WITH ONE (1) MISSILE ONLY

	ELEVATOR	RUDDER	AILERON
150 Knots	0	± 1.0 °	± 1.6 °
M = 0.9	0	± 0.3 °	± 0.5 °
M = 1.1	0	± 0.5 °	± 0.4 °

A preliminary check on the possibility of roll-coupling due to the redistribution of mass (item 2.4) reveals no unfavorable effects for the following conditions:

M = .9 at 35,000 feet

Starting from 1 g level flight:

0 to 90° bank
0 to 180° bank
Bank-to-bank maneuvers starting
From steady state turns corresponding to the following load factors:

1.4 g to 1.4 g
2.0 g to 2.0 g
3.0 g to 3.0 g

3. Structural Strength

The strength of the airplane is affected by:

3.1 Increase in gross weight and the redistribution of wing and body loads due to the presence of both or one (1) missile.

3.2 Pitch, yaw, & roll accelerations due to the changes in drag and weight due to separating one (1) missile from the airplane.

[redacted]

3.3 Pitch and yaw accelerations associated with the Ding Dong firing due to the momentary restraint (thrust build-up) of the missile in the launching rack.

3.4 Transient wake pressures and temperatures due to the jet blast of the missile in proximity to the wing.

Considering items 3.1, 3.2 and 3.3 above, the following investigations were made:

(a) Landing Loads' estimates were made of the transient loads for a symmetric two (2) wheel condition and for an unsymmetric one (1) wheel condition. From these loads, it appears that the basic F-102A load factors will not be exceeded.

(b) Flight Loads: Flight load investigations were made for the effects of firing both or one (1) missile. The launching rack restraint load of 3.3 above was also considered.

The assumption was made that the missiles would not be fired at limit vertical load factors greater than the presently used armament limitations for the basic F-102A (+ 3.0 g to - 1.0 g). For symmetric maneuvers with Ding Dongs attached, it appears that total limit load factors of + 5.5 g to - 3.0 g would apply, provided that local wing structure is revised for this condition.

Investigations of the effects of ballast on fuselage local structure for flight and landing conditions have not been made at this writing.

The roll coupling investigation noted above under "Handling Characteristics" also revealed no appreciable load factor build up.

In addition to the static conditions noted above, the airplane response due to missile firing was investigated. The basic conditions were those of item 3.2 and item 3.3 above. The resulting accelerations about all three axes are appreciably below those used for the structural design of the F-102A airplane.

4. Flutter

Although flutter calculations for the basic F-102A have not yet been completed at higher speeds for symmetric modes and over the entire speed range for anti-symmetric modes, it appears reasonable to assume that the externally mounted Ding Dong configuration would be free from flutter up to at least

[redacted]

M = 1.2. Supporting this assumption are the calculations which have been made up to M = .95 at sea level and M = .97 at 35,000 feet.

5. Missile Installation

The missile pylon structure attaches to the wing fittings provided for the jettisonable tank pylon.

A finite length launch rail approximately three feet long is indicated to alleviate missile "jump" difficulties caused by the up-wash ahead of the wing.

Frangible fairing is added to the tail of the missile to improve the drag characteristics of the installation.

6. Electronics

Some revision to the electronic fire control system may be required to compensate for missile "jump". This may be accomplished by programming the jump angle as a function of flight condition.

7. Fuselage Ballast

As noted in the weight statement, fuselage ballast is arranged to take advantage of the rocket tubes in the missile bay doors for stowing part of the ballast required. The remainder of the ballast is assumed to be positioned at fuselage station 200.

8. Problem Areas

8.1 One of the most direct effects of the external Ding-Dong installation on the effectiveness of the weapon system may be that caused by missile dispersion due to the aerodynamic effects of the under-wing launching position. As mentioned under item 6 above, some revision to the fire control system may be required.

8.2 Detailed structural investigations of the effects of the ballast on the fuselage and of the Ding Dong installation on the wing pylon attachments should be conducted.

8.3 The effects the missile rocket jet pressures and temperatures on the wing structure and on the handling characteristics of the airplane

[redacted]

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[redacted]

should be determined. A Convair test program to survey the jet wake effects of the Ding-Dong on the fuselage and missile bay doors of the F-102A is scheduled for August 1955. The results of these tests will aid in the evaluation of the problem.

8.4 A reduction of up to 180 lbs. in ballast may be made by positioning the external missiles farther forward than shown in this study. However, this appears to create an additional local attachment problem with respect to the wing pylon fittings. Additional structural investigation will be required to determine the practicability of this arrangement.

8.5 The effects of the external missile arrangement on the dynamic characteristics of the airplane should be investigated to a degree greater than noted in this report. In particular, the studies based on less than M = 1.10 should be extended to this value.

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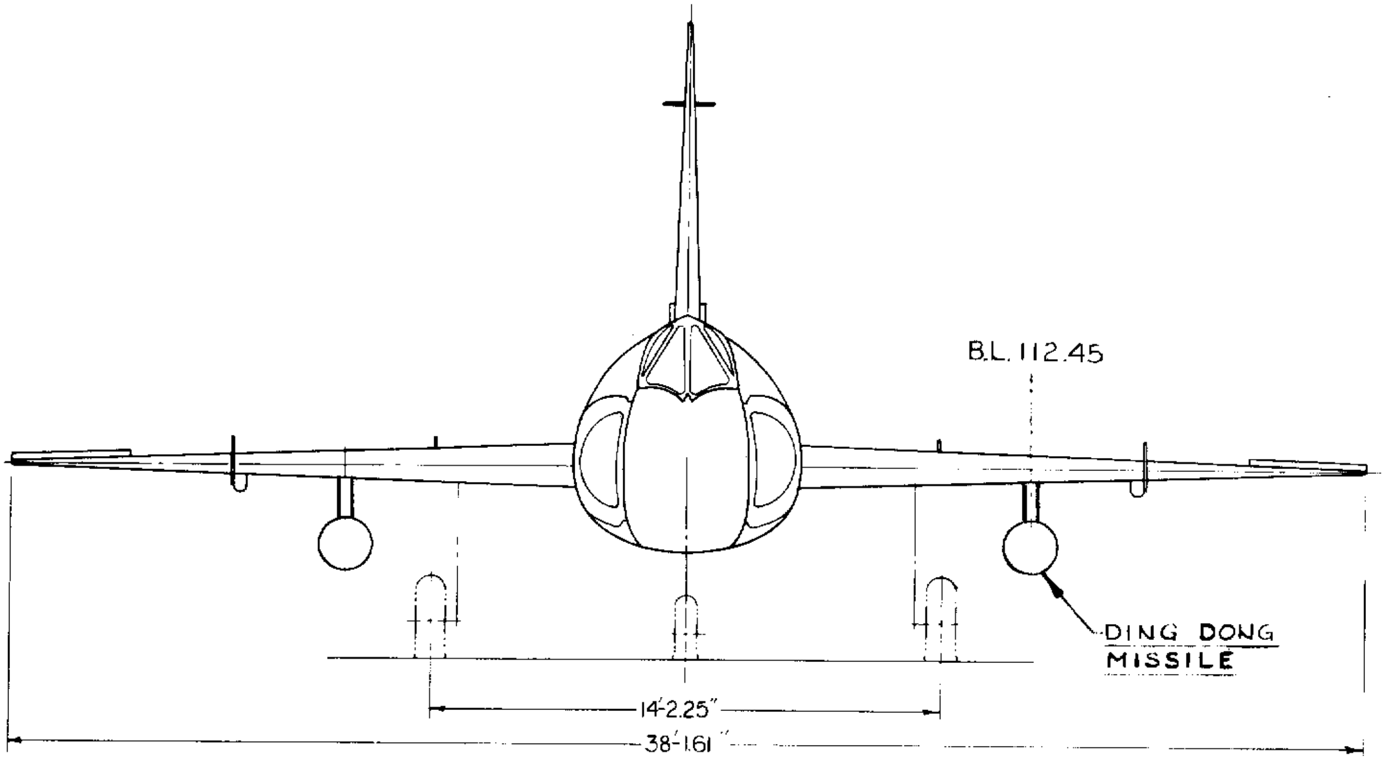
[redacted]

DATA

WING AREA	FT²	695.05
ASPECT RATIO		2.189
AIRFOIL	NACA	0004-65 (MOD)
DIHEDRAL	DEG	0
INCIDENCE	DEG	0
ELEVON'S AREA	FT²	68.00
VERTICAL TAIL AREA	FT²	68.33
ASPECT RATIO		1.10
AIRFOIL	NACA	0004-65 (MOD)
RUDDER AREA	FT²	7.43
ENGINE	P & W	J-57 P-23



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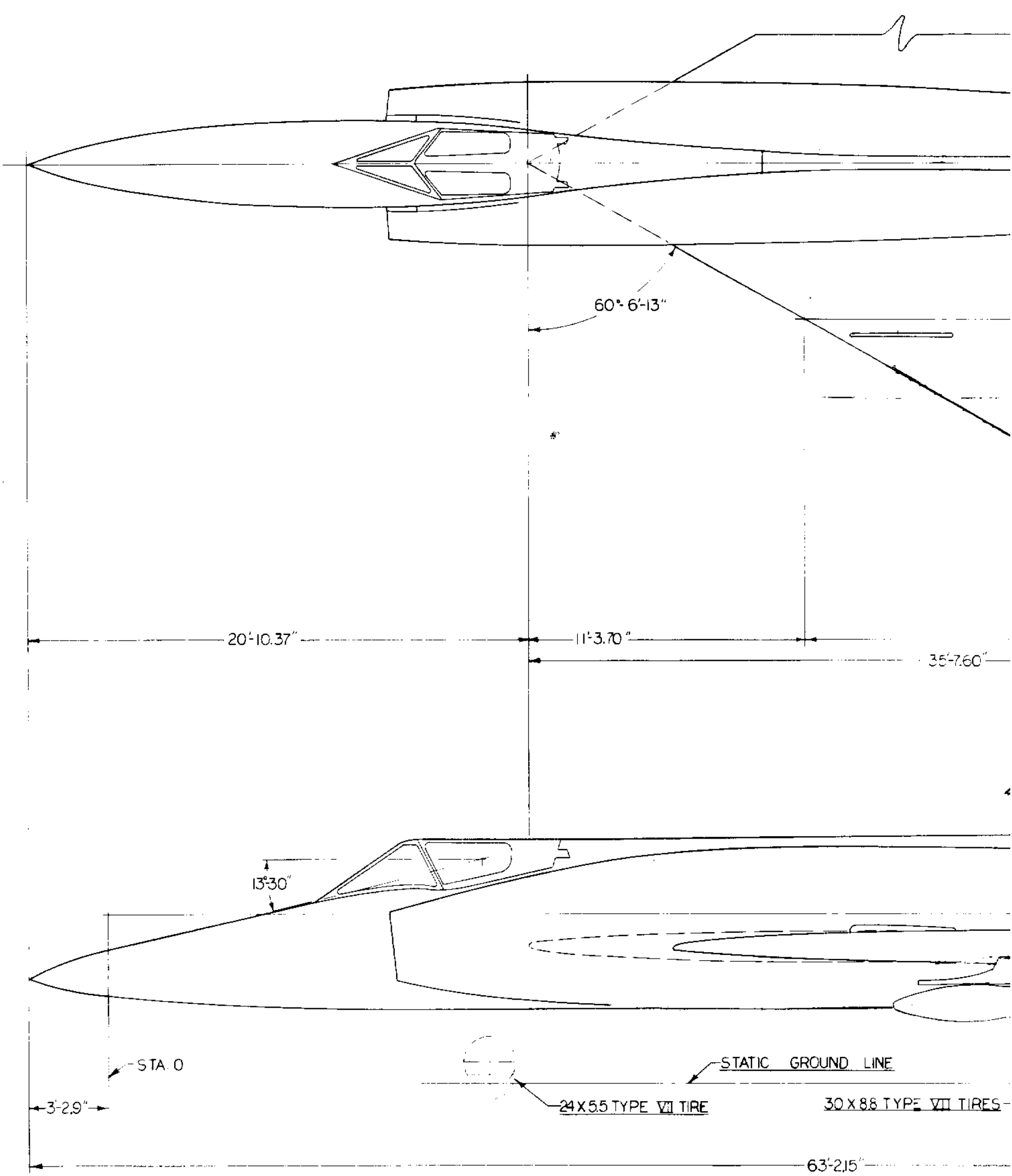

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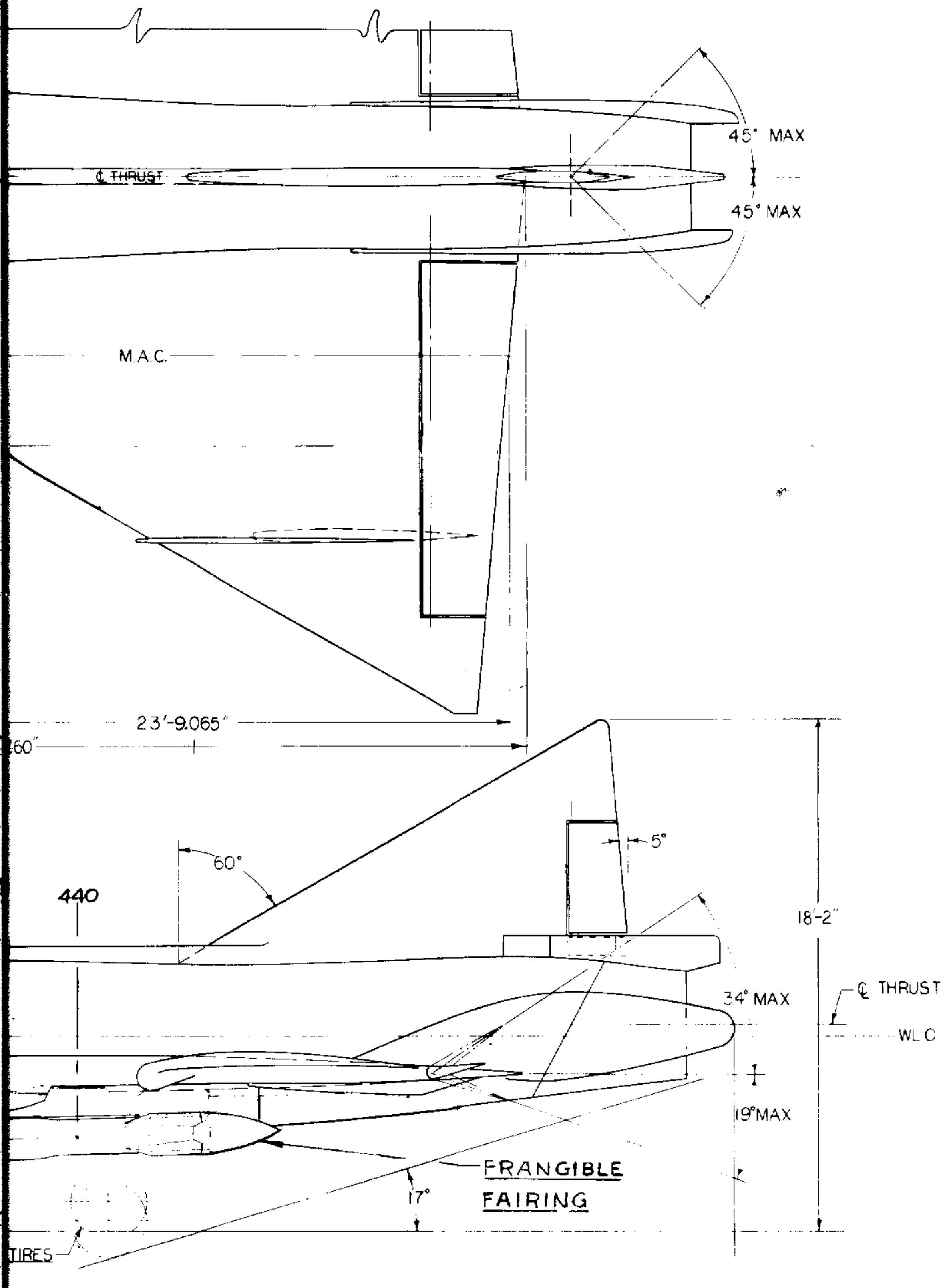

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CONVAIR

PRELIMINARY DESIGN DRAWING

GEN'L ARRANGEMENT
F-102A WITH (2) DING DONG
MISSILES EXTERNALLY MOUNTED

BY Dinkel | CHECKED | APPROVED | SCALE 1/20 | DATE 7 JULY 1955

CONVAIR—SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION
SD-55-08056

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

UN [redacted] IED

REPORT ZP-8-035 (Rev.)
DATE 6 April 1956
MODEL 8

Classification Changed to:
UNCLASSIFIED
Authorized by: 12-14-66
Reclassified by: Dept. Date
Dept. Date
139-1 4-4-67

TITLE

STUDY

TANDEM VERSION OF THE F-102B AIRPLANE

ZP-8-035

1 Feb. 1956
R [redacted] 1956

PREPARED BY R. C. Lingley
R. C. Lingley, Pre-Design

R. E. Craig, Aerodynamics

E. I. Seiden
E. I. Seiden, Weights

GROUP Preliminary Design

REFERENCE

APPROVED BY A. R. Lambert
A. R. Lambert

NO. OF PAGES

NO. OF DIAGRAMS

REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED

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FORM 1812 A-4

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

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REPORT ZP-8-035 (Rev.)
DATE 6 April 1956
MODEL 8

Classification Changed to:
UNCLASSIFIED
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R. H. Lingley 135-1 4-4-67

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FORM 1812A-4

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INTRODUCTION

This is a preliminary study of a tandem version of the F-102B airplane. Complete tactical capability is maintained since the performance approximates that of the basic airplane.

The external configuration is retained with no increase in fuselage length. Change in contour is confined to the canopy and dorsal area.

The existing armament arrangement of one MB-1 weapon and four Falcons will be maintained. The alternate armament capability has been reduced from six to five missiles by providing space for electronic equipment between the two forward missiles.

The existing landing gear, wheels, brakes and tires will be retained. The addition of a second cockpit could, in the case of a tactical version, provide space for the possible future addition of a radar operator as a backup to the proposed fire control system.

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SUMMARY

The addition of the second cockpit dictates a redesign and relocation of some equipment and systems.

The fuselage fuel tank is moved aft and reduced in capacity from 227 to 100 gallons. The difference of 127 gallons is put into the wing "T" tanks to maintain a combat radius more nearly approaching that of the basic F-102B airplane.

The subsonic center of gravity limits are maintained by the addition of fuel in the "T" tanks and by revision of the fuel sequencing system.

The average supersonic center of gravity is approximately 3% further forward than the F-102B airplane since there is less fuel to transfer aft.

The electronic rack containing the missile auxiliaries and landing, communication and navigation equipment will be eliminated. These units will be located in a compartment added between the two forward missiles.

With the tandem seating arrangement, the frontal area of the airplane will be unchanged except for a slightly higher canopy. The increased cross-sectional area through the canopy has a negligible effect on high speed when applied to the area rule theory.

The load factors will be slightly reduced over the comparable load factors of the F-102B interceptor due to the increased weight and further forward C.G. of the tandem version.

The exact values of allowable load factors will be determined at a later date.

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PERFORMANCE SUMMARY

			TANDEM F-102B**	
		F-102B*	1 PILOT	2 PILOTS
High Speed at 35,000 Ft. Alt.	Mach No.	2.0	2.0	2.0
Highest speed upon reaching 60,000 ft. altitude after zoom from lower altitude	Mach No.	1.6	1.5	1.5
Absolute ceiling (take-off weight minus 56% fuel)	Ft.	56,000	54,600	54,300
Combat ceiling (R/C=500 Ft./Min. at point intercept mission combat weight)	Ft.	54,200	52,900	52,600
** *Combat radius (Mil-C-5011A area intercept mission)	N. Mi.	375	350	340

* Based upon the spec. airplane with an armament load of two MB-1 weapons.

** Based upon the weight shown on Page 9 and the C.G. curves shown on Pages 10 and 11.

***(Combat radius shall conform to the MIL-C-5011A Area Intercept Flight Plan with he following qualification: In the event that the combat ceiling at supersonic speeds exceeds the combat ceiling at subsonic speeds, item 4 of MIL-C-5011A Area Intercept flight plan, "Climb on course with maximum power to Combat Ceiling" shall be interpreted as referring to the combat ceiling at subsonic speeds.) Combat radius shown has been computed accordingly.

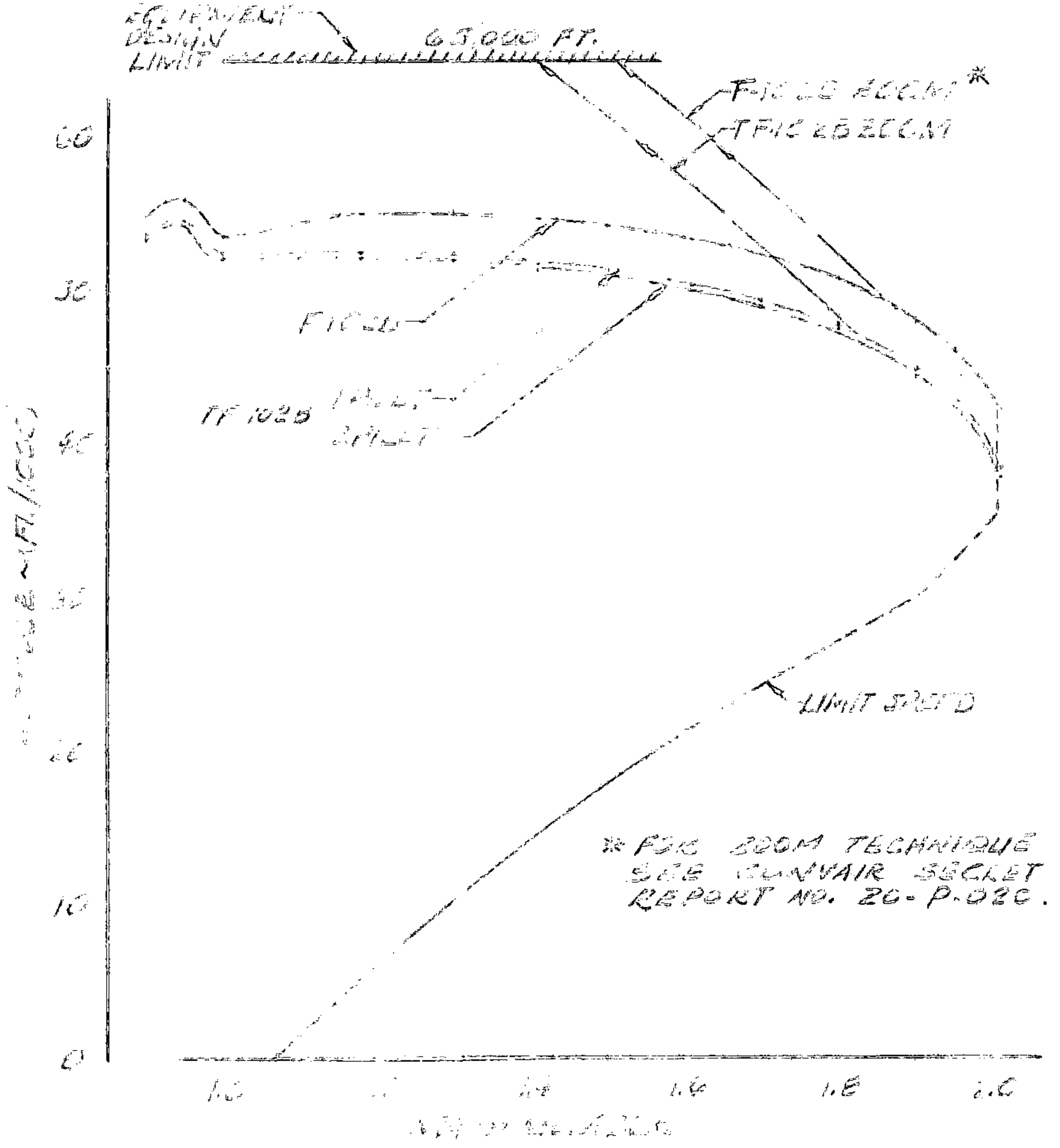
[redacted]

[illegible]

[illegible]

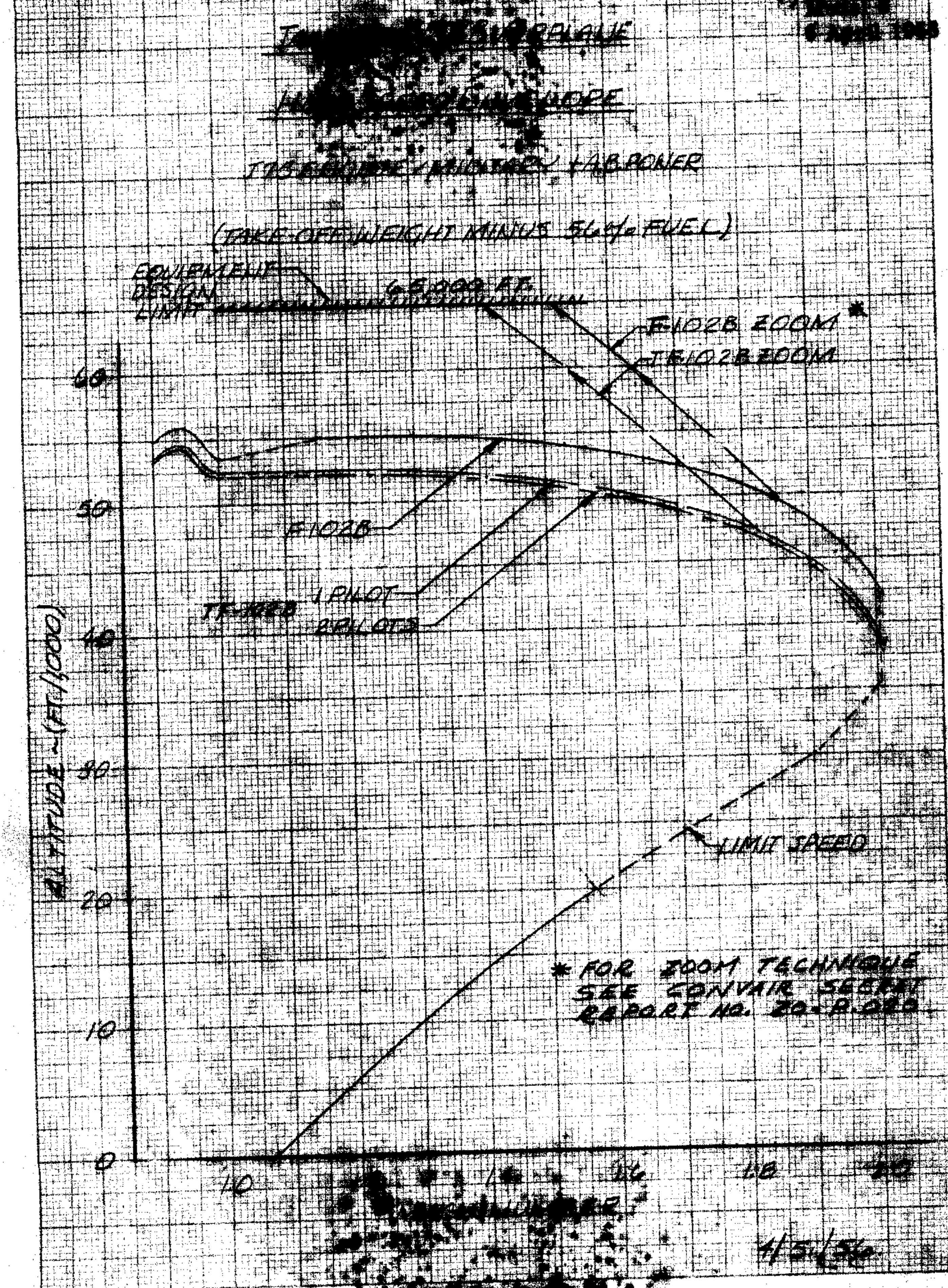
J75 [illegible] MILITARY [illegible]

(TAKE-OFF WEIGHT [illegible] 50% FUEL)



* FOR ZOOM TECHNIQUE SEE CONVAIR SECRET REPORT NO. ZG-P-020.

45 [illegible]



T[redacted] SUPERPLANE

[redacted] ENVELOPE

J75 ENGINE [illegible] + A.B. POWER

(TAKE OFF WEIGHT MINUS 50% FUEL)

* FOR ZOOM TECHNIQUE SEE CONVAIR SECRET REPORT NO. ZA-[illegible]-025

[redacted]

1/5/56

K+E KEUFFEL & ESSER CO. 10 X 10 TO THE 1/2 INCH 359-11G

[redacted]

Discussion:

The following is a brief description of the physical changes required to the F-102B airplane to produce a tandem trainer while maintaining as much of the existing design as possible.

A. General (See Drawing No. SD-56-08011)

1. Add second cockpit immediately behind the present cockpit. Provide 5° down vision from the rear seat over the front head rest in neutral seat position using the proposed Convair "supersonic seat". (If interim airplanes are equipped with existing type seats, approximately 2° down vision from the rear seat will be available.)

2. Move fuselage fuel tank aft and reduce capacity.

3. Move refrigeration unit aft and revise ducting.

4. Move electronic equipment into compartment between two forward missiles.

5. Provide individual launchers for two forward missiles (existing aft launchers may be used).

6. Tie-in control system to second cockpit.

7. Design MA-1 training equipment with duplicate displays for rear cockpit.

B. Fuselage

1. Add new pressure bulkhead and seat rails for second cockpit.

2. Add pressure floor for second cockpit.

3. Delete side electronics access door in area of second cockpit.

4. Delete fuselage fuel tank in area of second cockpit.

5. Delete existing canopy, canopy island and dorsal forward of Sta. 472.

6. Relocate and des[redacted] from windshield parting

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[redacted]

frame to Sta. 316. Provide minimum ejection clearance of 28 inches between longerons in cockpit areas.

7. Relocate and design new fuselage fuel tank from new pressure bulkhead to Sta. 273.0 and between top of missile bay and upper longeron.

8. Provide for new electronic compartments:
   a. In new dorsal between Sta. 273 and Sta. 306
   b. In center forward missile bay.

9. Design new canopy to cover both cockpits. (Retain existing windshield to parting frame) and investigate to determine optimum means of ejection.

10. Design new canopy hinges and canopy island at Sta. 270 approximately.

11. Provide for new canopy latching mechanism.

12. Provide for new canopy operating and removal mechanism at existing pressure bulkhead location.

13. Redesign structure in the area of the added pilot, fuel tank and relocated electronic equipment.

14. Revise missile bay door as required by addition of electronic compartment in forward missile bay.

C. Electronics

1. Relocate electronic missile auxiliaries and landing, communication and navigation equipment to new compartments:

   a. In dorsal above refrigeration unit.

   b. In center forward missile bay.

2. Design new racks, shock mounting and cabling for the above equipment.

[redacted] UNCLASSIFIED

[redacted]

3. Install duplicate displays of MA-1 fire control system in new cockpit.

D. Furnishings (See Dwg. SD-56-08013)

1. Install ejection seat in new cockpit. (Use proposed Convair "supersonic seat" modified to provide 5° down vision from rear cockpit.)

2. Install instrument panel, consoles and other equipment required in new cockpit.

E. Controls

1. Provide side stick, rudder pedals and throttle for new cockpit and tie-in to existing control system.

2. Revise existing control system as required by relocation of other equipment.

F. Air Conditioning

1. Move refrigeration unit aft approximately 7 inches and redesign ducting as required by this change.

2. Reroute ducting as required by addition of second cockpit and relocation of fuselage fuel tank.

G. Fuel System

1. Revise fuel lines to accommodate relocation of fuel tank.

2. Revise fuel system within fuselage fuel tank due to revised shape and size of tank.

3. Revise fuel system sequencing to maintain the airplane's center of gravity within limits.

H. Mechanism

1. Provide [redacted] mechanism.

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[redacted]

2. Provide new canopy operating and removal mechanism at existing pressure bulkhead location.

3. Revise forward missile bay door mechanism due to the addition of the electronic compartment between the two forward missiles.

See following pages for weight and balance information.

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COMPARATIVE GROUP WEIGHT STATEMENTS

F-102B AND TANDEM F-102B

		*F-102B	** Tandem F-102B (2) Pilots
Wing Group		3209	3209
Tail Group		638	638
Body Group		4679	5269
Alighting Gear Group		1226	1239
Surface Controls Group		435	469
Engine Section		37	37
Propulsion Group		(8198)	(8219)
Engine Installation		6125	6125
Air Induction System		891	892
Exhaust System		262	262
Cooling System		46	46
Lubricating System		51	51
Fuel System		725	729
Engine Controls		28	43
Starting System		70	71
Fixed Equipment Group		(4922)	(5570)
Instruments and Navigational Equipment Group		127	166
Hydraulic & Pneumatic Group		375	377
Electrical Group		537	556
Electronics Group		2570	2882
Armament Group		681	741
Furnishings & Equipment Group		271	458
Air Conditioning & Anti-Icing Group		308	335
Auxiliary Gear Group		53	55
WEIGHT EMPTY		23344	24650
Useful Load		(9980)	(10250)
Pilot, Survival Kit, and Parachute		270	540
Fuel-Unusable (JP-4)	36 Gals.	234	234
Fuel-Usable (JP-4)	1240 Gals.	8060	8060
Oil-Trapped & Engine		60	60
MB-1 Type Rocket	(1)	815	815
Ejector Cartridge	(5)	1	1
Falcon Missile GAR 1-A &/or 1-C	(4)	540	540
NORMAL TAKEOFF GROSS WEIGHT		33324	34900

* Based on Convair Status Report dated 4 April 1956.

** The normal Takeoff Gross Weight of the Tandem F-102B with rear pilot removed is 34630 lbs.

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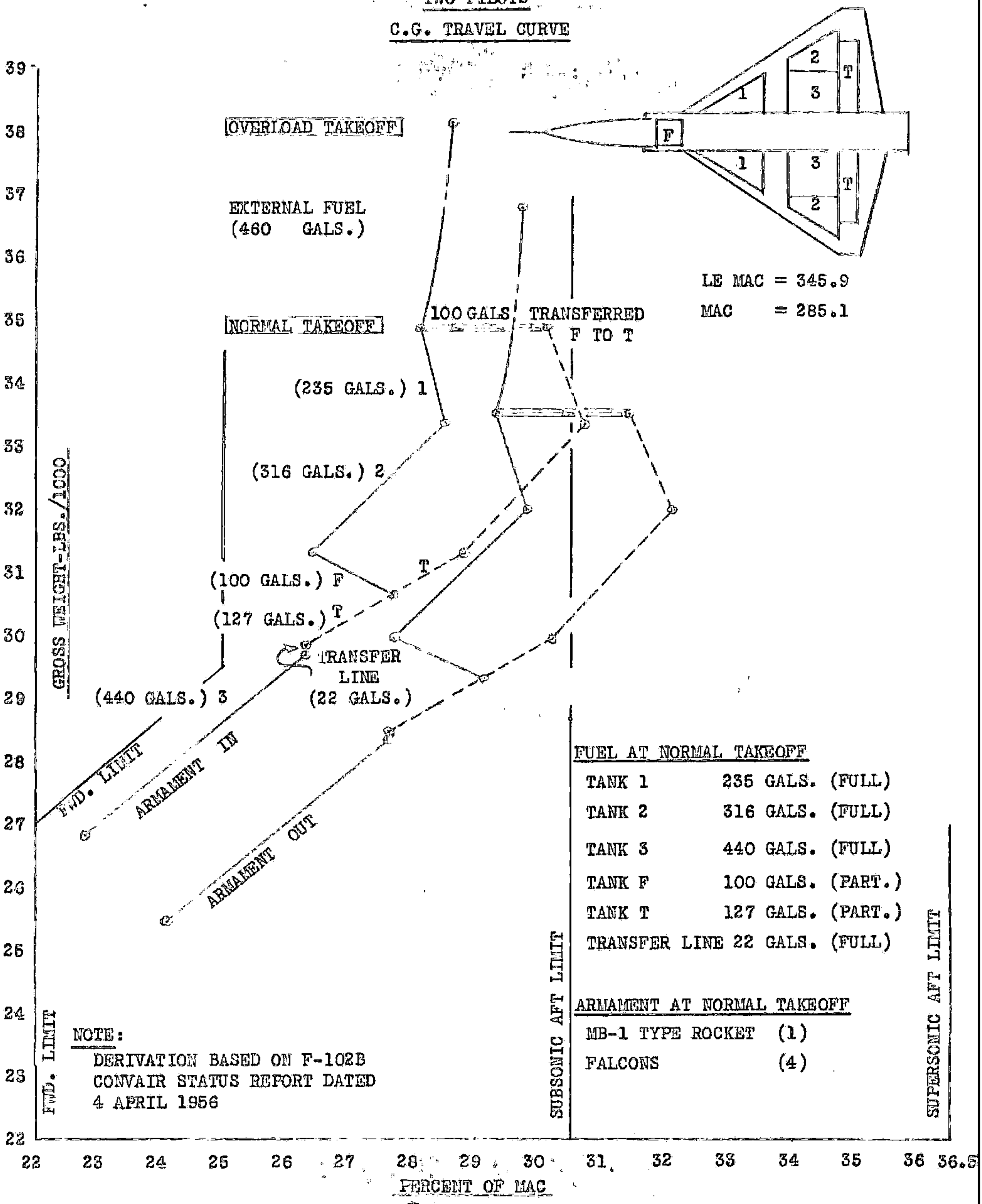
TANDEM F-102B AIRPLANE

TWO PILOTS

C.G. TRAVEL CURVE



FUEL AT NORMAL TAKEOFF

TANK 1	235 GALS.	(FULL)
TANK 2	316 GALS.	(FULL)
TANK 3	440 GALS.	(FULL)
TANK F	100 GALS.	(PART.)
TANK T	127 GALS.	(PART.)
TRANSFER LINE	22 GALS.	(FULL)

ARMAMENT AT NORMAL TAKEOFF

MB-1 TYPE ROCKET	(1)
FALCONS	(4)

NOTE:
DERIVATION BASED ON F-102B CONVAIR STATUS REPORT DATED 4 APRIL 1956

4-4-56

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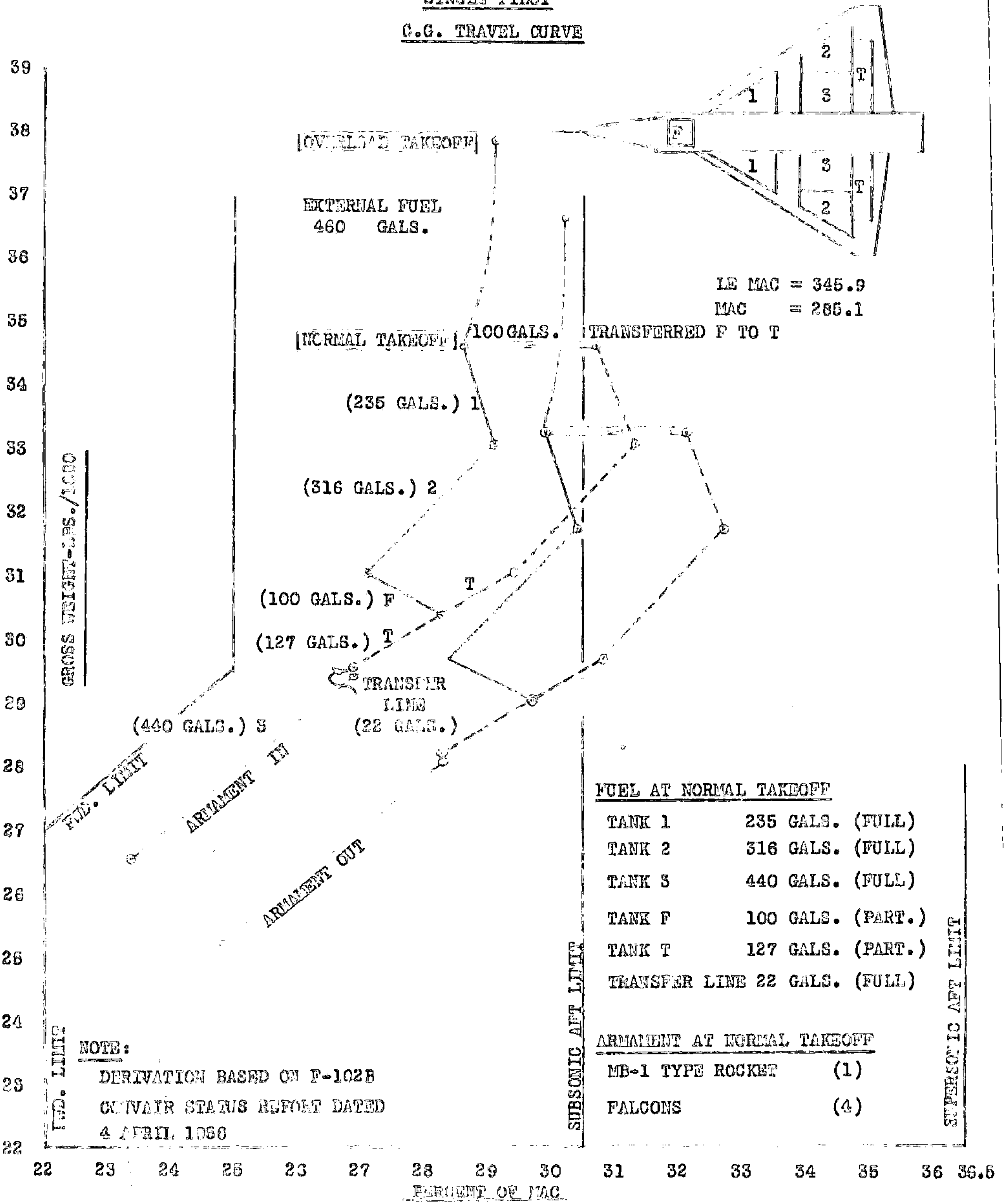
Report No: ZP-8-035
Model: 8
Date: 6 April 19

TANDEM F-102B AIRPLANE

SINGLE PILOT

C.G. TRAVEL CURVE



FUEL AT NORMAL TAKEOFF

TANK 1	235 GALS.	(FULL)
TANK 2	316 GALS.	(FULL)
TANK 3	440 GALS.	(FULL)
TANK F	100 GALS.	(PART.)
TANK T	127 GALS.	(PART.)
TRANSFER LINE	22 GALS.	(FULL)

ARMAMENT AT NORMAL TAKEOFF

MB-1 TYPE ROCKET	(1)
FALCONS	(4)

NOTE:
DERIVATION BASED ON F-102B
CONVAIR STATUS REPORT DATED
4 APRIL 1956

4-4-56

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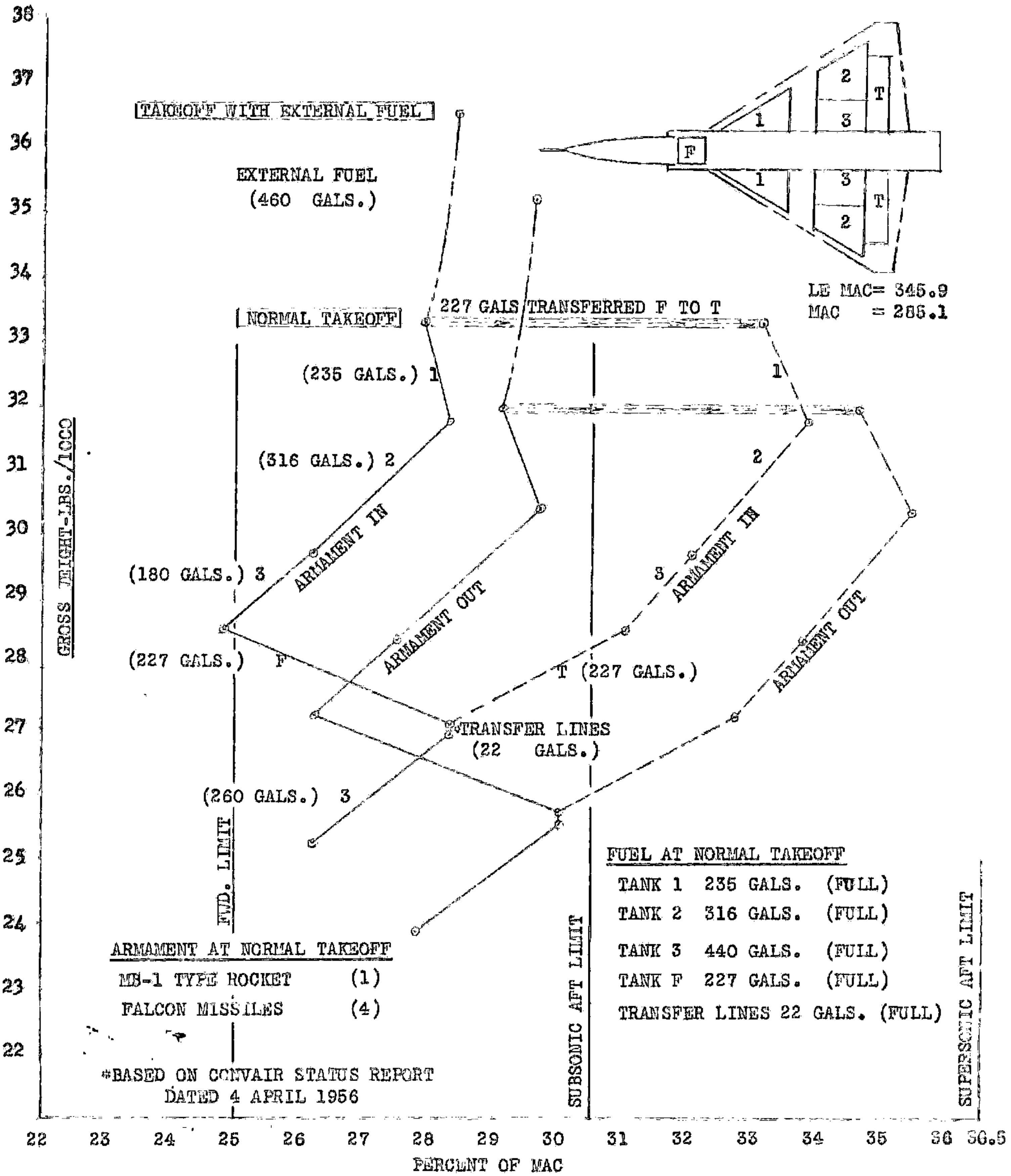
Date: 6 April 1956

F-102B
*C.G. TRAVEL CURVE

(This Curve Included for Reference Only)



ARMAMENT AT NORMAL TAKEOFF

MB-1 TYPE ROCKET	(1)
FALCON MISSILES	(4)

FUEL AT NORMAL TAKEOFF

TANK 1	235 GALS.	(FULL)
TANK 2	316 GALS.	(FULL)
TANK 3	440 GALS.	(FULL)
TANK F	227 GALS.	(FULL)
TRANSFER LINES	22 GALS.	(FULL)

*BASED ON CONVAIR STATUS REPORT DATED 4 APRIL 1956

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SECRET

ZP-8-035

VAULT

C O N V A I R
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-510
DATE 16 August 1957
MODEL F-106

[redacted]

TITLE

UNCLASSIFIED

THE PROPOSED
F-106C INTERCEPTOR

ZP-8-510

PREPARED BY R. C. Lingley
R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert
Chief, Pre-Design

NO. OF PAGES 13
NO. OF ~~DIAGRAMS~~ DWGS 3

REVISIONS

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FORM 1012A-4

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B. Speed vs. Altitude 10

C. Combat Radius 12

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INTRODUCTION

Included herein is preliminary data describing the proposed F-106C Interceptor. This airplane was formerly designated the Improved F-106, Configuration #5. (Ref. Convair Report ZP-57-08000, Addendums I and II)

This report includes only that information pertinent to the F-106C thereby eliminating any confusion that may exist due to the many versions of the Improved F-106 that have been previously proposed.

The F-106C interceptor has a performance capability of Mach 2.50 at 60,000 feet altitude. The use of zoom or snap up techniques will provide an effective attack capability against a target up to 80,000 feet altitude. The fire control system and armament are compatible with this mode of operation.

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II. SUMMARY

The F-106C Interceptor is recommended as the logical follow-on to the F-106A. It is scheduled for first flight in late 1959 and production build-up in 1960. (Schedules will be submitted at a later date.)

The F-106C incorporates improvements in performance, fire control system (including a 40" dish), and armament, which provides the maximum improvement in weapon system effectiveness in its time period.

The requirement for a one or two-place version has not been finalized. The radar operator and equipment can easily be incorporated into the airplane. Either version will have the same internal fuel capacity. Performance differences will be as follows:

Performance Comparison (Internal Fuel Only)

			F-106C	
		F-106A	One-place	Two-place
Take-off gross weight	Lb.	33,347	39,450	40,950
Internal fuel	Lb.	8,060	11,000	11,000
Absolute ceiling	Ft.	56,000	60,000	59,200
Time to accelerate (M = .95 to 1.9)	Mins.	3.2	2.9	3.1
High speed	Mach	2.00	2.50	2.50
Take-off speed	Knots	157	172	175
Take-off distance **	Ft.	3,900	4,750	5,000
Combat radius *	N.Mi.	375	515	485

NOTE: Two 230 gallon external tanks will increase the one-place F-106C combat radius to 710 nautical miles.

* MIL-C-5011A Area Intercept Mission (Modified) See page 5

** Over 50 ft. obstacle

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III. DESCRIPTION

A. General

The F-106C includes the following features which provide for improved weapon system effectiveness:

P & W JT4B-22 engine (Advanced J-75)
Internal compression inlets
Variable ejector
Hughes 40" radar dish and modified MA-1 system
Infrared augmentation to fire control system
(1) MB-1 rocket + (4) GAR-3/4 Falcons
Increased internal fuel capacity (from 8060# to 11000#)

The F-106C will be an aluminum airplane designed for full load factors of (7g) up to the restricted speed (M=1.75) and reduced load factors of (5.33g) to the limit speed (M=2.50). This is consistent with the present F-106A criteria.

Retention of the aluminum construction, eliminates the necessity of costly retooling for steel or titanium construction. The aluminum structure provides a total accumulative airframe life of 35 hours at V max. Based on a 5 minute combat time this provides for 420 sorties at Mach 2.50.

The F-106C airplane is aerodynamically similar to the F-106A except for the internal compression inlets and variable ejector. Thus a minimum flight test program is assured.

The existing F-106A wing planform with case XXIX camber is retained. Strength is increased to provide for increased gross weight and temperature effects at higher speeds.

[redacted]

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[redacted]

The drag estimate used in determining the F-106C performance was based on the following:

1. F-102A and F-106A flight test data.
2. F-106A wind tunnel tests of case XXIX camber.
3. F-106A/Advanced F-106 wind tunnel tests with internal compression inlets.

Thrust is provided by an advanced J75 (JT4B-22) Engine with internal compression inlets and a variable area divergent ejector. The inlet characteristics, pressure recovery and mass flow ratios were estimated from wind tunnel tests. The performance of the JT4B-22 engine was determined using Pratt and Whitney specification MR 401. Estimates of performance of a divergent ejector were made from tests made at NACA.

Design has been initiated and flight testing planned for development of a variable ejector for the F-106A.

Present planning calls for full scale wind tunnel tests at NACA, Cleveland, to develop the control system for the internal compression inlets.

The three view on the following page shows the external configuration of the F-106C airplane.

[redacted]

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B. Internal Arrangement

The internal arrangement of the F-106C differs from the F-106A as follows:

1. The nose section is revised to provide for the 40" dish and larger radome.
2. Cockpit is moved forward to maintain visibility over the nose.
3. Electronic components are relocated into compartments created by the cockpit movement.
4. The fuselage mid-section is similar to the F-106A except for incorporation of the internal compression inlets.
5. The aft section of the fuselage is revised to provide for the larger JT4B-22 engine and variable ejector.
6. Increased gross weight requires additional speed brake area. Two speed brakes will be added to the lower fuselage section in addition to the existing brakes at the base of the fin.
7. The drogue chute will be increased from 14 feet to 20 feet diameter to maintain existing F-106A ground roll on icy runways.
8. Dual wheels will be incorporated to maintain existing UCI ratings.

The inboard profile drawing on the following page shows the internal arrangement of the F-106C, including the one or two-place arrangements.

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C. Fire Control System

A Hughes 40" radar dish and modified MA-1 system are proposed.

D. Armament

The MB-1 Rocket plus (4) GAR -3/4 Falcons is considered as basic armament. In addition one (1) whing ding weapon can be substituted for the MB-1 when and if it becomes available.

The drawing on the following page shows a few of the typical armament arrangements possible in the F-106C. The GAR-Z weapons shown represent a typical guided atomic weapon. Other weapons can be carried in the same space, such as Sparrow -X or Diamondback. These type weapons could be phased into the F-106C later in the program.

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E. Weight Comparison

	F-106A One place	F-106C One place	F-106C two place
Wing Group	3,338	4,222	4,222
Tail Group	684	784	784
Body Group	4,525	5,275	5,862
Landing Gear	1,186	1,454	1,463
Surface Controls	467	467	467
Nacelle Group	38	38	38
Propulsion	(8,065)	(8,860)	(8,861)
Engine Installation	5,802	6,096	6,096
Air Induction	1,071	1,471	1,471
Exhaust System	276	287	287
Cooling System	55	55	55
Lubricating System	53	103	103
Fuel System	718	758	758
Engine Controls	26	26	26
Starting System	64	64	65
Instruments & Equipment	141	141	152
Hydraulics & Pneumatics	377	457	457
Electrical Group	576	599	634
Electronics Group	2,492	2,557	2,816
Armament Group	681	730	757
Furnishings & Equipment	409	439	695
Air Conditioning	356	431	466
Auxiliary Gear	58	58	58
Weight Empty (Lbs.)	23,394	26,512	27,732
Useful Load	(9,953)	(12,938)	(13,218)
Crew	280	280	560
Oil (Trapped & Engine)	60	60	60
Fuel (Unusable)	174	219	219
Fuel (Usable)	8,060	11,000	11,000
Armament	(1,379)	(1,379)	(1,379)
One MB-1 & Cartridge	829	829	829
4 GAR -3/4	550	550	550
Gross Weight (Lbs.)	33,347	39,450	40,950

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IV. PERFORMANCE

A. Performance Comparison (Internal Fuel Only)

			F-106C	
		F-106A	One-place	Two-place
Take-off gross weight	Lb.	33,347	39,450	40,950
Internal Fuel	Lb.	8,060	11,000	11,000
Absolute ceiling	Ft.	56,000	60,000	59,200
Time to accelerate (M = .95 to 1.9)	Mins.	3.2	2.9	3.1
High speed	Mach	2.00	2.50	2.50
Take-off speed	Knots	157	172	175
Take-off distance **	Ft.	3,900	4,750	5,000
Combat radius *	N.Mi.	375	515	485

NOTE: Two 230 gallon external tanks will increase the one-place F-106C combat radius to 710 nautical miles.

* MIL-C-5011A Area Intercept Mission (Modified)

1. Take-off allowance:
   1 min. max. power at sea level, V = 0
   2 min. N.R.P. at sea level, V = 0

2. Military power climb to cruise altitude.

3. Cruise Out.

4. Max. power climb to subsonic combat ceiling.

5. Combat allowance:
   Fuel required to accelerate with maximum power for 5 mins. at 50,000 ft., starting from subsonic max. power climb speed.

6. Cruise home.

7. Landing reserves: 5% Initial fuel plus 20 min. loiter at S. L.

[redacted]

** Over 50 ft. obstacle

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[redacted]

B. Speed vs. Altitude

On the following page is a speed/altitude comparison of the F-106A and F-106C airplanes.

A noticeable improvement in speed and altitude capability is apparent for the F-106C.

High-Altitude Combat Effectiveness

Analog simulation studies * have indicated that the F-102 armed with falcon missiles can achieve target kills at 60,000 feet utilizing a snap-up attack from 40,000 ft.

Preliminary studies on the F-106A indicate that effective kills can be achieved against targets at 75,000 feet by initiating a snap-up attack from 40,000 to 45,000 ft.

Based on these results it is expected that the F-106C will have a highly effective snap-up attack capability against targets up to 80,000 ft.

* Convair Report 0R-8-62/DC-8-434

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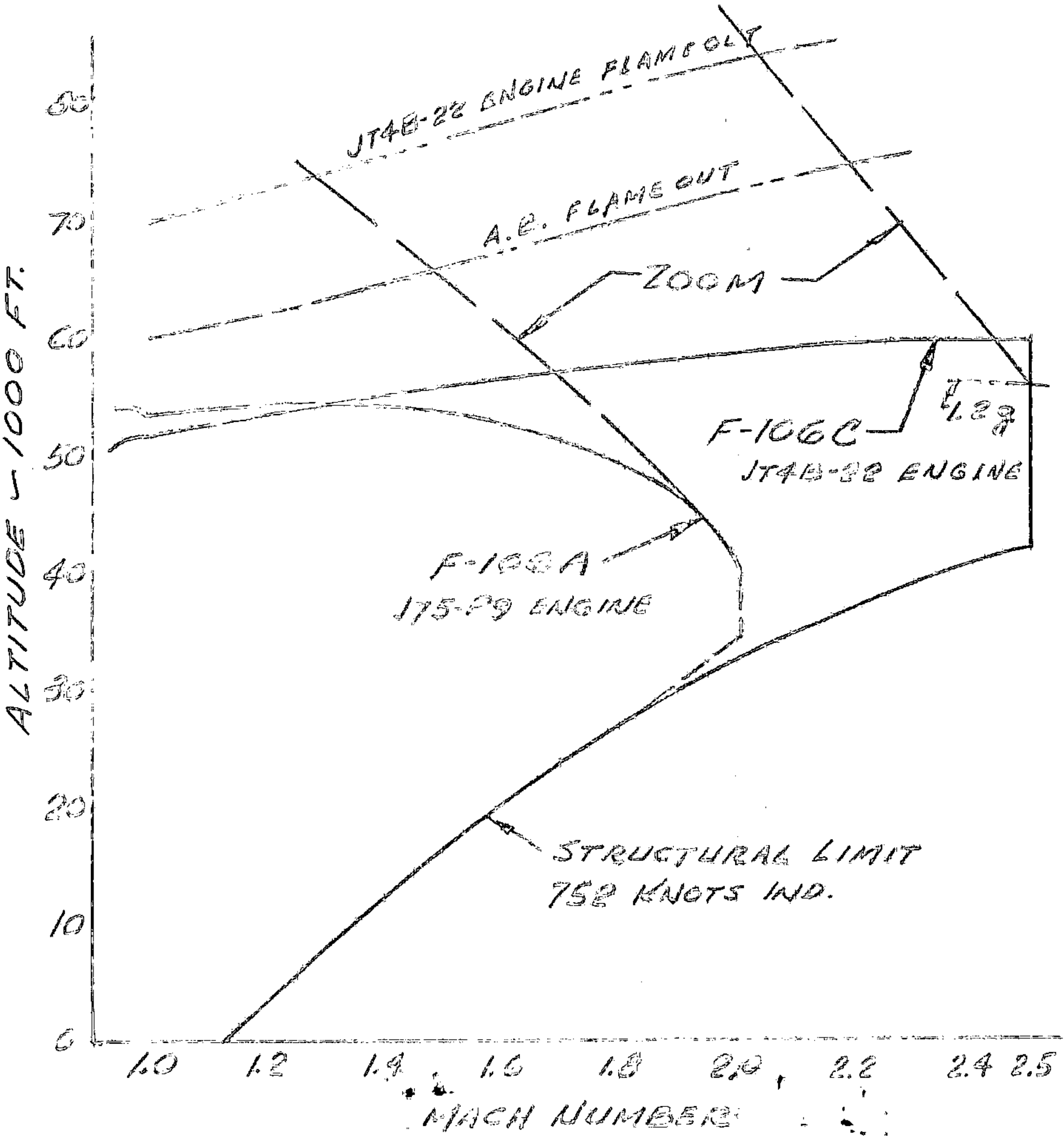
SPEED ALTITUDE COMPARISON

F-106A – F-106C

ONE PLACE



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C. Combat Radius Comparison

On the next page is a comparison of combat Mach number vs. radius for the F-106A and F-106C.

Mission Description:

1. Warm up, take-off and accelerate to best climb speed.
   2 min. N.R.P.
   1 min. AB power.
2. Mil. power climb to cruise altitude.
3. Cruise at best speed and altitude.
4. Accelerate at 35,000 feet to combat Mach number.
5. Climb to 1.2g ceiling.
6. Combat at 1.2g ceiling.
7. Cruise back at best speed and altitude.
8. Land with SAGE reserves:
   10 min. loiter at S. L. and
   15 min. loiter at 35,000 feet.

NOTE: The two minute combat shown on the curve for the F-106C is based on utilizing the cut-off collision course tactic instead of an offset turn as discussed in F-106A SAGE integration conference. The 40" dish and Improved MA-1 system permits this tactic.

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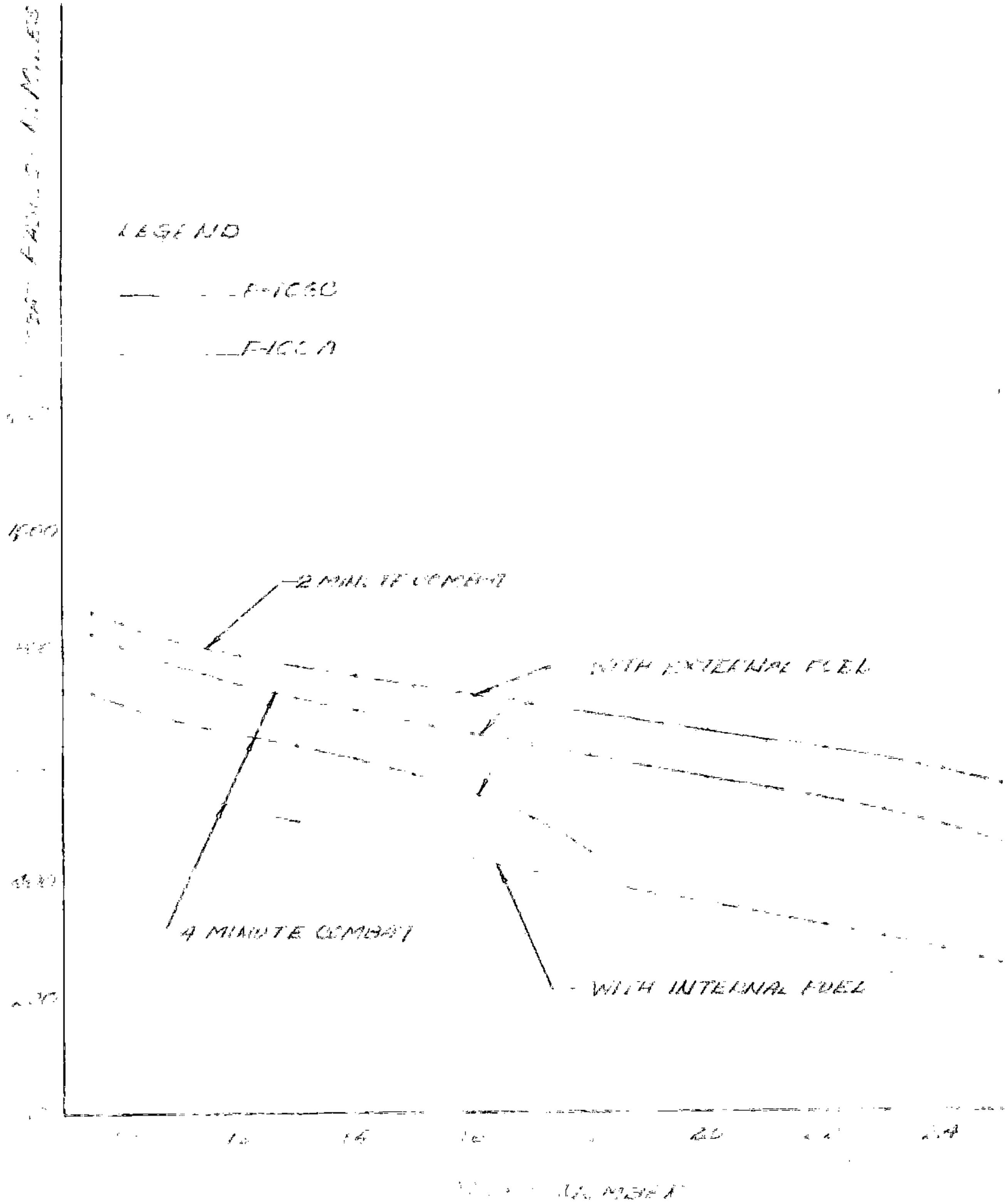
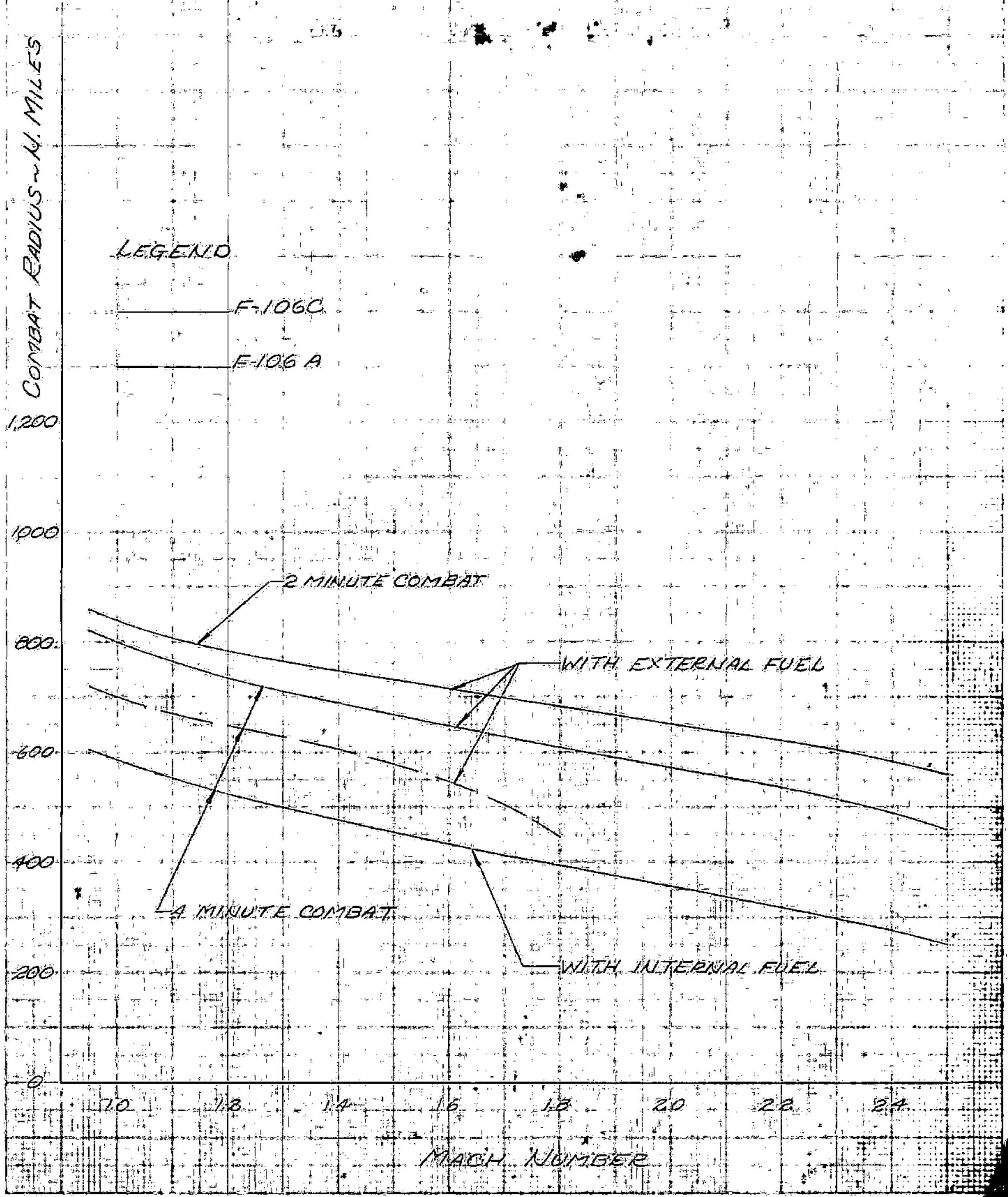
COMBAT RADIUS COMPARISON



COMBAT RADIUS COMPARISON

F106A — F-106C



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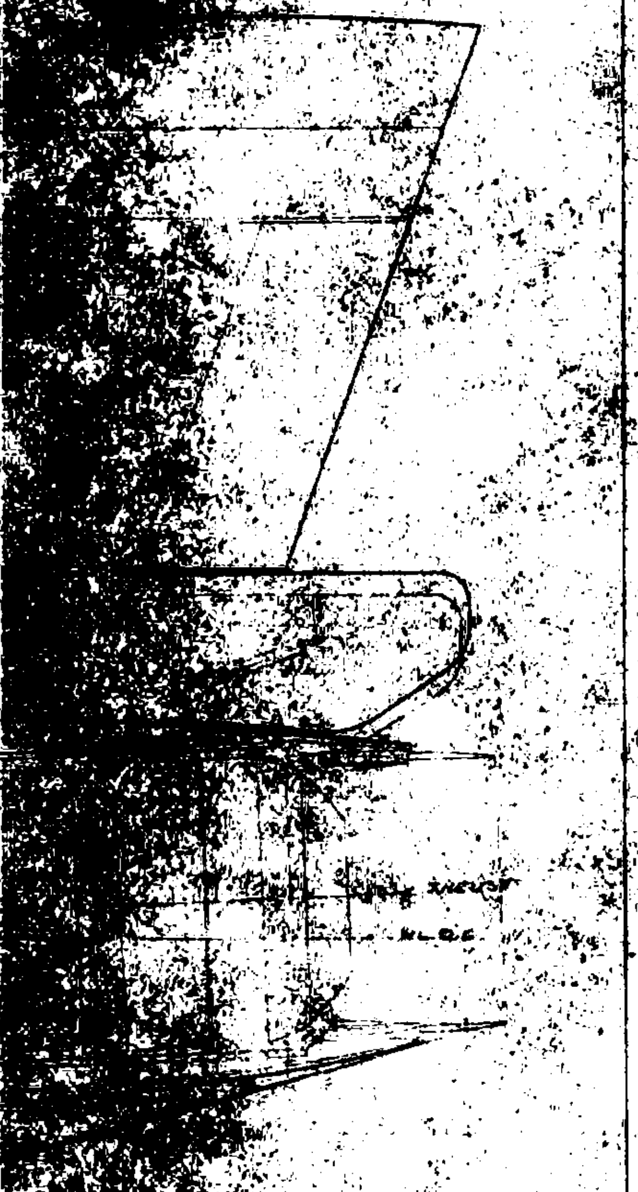

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F-106C

40 DISH &

ENGINE

SCALE 1/10

SD-57-08066

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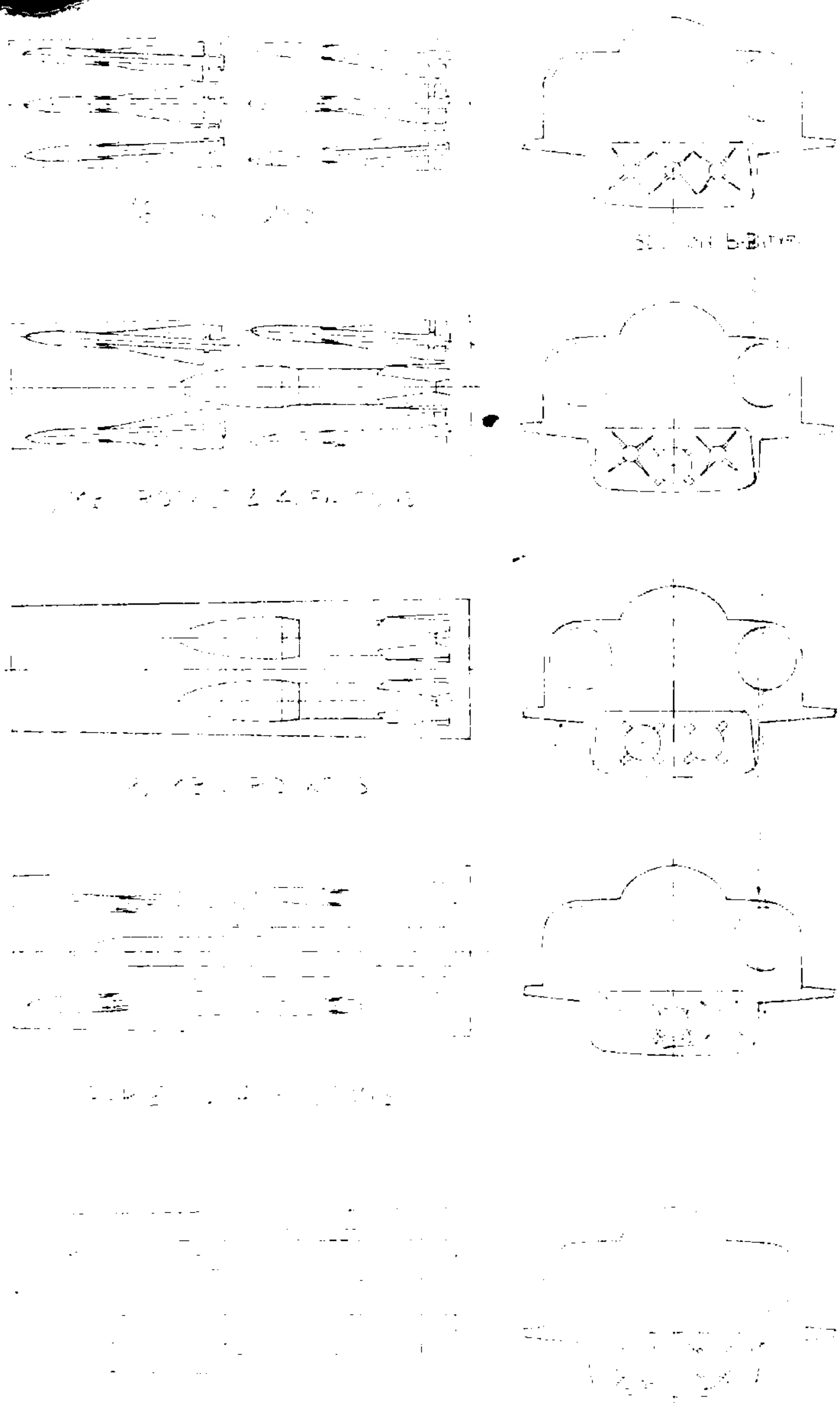
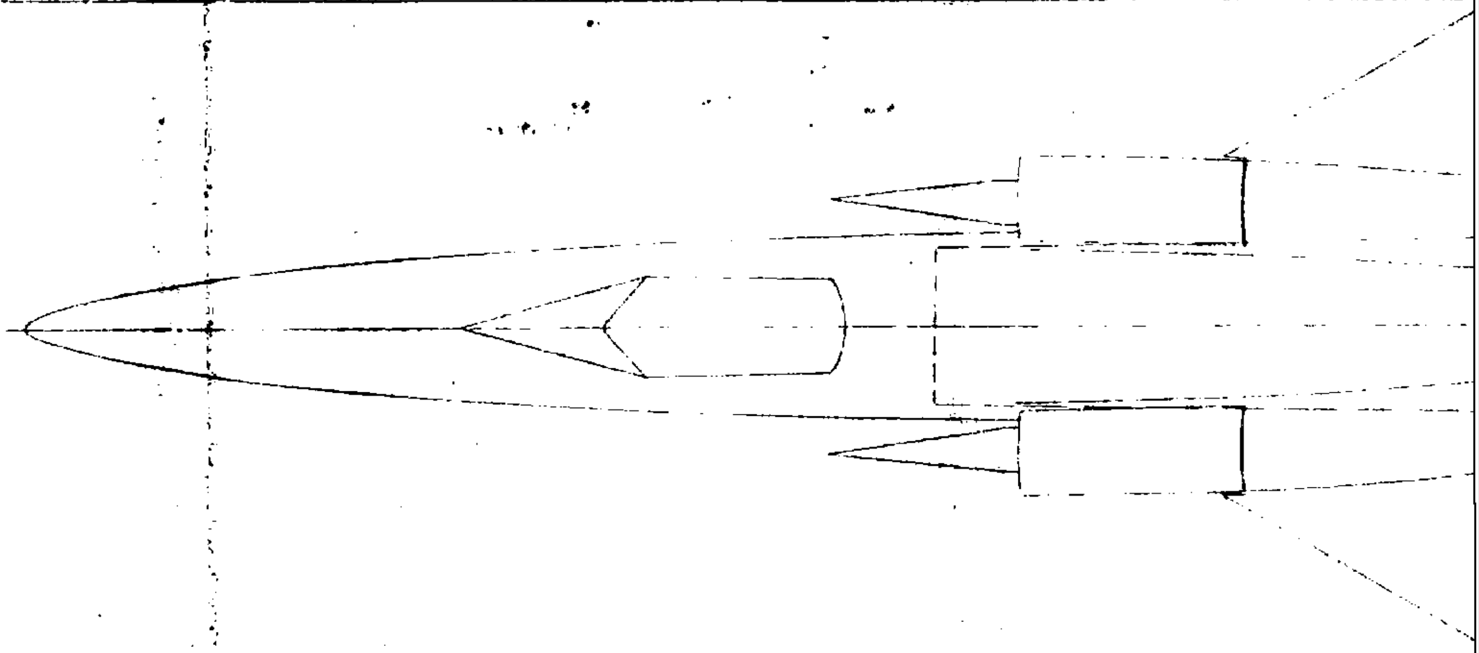
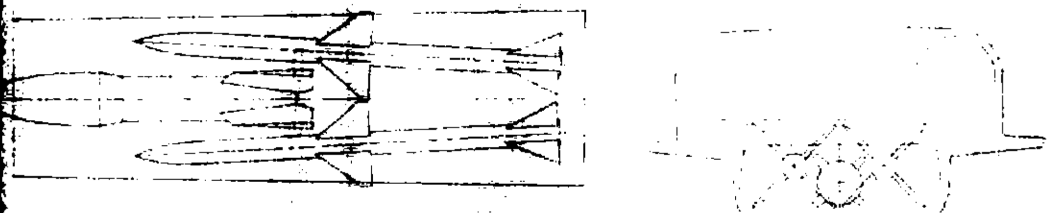
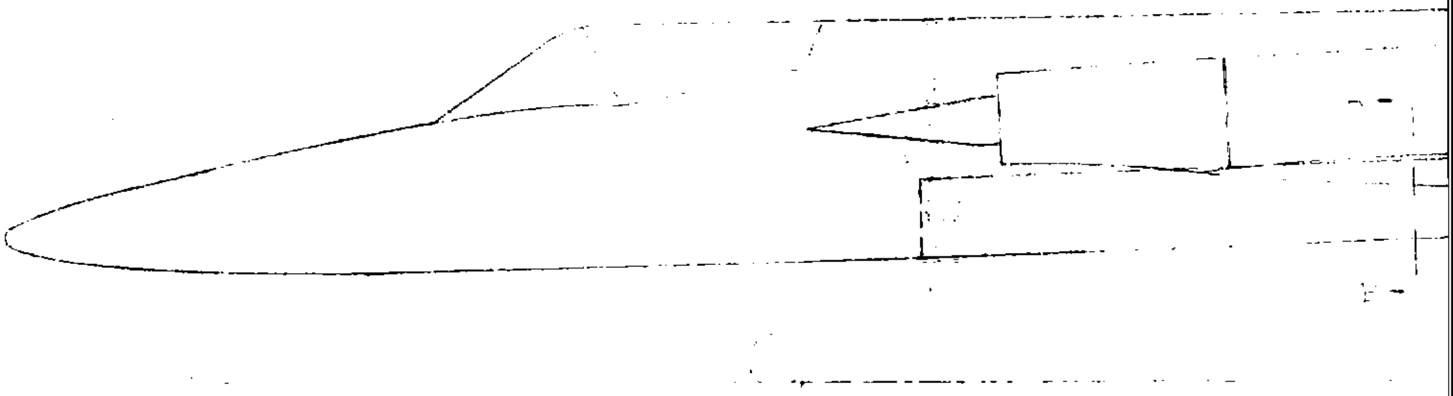


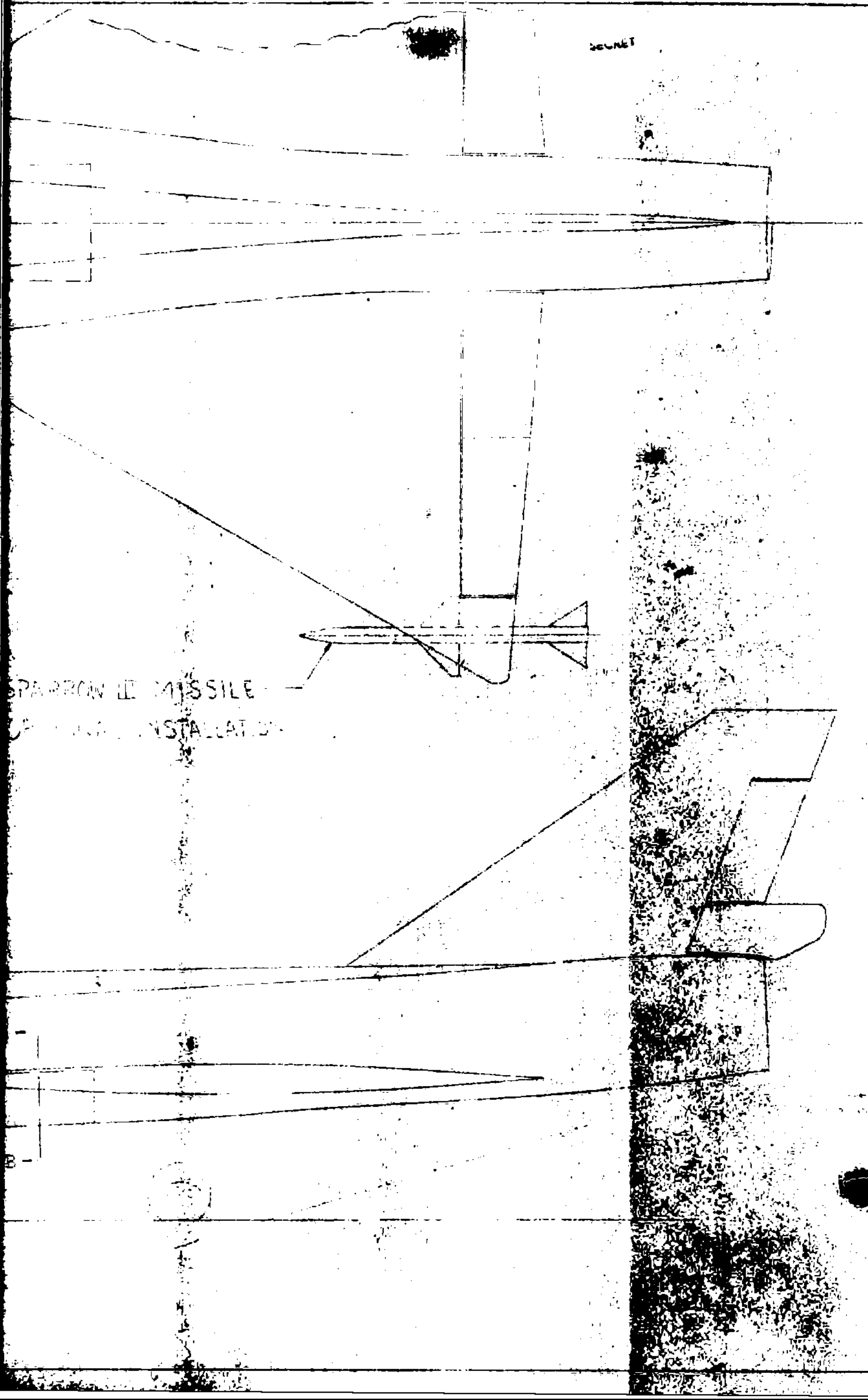




(2) SPARROWS & (1) MB-1 ROCKET

SPAR





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ONVAIR

ARY DESI[illegible] [illegible]WING

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F106C AIRPLANE

SAN DIEGO, CALIFORNIA [illegible]

ZP-264A

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CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-264 A
DATE 19 February 1959
REVISED: 3 March 1959
MODEL

TITLE

PRELIMINARY COMMENTS AND (U)
EVALUATION OF BELL XF-109
19 FEBRUARY 1959
REVISED 3 MARCH 1959

REPORT NO. ZP-264

PREPARED BY Technical Groups
GROUP

COORDINATED BY A. R. Lambert
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	3 Mar. 1959	J.C.Ramsey	Revised and up-dated	All

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FORM 1812 A-4

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Revised: 3 March 1959

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INTRODUCTION

Convair has been requested to enter a joint effort with Bell Aircraft to produce the XF-109 airplane. This airplane, a VTOL tactical fighter, is described in Bell Brochure, Report No. 2000-953001, dated 17 February 1959.

In order to evaluate the engineering effort and determine its status, a Convair engineering team was sent to Bell Aircraft Corporation to gather data. Subsequently a Bell engineering team visited Convair-San Diego. Members of the Convair evaluating team that visisted Bell included the following:

J. E. Bowen	Assistant Electronic Engineer
D. I. Hixson	Design Weight Engineer
C. E. Chapman	Chief of Thermodynamics
C. C. Pope	Tool Project Engineer
H. J. Richardson	Military Requirements Engineer
P. D. Ferrara	Chief Design Engineer
A. R. Lambert	Chief of Preliminary Design
N. R. O'Brien	Design Specialist (Aerodynamics)
H. K. Cheney	Engineering Staff Specialist
C. O. Ekrem	Chief of Stress
Y. Weisman	Senior Dynamics Group Engineer

Members of the evaluating team that visited Convair included the following:

J. W. Rane, Jr.	Director of Engineering and Sales
R. Goldin	Chief, Structural & Mechanical Design Dept.
F. J. Mambretti	Chief, Weapon System Section
J. A. O'Malley	Chief, Aerodynamic & Propulsion Design Dept.
V. B. Paxhia	Chief, Advanced Design Dept.

A preliminary report of the comments are assembled herein.

The XF-109, except for the engine pods on each wing tip, is of conventional appearance and is somewhat similar to a F-104 in that the straight wing and landing gear give this impression.

The airplane gains its vertical capability by the use of jet thrust from two vertical engines mounted just aft of the pilot, two engines with afterburners mounted on each wing tip in rotatable pods, and two engines

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conventionally installed in the aft fuselage section. The aft engines are provided with a modulating diverting valve between the engines and afterburners so as to give downward thrust. In the VTO mode, lateral and yaw control are provided by the use of reaction jets at the respective extremities. Longitudinal control is provided by the throttling of the forward or aft engines. All action is from control stick and rudder pedals.

Transfer from the VTO mode to normal, or vice versa, is by modulating the rotatable wing tip pods and the aft diverter valve and either stopping or starting the two forward engines, dependent upon which mode is selected.

Preliminary surface impressions are that the VTO concept has been well thought out in specific areas and that design assumptions were based upon data from two previous experimental VTOL airplane and laboratory tests.

Individual representatives of the engineering team have included their comments covering their respective fields.

Basically the preliminary comments lend themselves to the following areas:

1. What is the environment for the airplane?
2. What tactical provisions must be added?
3. The effect of adding the weight of possible omitted items, or items necessarily redesigned by additional development testing.
4. The effect of adding the weight of equipment requested by the WADC labs.
5. How firm is the engine data?
6. How valid is the proposed schedule?

The engineering team has been severely handicapped in that Bell has concentrated upon a prototype airplane to be followed by the tactical version. The tactical version appears to have a multiple capability. However, no studies were available which tended to show its actual capability and operational use as a weapon or over-all weapon system. Also, no support studies were available. Until this data is generated only its mechanical capability can be evaluated.

Engineering team comments follow, in summary, and then in detail.

SUMMARY OF COMMENTS

(Under Existing Ground Rules)

Systems Analysis

If a manned fighter/bomber is considered a requirement, the XF-109 appears to fulfill the technical requirements in broad terms. That is, there appear to be no gross technical deficiencies. Further, the dispersed basing concept which is made possible through the VTOL capability, should contribute significantly to its survival probability. However, dispersed basing raises numerous questions concerning operational support, maintenance and logistics particularly of an eight-engined aircraft. Such questions indicate the need for more detailed analysis of the TAC mission together with the XF-109 and its employment concept.

The XF-109 shows an advantage over conventional aircraft in operating at the DEW line with its ability to land at the DEW line under all-weather conditions.

Electronics

1. The electronic system of the airplane will adequately perform the required mission and it is highly recommended that Convair do everything practical to put this airplane and system into production.
2. It is recommended that Convair be assigned as General Electronic System Manager for Testing, Checkout and Flight Test.

Aerodynamics

1. Examination of longitudinal stability,data from wind tunnel tests indicate instability at transonic Mach numbers at sea level for the rigid airplane.
2. The horizontal tail is inadequate for longitudinal trim at Mach numbers above about 1.5 and at high altitude.
3. The current configurations is directionally unstable at high Mach numbers.

4. Specified low speed performance appears obtainable.

5. Specified high speed performance may be deficient in range. According to our concepts, an adequate amount of fuel for take-off and landing requirements has not been used and the drag basis at high supersonic speeds is doubtful. (Estimated radius reduction 22% in supersonic region.)

6. Acceleration and supersonic ceiling will also be decreased due to the difference in estimated wave drag.

7. Aerodynamic data could be available to permit start under a risk basis in approximately 4 months after go-ahead, or 6 months normal basis. Wind tunnel testing would be completed in 12 months.

Dynamics

1. Areas under question might effect the weight by an increase of 1,400 lbs. This would be the addition of acoustic protection.

2. Stability and control studies might necessitate configuration changes in the empennage area.

3. Dynamic investigations might delay release of structure by approximately 3 months.

Thermodynamics

1. Static and Low Speed Performance

Performance is marginal for any above standard temperature day. Full consideration has not been taken of the reaction control thrust loss and of the effects of ingestion of exhaust gases. See Figure 1.

2. High Speed Performance

Inlet recoveries while high are not inconsistent with test data. Inlet size and engine airflow will have to be watched carefully. Ejector nozzle air source does not appear to be adequately worked out.

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3. Aerodynamic Heating

Good coverage for steady state conditions. Considerably more transient detail needed. Engine cooling has to be watched as a source of weight increase.

4. No anti-icing has been considered.

5. If immediate go-ahead were to be obtained it would require 4 to 6 months to firm up the engine inlets, anti-icing and air conditioning date.

Power Plant

1. The F-109 must have the J85 Phase II engine or equivalent to be successful. We are told by G.E. that the Phase II development and possibly Phase I are contingent on the F-109 project being funded. The present implication is that the F-109 project must bear the entire J85 Phase II development cost. The approximate cost of the eight engines after about the first 400 units (excluding initial development) will be of the order of $400,000.

2. From a technical standpoint, the J85 Phase II performance and weight would seem to be obtainable. The time allowed for the development, and consequently the cost are in our opinion the most probable pitfalls in regard to the engine itself. It is to be noted that G.E. is at present, two growth phases away from the final product. It is also our opinion that there is no further growth potential beyond the Phase II engine.

3. We believe that the JT12 has the potential to meet the F-109 requirements. Pratt & Whitney has stated that engines with thrust ratings approaching (but still less than those required) could be available in three to five years from go-ahead.

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Weights

1. Weights appear optimistic by approximately 1,084 lbs. Inadequate areas are:

Fuselage	450 Lbs.
Wing & Tail	90 Lbs.
Propulsion Group	110 Lbs.
Nacelle Group	60 Lbs.
Air Conditioning & Anti-Ice	138 Lbs.
Systems	127 Lbs.
Miscellaneous	107 Lbs.

No contingency has been allowed.

2. No allowance has been made in the above to provide acoustic protection, as discussed in our Dynamics Group comments (approximately 1,400 lbs.)

3. All design approaches show a difference over Convair's design concept in that durability is not provided.

4. To obtain the optimum weights shown by Bell it will be required to optimize design approaches and obtain a degree of weight control which may lengthen the proposed release schedule.

Structures

1. Structural criteria and structural arrangement appear to be satisfactory.

2. It is believed that with immediate go-ahead, structural release could be started in approximately two months.

Tooling

1. The XF-109 and F-109 program will be well suited for our facility. The materials and construction of this aircraft are well within our current capabilities both in know-how and facilities; and it is felt that no formal manufacturing development programs would be required prior to production. Further, a review of the procedures of Bell's production departments reveals that they are very similar to Convair's, and would be compatible to our procedures should a joint program develop.

General

A review of the above summaries indicates three specific areas of concern, i.e., hot gas ingestion, acoustic structural protection, and aerodynamic performance, stability and control. It is believed that further discussions with Bell are necessary and may cast light on possible solutions to these areas.

The effect of increased wieght is also of concern, however, it by itself results in a loss of radius which may not be too detrimental to the mission. Bell has done an extensive amount of development work which cannot be discounted and mechanically the airplane will fly.

It is noted that Convair believes that approximately 3 to 6 months are required prior to full release. Release of some areas could start immediately.

Weapon system and operation support studies must be expedited. One item in addition to the above cannot be realistically evaluated - this is the use and support of eight engines.

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Systems Analysis Comments on Potential Tactical Employment of the XF-109

The following comments concerning the XF-109's potential tactical employment are derived from the XF-109 brochure, discussions with Bell Aircraft representatives, and feelings based upon no study of the TAC mission area. Only limited information concerning the expected operational capability of the electronic or armament sub-systems as proposed by Bell is available at this time.

The TAC mission in recent years has been really the SAC mission from forward bases. A tactical limited war capability and the requirements of such have received considerable summer study attention by high level AF, Navy and Army planning groups this year. It is not clear as yet that the XF-109 weapon system is a desirable one for this mission. The Army proposes several field-type surface-to-air missile systems for protection against tactical air attack. Such missile systems may relieve the requirement for fighter support in forward areas. In addition, the possession of such missiles by the enemy provides an equivalent defense against such all-weather attack aircraft as the subsonic Navy A2F and possibly the XF-109 as well. Other Army proposals include missile systems to accomplish the tactical bombardment mission. Such developments may be expected to reduce military interest in more conventional methods of tactical weapon delivery.

If a manned fighter/bomber is considered a requirement, the XF-109 appears to fulfill the technical requirements in broad terms. That is, there appear to be no gross technical deficiencies. Further, the dispersed basing concept which is made possible through the VTOL capability, should contribute significantly to its survival probability. However, dispersed basing raises numerous questions concerning operational support, maintenance and logistics particularly of an eight-engined aircraft. Such questions indicate the need for a more detailed analysis of the TAC mission together with the XF-109 and its employment concept.

The unique aspects of this aircraft (M = 2.5/VTOL combination) may suggest other missions which would warrant further investigation. For instance, the aircraft might be used as an ASW interceptor to investigate SOSUS contacts about 300 miles off-shore. The hovering capability used in ASW search combined with very high dash speed to minimize "time late", when supported by a miniaturized ASW localization, classification and kill payload could make a system far more desirable than a P3V or future follow-on.

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Within the Air Defense Command mission there exists an attractive possibility that the XF-109 might be uniquely suited to policing the DEW line. The VTOL feature would constitute a significant improvement over zero-length-launching of conventional aircraft and at the same time would simplify the recovery problem. In this role, the intercept capability of the XF-109 should be adequate and its speed/altitude performance should be well suited for visual identification missions. As a trailer aircraft the XF-109, as presently configured, appears to have a relatively limited capability. However, there may be potentialities in this mission worthy of further developments.

It is hoped that the planned limited war studies for 1959 by Operations Analysis will provide a more concrete basis for appraising military systems of this type.

PRODUCT SUPPORT COMMENTS

1. Discussion centered around the Cooperational Ground System and Support System of the weapon system. Very little operational planning has been done to date by Bell Aircraft Corporation on the Cooperational and Support Systems.

2. Bell is ready at this time to submit to the Air Force the preliminary operational plan of the XF-109. F. Mambretti, Chief Weapon System Engineering at Bell was somewhat in doubt as to what should be in this plan and did not know of AFR-5-47 which outlines what must be in the plan, i.e., cooperational system plan, ground support system plan, and maintenance analysis plan.

3. Although Mr. Mambretti was familiar with MIL-D-9412C, he was amazed at the magnitude of its coverage throughout the weapon system program. For MIL-D-9412C, the following tasks are necessary in a Weapon System Management program:

   a. Write preliminary operational plan for USAF.

   b. Prepare Ground Support System Plan (Based on 1).

   c. Prepare Operational Ground Support Equipment System Specification (based on 1 & 2).

   d. Prepare Maintenance Analysis Specification (based on 1, 2 & 3).

   e. Prepare Ground Support Equipment Recommendations data per MIL-D-9412C, Figure 1 (based on 3 & 4).

   f. Prepare Ground Support Equipment Development Status Reports.

   g. Prepare Ground Support Equipment Test Schedules.

   h. Prepare Ground Support Equipment Qualification Test Reports.

   i. Prepare Support System Suitability Reports.

   j. Prepare Ground Support Equipment Procurement Data.

   k. Prepare Ground Support Equipment Illustrations.

l. Inspect all data, etc., (2 through 11) for quality.

m. Coordination and scheduling of all items (1 through 12).

4. Dispersed Basing: Generally the dispersed base aircraft must have everything in the way of operational support that a consolidated base aircraft must have, but all the support must be mobile. Mambretti agreed on this but maintained that this could be overcome by designing self-sufficiency into the aircraft. With the exception of a few "ease of maintenance" items on the XF-109, the only self-sufficient items seemed to be the integral pilot's ladder and the engine self starter. (To be realistic, these are the items that will be the first to be omitted in a weight saving program, and the XF-109 looks like it is due for some.)

5. Flight Engine Aircraft: (Maintenance and Logistics): - "All engines (J-85) have to be operating under full control and balance"per Mambretti. The control and balance will present a field checkout problem on a field engine change. All eight engines should be interchangeable to relieve the logistics problem. This can be done by multiple Q.E.C. interchangeability kits.

6. Recommendations:

a. That Convair do the cooperational system, Ground Support System, Maintenance Analysis and Logistic Planning for the weapon system.

b. Assuming that Convair joins Bell in this Weapon System, that the preliminary Operations concept not be released to the A.F. until Convair concurs, per AFR-5-47.

RESULTS OF PRELIMINARY INVESTIGATION OF ELECTRONICS FOR THE XF-109

1. Preliminary investigation of the XF-109 Bell Model 2000 Electronic Equipment indicates the system will adequately perform the required missions. The report on Electronic Equipment and Systems, which is on file in Preliminary Design, completely defines the electronic system for the Bell Model 2000 electronics. It outlines the major subsystems of the CNI group (Communications, Navigation and Identification), the Doppler-Inertial Navigation equipment, the central computer, the Fire Control Radar and the Displays and Controls. These units are identified by part number where standard equipment is used and by manufacturers noting the type equipment from which the unit was designed. Reference (2) further breaks down the units into individual boxes and shows size, weight and current status. Reference (3) details cooling required for all units and notes that approximately 8 pounds of cooling air per minute at 0° C of the 40 pounds available is needed to cool the electronic packages. Reference (4) shows pictures of each of the control panels and the display system.

2. Over-all Comments:

   A. The electronic system utilizes a central digital computer which eliminates the bombing computer, the navigation computer, the air data computer, etc. This is the primary reason for a system weighing between 900 to 925 pounds which has all the required capabilities as follows:

      (1) Intercept missions using radar and/or Ground Controlled Intercept and firing A-A missiles.

      (2) Attack missions to drop any type of weapon or to fire ASM.

      (3) Reconnaissance mission utilizing a bomb bay package side-looker radar and cameras.

   B. Production Program - No production program exists at this time. The following was set up for flight testing requirements and system testing. Delivery of the units as follows:

      1 Developmental Unit - 1 July 1960
      1 Developmental Unit - 1 September 1960

The units should be ground tested for 3 months and then flown in an RB-66 or equivalent type aircraft.

1 Prototype CNI, Computer and Doppler Interial Subsystem, 1 May 1961 to be flight tested in XF-109.

1 Prototype Computer and Fire Control Sub-system, 1 August 1961 to be flight tested in XF-109.

Production units for flight test aircraft at a rate of one per month, beginning in January 1962; two per month, in October 1962; and three per month thereafter, building up with aircraft requirements.

C. The General Electronic System Manager for Testing, Checkout and Flight Test has not been assigned. It is recommended that Convair be assigned as general manager for this task. This is considered the major problem area of the system.

D. Deficiencies in the Electronics System:

(1) No method of recording radar scope information. This should be added to the system.

(2) The autopilot and fire control radar should be interconnected so that the plane can fly hands off on an intercept mission. Some changes are required for this operation.

(3) The controls should be further tested to make sure they are not too close together for pilot use. Since the cockpit instrument panel and console are non-standard and therefore subject to customer controversy, a conventional cockpit arrangement should be drawn and mocked-up. The cockpit layout should be investigated further to ensure optimum "Human Factors" consideration.

(4) The two hydraulic motor driven 10 KVA generators are located in the Aft electronic compartment. This situation is objectionable because (1) close proximity of 400 cycle main power cables to electronic equipment will likely result in radio noise, (2) Hydraulic line breakage leaks in this compartment would create serious electronic operational problems and (3) the generators are not accessable for maintenance.

E. The Central Computer is the heart of the airplane and if this fails, the airplane has no attack or intercept capabilities and has few instruments operating on the panels. There is enough information available for the pilot to navigate by radar and land upon return to base. The computer should be highly reliable, however, and this is considered a practical system. The reliability of this unit should be completely investigated.

F. A Data Link is available to transmit and receive attack information between the plane and the ground. This could also be used for reporting attack information, etc.

G. A study is required to evaluate the best equipment for use in the aircraft as noted in Exhibit WCLN58-12 and -18. Equipment is listed as follows:

Exhibit WCLN58-12	Present Bell XF109
ARC-62 UHF Communications	ARC-51 (XN-3)
ARR-44 Data Link	ARC-51 (XN-3)
AIC 18 or 20 ICS	AIC-10
APN-94 Radar Altimeter	APN-116

Note: Other equipment is the same. Some of the above substitutions were made to save size and weight and others to provide newer equipment. This will have to be cleared with the Air Force and a final decision made as soon as possible.

H. Other items which should be investigated are as follows:

1. Study transient heating effects thoroughly.
2. Development of a mock-up program and cockpit simulator for electronic system tests.
3. Setup detailed flight test program.
4. Insure that the electronic system is completely integrated.

3. The Electronics System of the Model 2000 aircraft will adequately perform the required mission and it is highly recommended that Convair do everything practical to put this airplane and system into production.

References: (On file in Aircraft Preliminary Design)
(1) Electronics Equipment and Systems
(2) Status of Electronic Equipment
(3) Summary of Cooling Air Requirements
(4) Drawings 2000-748003, -748004, -748006, -748007 Consoles and Instrument Panels

F-109 EVALUATION - AERODYNAMICS

An evaluation of the data used by Bell Aircraft for performance estimation has been initated. The evaluation is based on the available published wind tunnel data and Bell analyses and therefore pertains to the Mach 2.3 version.

I. Mission Definition

The basic intercept mission has been selected for evaluation because it reflects the supersonic capability. This mission involves an acceleration to M_{max} with cruise out and combat at that speed. The definition of take-off allowance and landing reserve fuel quantities to be used for performance calculations radically affects the mission radius. The values assumed by Bell are felt to be optimistic although Mr. O'Malley, Chief of Aerodynamics and Propulsion at Bell, stated that the Bureau of Aeronautics has concurred with the Bell definitions. He further stated that Air Force concurrence could be inferred from the fact that WADC has monitored the progress of the XF-109 project. This matter could perhaps be resolved by a further study of the tactical deployment of the airplane and the associated operational techniques. The take-off allowance is inadequate to accelerate to climb speed, the landing reserve is inadequate to allow a go-around, and it has been assumed that armament is expended in combat. This latter assumption is contrary to MIL C-5011A (AF Perf. Spec.). However, lacking a prior agreement with the Air Force, this area of doubt represents a possible 22% reduction in mission radius as shown on the following table.

II. Drag Basis

A study of the drag basis used in the Bell performance estimates has shown that, with the exception of wave drag, the various components of the drag basis are consistent with Convair practice. However, the possible discrepancy in the wave drag is a serious one which must be resolved before the high speed performance estimates could be substantiated at WADC. Discussions with the Bell people who analyzed the wind tunnel data will be required to resolve this point.

A band of performance has been established, the width of which reflects the uncertainty in the wave drag estimate. This effect, along with other examples of optimism in the estimated performance, are tabulated below.

III. Performance

A. Radius Performance (Bell weight and thrust basis, rigid airplane)

		Relative Radius
1.	Bell Mission, Bell Drag Basis.	100%
2.	Bell Mission (SAGE reserves and MIL C-5011A Take-off Allowance)* Bell Drag Basis.	78
3.	Same as 2 but armament brought home.	76
4.	Same as 1 but including likely drag increase.	93

B. The doubtful areas in the drag basis represent significant effects on time to accelerate and supersonic ceiling. Subsonic performance appears to be on a firm basis and most fighter bomber missions are unaffected by the doubtful drag area.

* Take-off Allowance: 1 min. maximum power, 2 min. at 80% military power.

SAGE landing reserves: 10 min. loiter at 35,000 ft. and 10 min. at sea level.

IV. Stability and Control

An examination of the longitudinal stability data from wind tunnel tests of the current configuration indicates a serious stability problem at transonic Mach numbers. The attached curve (Figure 1) shows the elevator required for trim at $M = 0.9$ with the center of gravity at 33% $\bar{c}$. The curve shows that the rigid airplane would be slightly unstable at sea level and neutral at higher altitudes. Corresponding stick forces would be light or actually unstable. The effects of elastic deformation of the airframe have not been included, but are felt to be seriously destabilizing because of wing divergence. The airframe stability is obviously unacceptable in its present configuration. A number of fixes are possible: rebalance with the c.g. further forward with respect to the wing; resize and/or relocate the horizontal tail; accommodate the instability with the augmentation system and sell a deviation to MIL F-8785. Objections to each of these approaches can be raised and a solution in the problem would require further wind tunnel testing, as well as considerable analytic work and redesign.

With its quoted maximum deflection of 12°, the horizontal tail is also inadequate for longitudinal trim at Mach numbers above about 1.5. Figure 2 shows CL_{max} as limited by horizontal tail deflection (for the rigid airplane). The curve shows that trimmed level flight is not possible above about 60,000 feet at $M = 2.3$. The estimated ceiling at this speed is about 70,000 feet. Since a reasonable objective might be 2 to 3 g capability at ceiling, a considerable increase in control moments will be required.

Finally, the current configuration is directionally unstable at high Mach numbers. Mr. O'Malley believes that larger ventral surfaces, rather than a larger fin, will be required to achieve adequate stability.

The above stability and control problems do not appear to be insurmountable. However, their solutions could seriously affect the development schedule. Further, since these problems have been observed as a result of the most cursory investigation, the probability is clearly suggested that others may exist in areas not yet examined.

SUMMARY

The aerodynamic information generated since the preliminary evaluation has been, in general, adverse and contradictory to the verbal information obtained initially.

An attempt is made below to show the schedule on which Aerodynamics could proceed if Convair accepts engineering responsibility for the F-109 project. The schedule is paced by the availability of high speed wind tunnel data on which to base configuration modifications and performance estimates. Assuming immediate go-ahead, the schedule coincides roughly with the next high speed test in the Bell program.

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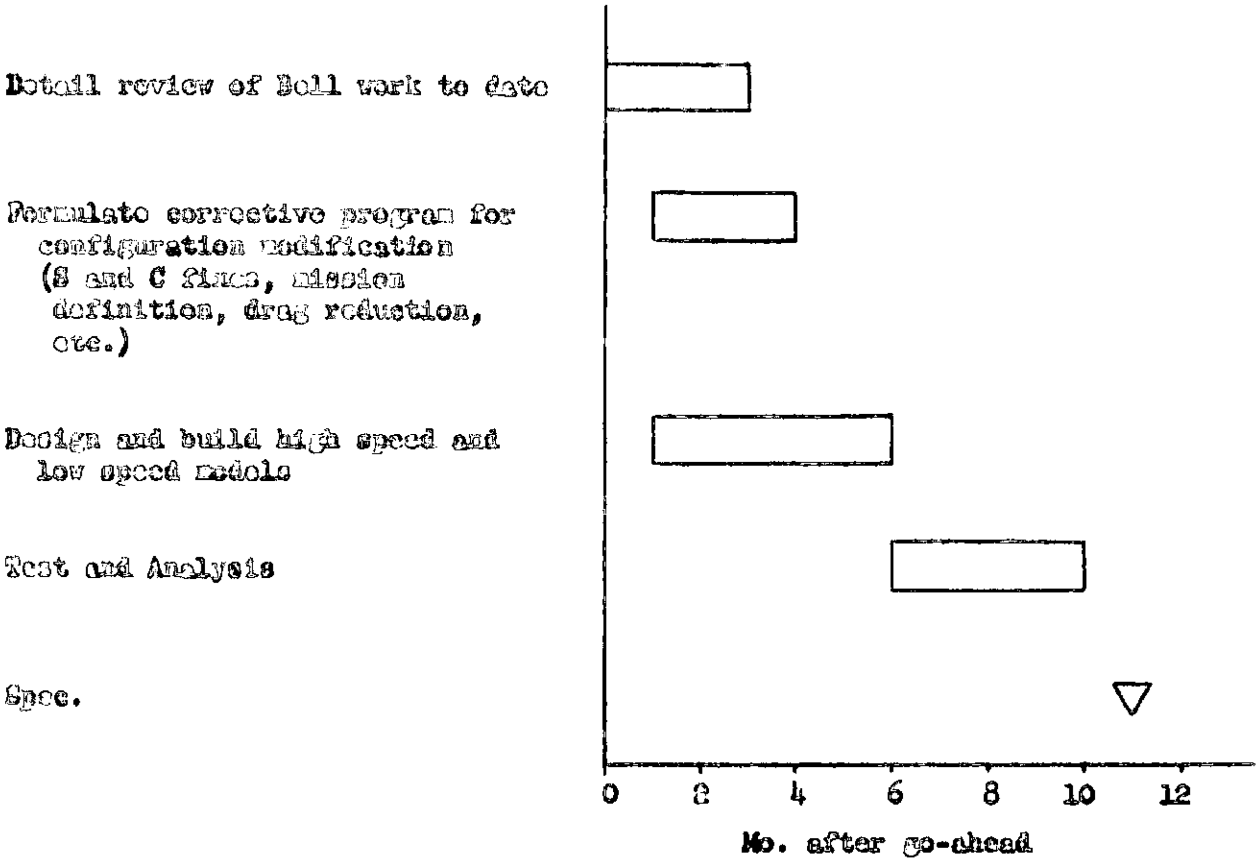

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BELL XF-107

ELEVATOR ANGLE REQUIRED
TO TRIM

M = .9

RIGID AIRPLANE

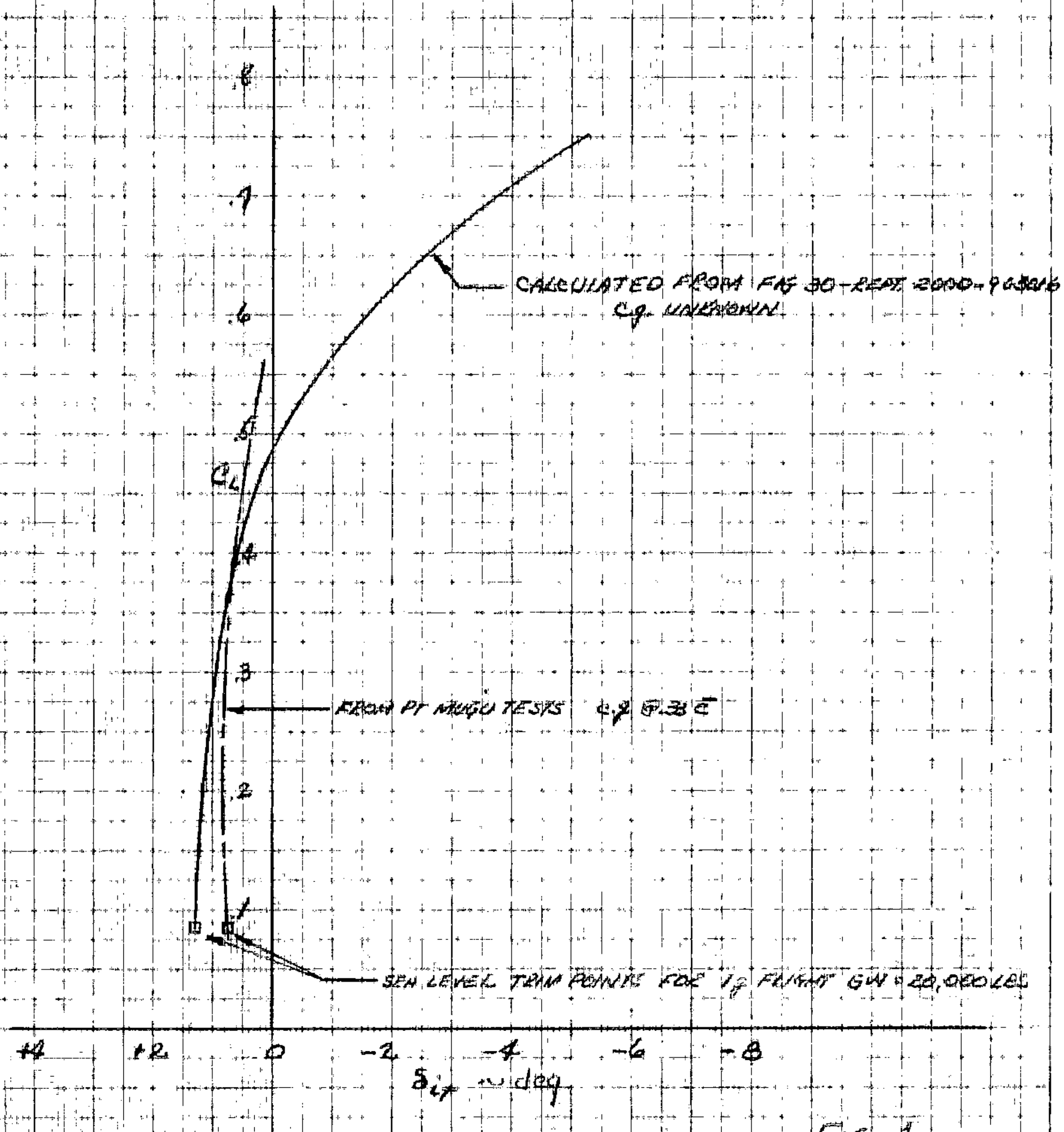


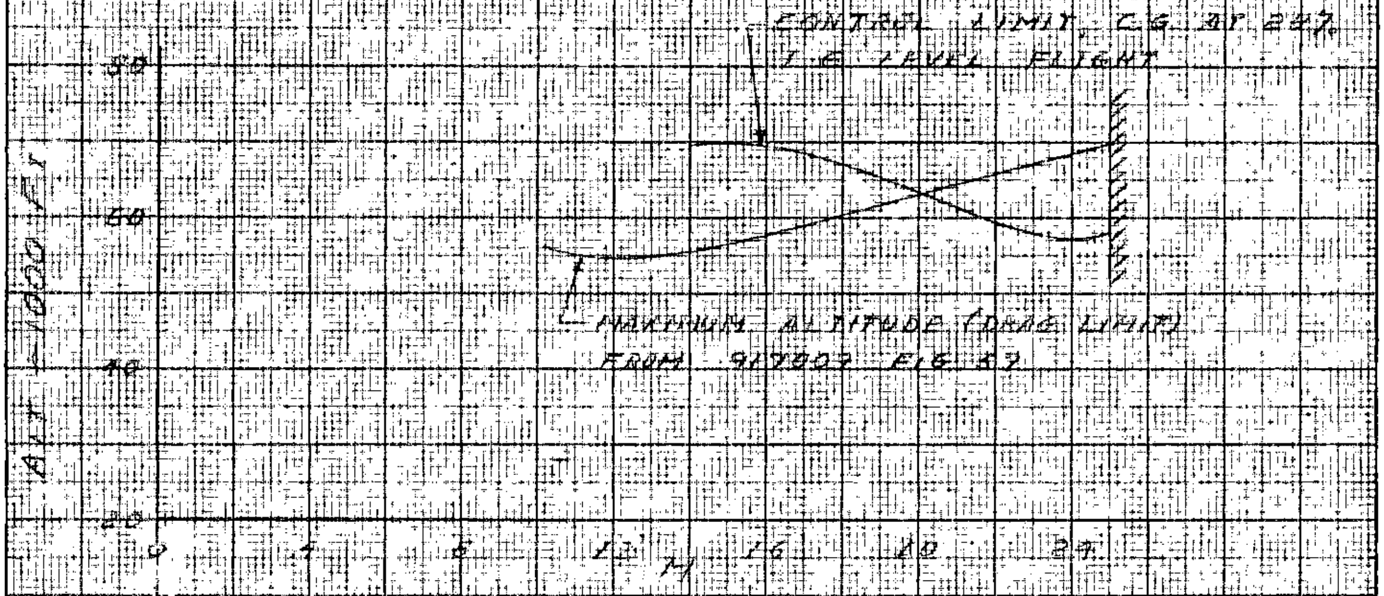
FIG. 1

10 X 10 TO THE CM. 359T-14G
KEUFFEL & ESSER CO.
ALBANY

XF-109

MAXIMUM ALTITUDE VS MACH NO

RIGID AIRPLANE



$C_{L_{MAX}}$ VS MACH NO

RIGID AIRPLANE

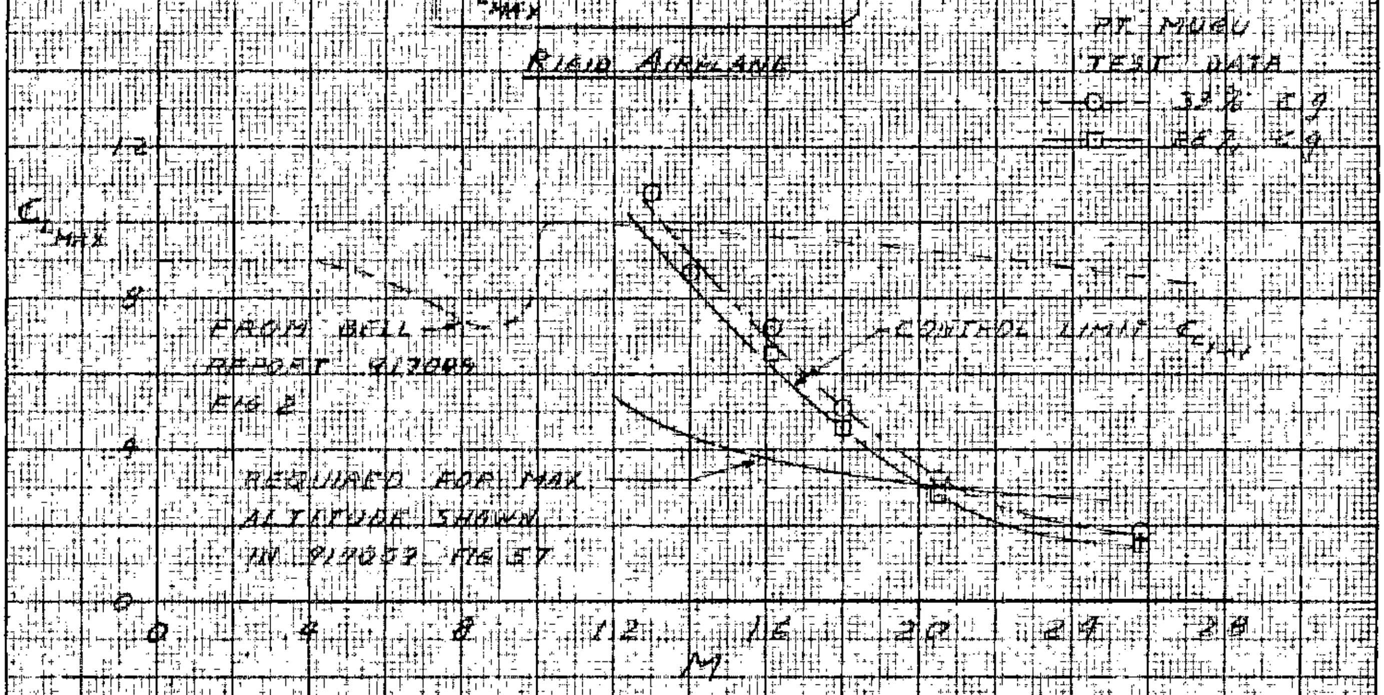


FIG. 2

DYNAMIC GROUP COMMENTS

The major problems in the Dynamic Group areas would appear to be in the acoustical excitation and the dynamic stability areas. The former problem represents a considerable weight addition to the weight estimates. The required data to evaluate the latter problem is not available at Convair.

A. Loads

It is anticipated that the major portions of the airplane structure will be designed from stiffness considerations. A possible exception to this could arise from consideration of the fatigue effects of low altitude turbulence on a wing which has a divergent characteristic. The general load picture is adequate, however, to make it feasible to consider putting the project on the design floor.

B. Flutter

Several potential flutter problem areas are immediately discernible upon examining the F-109 configuration. Bell is quite obviously aware of these areas and has conducted a series of Wind Tunnel tests designed to detect and correct design defects conducive to flutter instabilities. In general it can be said that Bell is considerably further along in their flutter program than is usual at the pre-design level.

C. Acoustics

The noise level as estimated by Bell in the region of the cockpit and impinging upon the aircraft structure have been briefly reviewed. The latest Bell data received is felt to be unrealistic in both the external environment and the transmission factors into the aircraft. The weight penalty involved in an adequate solution to these problems has been conservatively estimated at 1400 pounds. A reduction of this estimate might be possible by a careful consideration of the relatively short time requirements of this aircraft. Engine test stand data in this area employing at least a single engine is urgently needed to clarify the effects of ground reflection of the jet noise.

D. Armament Development

During the past week we have been verbally informed that all armament is to be either rocket fired or ejected from the trapeze. Ejection velocities of approximately 15 ft. per second are contemplated. This approach would seem to be realistic. Since the design of a trapeze system is a time consuming exercise its early start is recommended and an experimental verification program is urged.

E. Stability and Control

A major concern here is whether or not an airplane should be designed that is unstable within the flight regime', depending upon auto stabilization systems for satisfactory flight. This approach has previously been discussed but has not been adopted by Convair. The latest wind tunnel data available from Bell indicates unstable regions exist in both longitudinal and directional planes. We have been verbally informed that aeroelastic effects have been considered in their stability analyses but data indicating the magnitude of these effects have not been available to Convair. It can only be assumed from the basic aeroelastic response shown in the loads work that these effects would throw a large added responsibility into the autostabilization system. Moving the c.g. forward, to alleviate the stability problem will certainly accentuate the coupling problem between the rigid body short periods and the wing bending that must be present in the aeroelastic analysis.

Whether this problem is best solved by a rearranged empennage, a larger dependence on automatic control, or automatic c.g. control is a matter of considerable study.

The effect of engine duct instability on airplane response has not been studied. The stable range of the Nacelle ducts is considered marginal. With their out-board location this could lead to a very serious airplane response problem. The solution may lie in the autostabilization system design, the structural design, or in a design of the coupling mechanism between the paired engines. In any event, an approach must be selected.

F. Controls

The control system as outlined will apparently work. Simplifications, particularly pointed toward reliability is probably possible and desirable. Several months work in this area would benefit the aircraft.

Summary

In summary it would appear that the areas under question might effect the weight approximately 1400 lbs. and delay the firming of the configuration at least three months.

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THERMODYNAMICS COMMENTS

The previously submitted Thermodynamics comments regarding the XF-109 evaluation are revised and expanded as a result of discussions with personnel of Bell Aircraft Corporation during the week of 23 February. Most of the Thermodynamics discussions were with Mr. J.A. O'Malley, Chief of the Aerodynamic and Propulsion Design Department.

I. PROPULSION SYSTEM ANALYSIS

1. Static and Low Speed Performance

Preliminary scale model test data indicate that the estimated auxiliary inlet pressure recovery and the losses associated with the thrust diverter may be a little conservative. Analysis of the final scale model data and monitoring of the continuing test programs are required to verify these results.

The reaction control system for VTOL uses engine compressor bleed that is ducted to variable orifice nozzles at the airplane extremities for yaw and roll control. Thrust modulation is used for pitch control. Since the system is a demand bleed system, any control motions result in a thrust decrement. Full application of the roll control causes a thrust decrement of 1328 lbs., full yaw control, 972 lbs., and full pitch control 2200 lbs.

Bell Aircraft has done considerable computer and simulator study of the reaction control required during VTOL and transition. This required control produces thrust losses which are less than those used in the installed thrust analysis. An area which requires watching is the dynamic response of the engines to the varying bleed and thrust modulation demands required for control.

The installation losses arrived at in references 1 and 2 were used to calculate approximate thrust to weight ratios utilizing the advanced engine of reference 5 together with the latest gross weight of 25,963 lbs. These ratios are presented in figure 1 as a function of ambient temperature for the Phase II and phase I growth engines. It can be seen that the Phase II engines are a necessity.

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The re-ingestion of exhaust gases seems to be the most serious take-off problem. Bell has some question about the validity of the quantitive date obtained from the 1/10 scale jet impingment model. However, since this is the only data available, it is used. Their most recent work shows that the use of H beam baffles attached to the ground mat results in temperature rises of only 2 to 3° F. However, this is based on the optimistic assumption that take-off will be accomplished in 2 seconds after afterburner ignition, thus preventing full temperature rise. We believe that a take-off time of 15 to 20 seconds would be more realistic. Testing of the 1/10 scale model is continuing and a full scale test stand is scheduled to begin operation this year.

It is our feeling that take-off can be accomplished satisfactorily by using some amount of ground equipment - the exact amount and type to be determined by the model and flight test programs. Landing could probably be accomplished on any hard surface.

2. In-Flight Performance

In our opinion the supersonic performance is optimistically presented. Preliminary data indicate that the inlets as presently conceived are not satisfactory at high supersonic speeds. Data from wind tunnel tests of a 1/6 scale single nacelle inlet model indicate that the nacelle inlets have no subcritical stability margins at Mach 2 and above. The fuselage inlets have not yet been evaluated at supersonic speeds in the presence of a forebody and wing. At present high speed wind tunnel test programs are beginning on a 1/4 scale complete nacelle inlet model and on a 1/8 scale fuselage inlet model. It is our opinion that the inlet stability should be stressed as much as the pressure recovery during these tests and that the tests should be extended until satisfactory inlet operation can be obtained at high speeds.

Because of the very close inlet sizing used, it may be necessary to have the engine manufacturer guarantee the airflows at the high supersonic speeds as well as the thrust and fuel flow.

Our experience with the F-106 has indicated that inlet "buzz" is a bigger operational problem than performance. On the XF-109 inlet buzz of one of the nacelle inlets and the associated compressor stall and loss of thrust could cause a large airplane yaw. Instability in one nacelle inlet might also trigger instability in the other inlet of the same nacelle and thus aggravate the airplane control problem.

It is recommended that the inlet control development program be given more emphasis. Inlet control parameters should be studied during the previously mentioned wind tunnel tests.

As of yet, Bell has not done any detailed analysis of the secondary airflow required for the engine ejector nozzles. They plan to bleed air through a ramp slot for inlet boundary layer control and then to use this air as ejector secondary air. It is our opinion that the pressure level of this air will be too low for this purpose - expecially in the subsonic and low supersonic speed range.

II. AIRPLANE SYSTEMS

Insufficient time was available to discuss the other airplane systems with the Bell personnel. However, the following comments are felt to be pertinent.

Extensive data was generated by Bell Aircraft for Aero Heating of the wing during two flight considerations. Steady state flight and acceleration to V_{max}, (transient).

Appreciable thermal gradients can be realized between skin and stiffeners. Up to 100°F, in the transient case evaluated.

Our studies on aircraft with similar flight capabilities have shown some critical structural design consideration have been associated with higher transients; dive toward limit line, or in rapid deceleration from the Max. Temp. condition. These maneuvers were not evaluated for the effects of Aero Heating.

The extent of conservation inherent in the weight allowances are not presently known, however, consideration must be given to weight increases due to effects mentioned above.

Cooling of the engine compartment, hydraulic system, and accessory components are discussed in a general manner negating the possibility of any comment. Further such items, while contributing to the weight are not considered of major importance at this stage of aircraft design but must be examined in detail after general configuration stabilization.

Air conditioning data and details of the air conditioning unit were not available.

No anti-icing has been considered. If anti-icing is required, it could affect the inlet design and possible other areas.

III. SCHEDULE

In general it is felt that about six months will be required to reach a point where the airplane configuration can be defined enough to proceed with some assurance on detail design and guarantee performance. From a Thermodynamic standpoint this time is necessary to revise the inlet design, wind tunnel models, and model test programs, and to re-estimate the performance.

It may be possible, after about one month of detailed study, to release the present inlet design for prototype airplanes. However, these designs may require severe high-speed flight restrictions.

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References

1. Bell Aircraft Corp. Report No. D188A-945-216 "Propulsion System Analysis" dated 15 November 1957.

2. Bell Aircraft Corp. Report No. 2000-902046 "Propulsion System Analysis Revision No. 1" dated 30 October 1958.

3. Bell Aircraft Corp. Report No. 2000-902021 "Supersonic Inlet Model Test 147. Interim Data Report Nacelle Inlet." dated 8 August 1958.

4. Bell Aircraft Corp. Report No. 2000-902027 "Jet Impingement Tests Interim Report, October 1958" dated 15 December 1958.

5. General Electric "J85 Growth Engine. Preliminary Performance and Installation Manual". February 1959.

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CONVAIR PRIVATE DATA

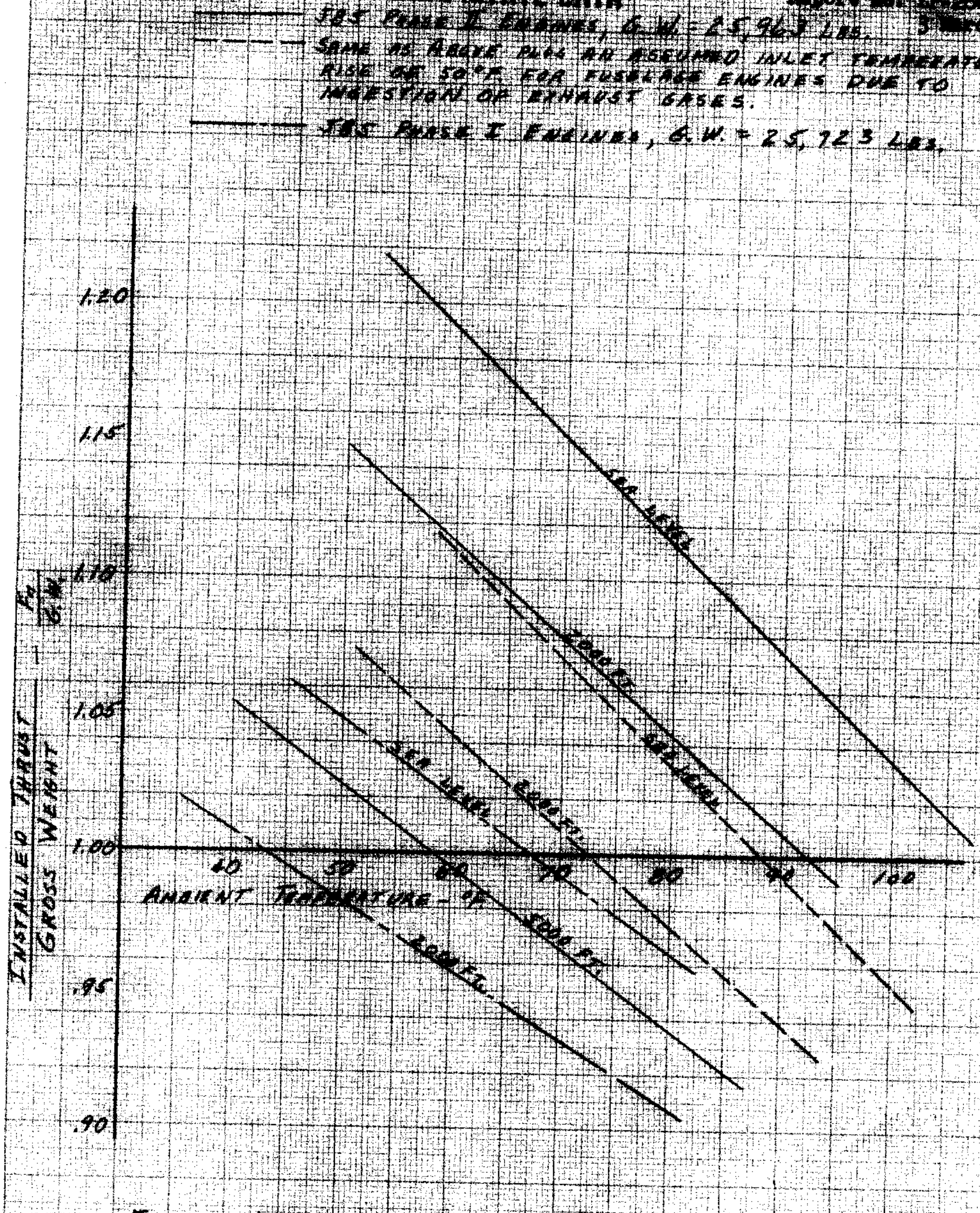

FIGURE 1: XF-109 Take-off Thrust to Weight Ratio

2-19-59 JEL

J85 ENGINE COMMENTS

The following comments on the J85 engine program are based on a very superficial and brief investigation.

The J85 program to date has had several rough periods, with G.E. having to redesign the compressor and other components to meet performance. This resulted in schedule delays and undoubtly higher costs. These problems appear to have also triggered off rather extensive organizational changes at Lynn.

The F-109 must have the J85 Phase II engine or equivalent to be successful. We are told by G.E. that the Phase II development and possibly Phase I are contingent on the F-109 project being funded. The present implication is that the F-109 project must bear the entire J85 Phase II development cost. The approximate cost of the eight engines after about the first 400 units (excluding initial development) will be of the order of $400,000.

From a technical standpoint, the J85 Phase II performance and weight would seem to be obtainable. The time allowed for the development, and consequently the cost are in our opinion the most probable pitfalls in regard to the engine itself. It is to be noted that G.E. is at present, two growth phases away from the final product. It is also our opinion that there is no further growth potential beyond the Phase II engine.

We believe that the JT12 has the potential to meet the F-109 requirements. Pratt & Whitney indicated that engines with thrust ratings approaching (but still less than) those required could be available in three to five years from go-ahead.

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GENERAL ELECTRIC J85

Version	Alt.	Temp.	F_n	Wt.	Wt/F_n	Wa	SFC	F_n/W_a	W_a/A_i	50 Hr.	150 Hr.
J85-5(AB)	0	59°F	3850	525	.136					July 59	April 60
J85 Phase I(dry)	0	59°F	2800	400	.1429					Jan 61	Sept 61
	2000	90°F	2505		.1597						
J85 Phase I(AB)	0	59°F	T.O. 4210	565	.1342	42.5	2.08	99.0	32.7	Mar 61	Jan 62
			Max 4125		.1370	42.5	2.13	97.0	32.7		
	2000	90°F	T.O. 3660		.1545						
	45000 (M_o = 2.0)	-69.7°F	Max 2730			27.8	2.18	98.2	21.4		
J85 Phase II(dry)	0	59°F	T.O. 3355	430	.1282	52.0	0.95	64.5	34.7	Apr 62	Jan 63
			Mil 3220		.1335	51.2	0.94	63.0	34.1		
	2000	90°F	T.O. 2800		.1535		0.97				
J85 Phase II(AB)	0	59°F	T.O. 4960	595	.1200	51.2	2.12	97.0	34.1	Nov 61	Nov 62
			Max 4870		.1220	51.2	2.17	95.1	34.1		
	2000	90°F	T.O. 4260		.1398		2.27				
	45000 (M_o = 2.0)	-69.7°F	Max 3100			36.1	2.30	86.0	24.1		

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PRATT & WHITNEY J60

Version	Alt.	Temp.	F_n	Wt.	Wt/Fn	Wa	SFC	F_n/W_a	W_a/A_1	50 Hr.	150 Hr.
J60-A(non AB)	0	59°F	2900	430	.1483					Oct 59	
J60-A(AB)	0	59°F	3900	645	.1654					Oct 60	
J60-B(non AB)	0	59°F	3000	430	.1433						Dec 59
J60-B(AB)	0	59°F	4025	645	.1602						Dec 60
J60-C(non AB)	0	59°F	3300	430	.130						Dec 61
J60-C(AB)	0	59°F	4400	645	.146	49.5	2.42	88.9	38.1		July 62
	45000 (M_o = 2.0)	-69.7°F	3020			37.3	2.41	81.0	28.6		
J60-D (non AB)	0	59°F	3675	450	.122					1962	Mid 63
J60-D (AB)	0	59°F	4400*	650	.146					July 62	Mid 63

* 5000# can be obtained thru development.

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WEIGHTS GROUP COMMENTS

Discussions with Bell Aircraft Corporation Engineering Staff regarding the weight of the F-109 indicates that they are optimistic about their ability to meet the weights they are presently quoting within some reasonable tolerance. This attitude is understandable since the airplane would be of questionable value if the weights increased by the amount indicated by Convairs weight group.

The largest single increase in weight is probably the one connected with engine noise effect on the pilot, electrical gear and airplane structure. It was agreed by Bell and Convair that this problem required additional detailed study. It was also agreed that Convair's brief weight evaluation was probably conservative and the 1400-1800 pound penalty could be reduced.

The second largest weight increase estimated by Convair was in the fuselage structure. This was reviewed in considerable detail with Bell Engineers. If it were possible to start with an F-106 fuselage and eliminate all unusual weight penalties in that structure and at the same time not add any weight increments for possible problem areas on the F-109 then it would be possible to meet the weight allowed for the F-109 fuselage. This is certainly an optimistic and possibly unrealistic viewpoint.

Another area of possible overweight indicated by Convair is in mechanisms, systems and wiring catagories. These areas are very difficult to evaluate. Convairs evaluation was based on experience on the F-102 and F-106. There have been advancement in the state-of-the-art on equipment design since the F-106 went into production however, it does not seem possible to build some of the complicated mechanisms for the weights allowed by Bell.

Another problem area which is closely related to the engine noise problem is in the light construction of wing and tail trailing edges. The .010" covering and 2.3 lb. density honeycomb is much lighter than Convair has ever considered using on their airplanes.

Convair has never been in a position where they could continue redesigning until the ultimate had been achieved.in light weight design. This will be necessary on the F-109; however, this is not compatable with a production program.

A weight summary follows with brief comments on the various weight changes.

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CONVAIR'S WEIGHT SUMMARY FOR M = 2.3 XF-109

	BELL M = 2.3	CHANGES	CONVAIR M = 2.3
Wing Group	1081	+ 50	1131
Tail	607	+ 40	647
Fuselage	2596	+ 450	3046
Landing Gear	648	+ 34	682
Surface Controls	525	+ 10	535
Nacelle Group	1286	+ 60	1346
Propulsion Group	(5641)	(+ 110)	(5751)
Engine	4470		4470
Air Induction	194	+ 30	224
Exhaust System	327	+ 40	367
Fuel System	490		490
Engine Controls	120		120
Starting System	40		40
Engine Cooling	0	+ 40	40
Instruments & Navig.	144	+ 23	167
Hyd. & Pneu.	565		565
Electrical	364	+ 76	440
Electronics	867	+ 30	897
Armament	0		0
Furnishings & Equip.	426	+ 63	489
Air Conditioning & Anti-icing	179	+ 138	317
Auxiliary	25		25
WEIGHT EMPTY	14954	+1084	16038
Useful Load	(11009)	(- 847)	(10162)
Pilot	215		215
Unusable Fuel	90		90
Oil	71		71
Usable Fuel	9750	- 911	8839
Armament	603		603
Prov. for Armament	280	+ 36	316
Survival Kit		+ 28	28
TAKE-OFF GROSS WEIGHT	25963	+ 237	26200

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GENERAL WEIGHT GROUP COMMENTS ON

WEIGHT INCREASES ON XF-109 - M = 2.3 AIRPLANE

WING GROUP

Basic check of wing box weight appears reasonable but trailing edge, spoilers, and deflectors appear low based on F-106 elevons of 3.9 lbs./sq. ft. + 50

TAIL GROUP

Based on F-106 and 880 data, the weights allowed for the horizontal tail leading edge and trailing edge and also the vertical trailing edge appear light. + 40

FUSELAGE

Based on data available and comparing with F-106 and F-104 data the fuselage weight appears very light. This is based on having a missile bay, numerous cut-outs for access, and forward engines compartment, and three fuselage fuel tanks. The door actuating and fuel tighting are also included in AN fuselage weight. + 450

LANDING GEAR GROUP

A weight evaluation by the design group and information received from CFT the main landing gear appears light. + 34

SURFACE CONTROL

The trim and feel control are short 10 lbs. based on F-106 control systems. + 10

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GENERAL WEIGHT GROUP COMMENTS ON

WEIGHT INCREASES ON XF-109 - M = 2.3 AIRPLANE
(Cont'd.)

NACELLE GROUP

Estimated increase in variable ramp controls and operating mechanism based on F-106 and F-102 data. + 30

It is questionable wiether sufficient weight has been allowed in the nacelle for the actuation, guides, tracks, aerodynamic seals, flexible connections etc. required for the movement of the inlet structure to provide an alternate engine inlet. + 30

EXHAUST SYSTEM

Bell data did not indicate expansion joints in diverter and tail pipe weights. This could amount to considerable weight. The expansion joints plus mounting rings are estimated to weigh an additional 40 lbs. + 40

AIR INDUCTION

Estimated increase in variable ramp controls and operating mechanism based on F-106 data. + 30

INSTRUMENTS

Weight allowed for instrument installation provisions appears inadequate. Bell data did not list instrument and installation separately. + 23

ENGINE COOLING

No weight allowed for engine compartment cooling and ventilating. It seems unlikely that proper airflow can be obtained through the entire operating range without some sort of distribution system, scoops, scrolls, ejectors, etc. For eight engines, isolated from each other, this could be considerable. Est. 5# per engine. + 40

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GENERAL WEIGHT GROUP COMMENTS ON

WEIGHT INCREASES ON XF-109 - M = 2.3 AIRPLANE

(Cont'd.)

ELECTRICAL GROUP

Estimated increase for wiring, relays, electrical equipment supports and lites. (Based on F-106)	+ 56
Electrical provisions for armament actuation	+ 20

ELECTRONICS GROUP

Based on comments of E.E.L. the weight for electronics appears satisfactory except for miscellaneous brackets and hardware.	+ 30

FURNISHINGS AND EQUIPMENT

Delete Survival Kit contents. Should be in Useful Load.	- 19
Convair feels that N.A. ejection seat will not do the job without modification. A weight panelty will be necessary.	+ 36
The glare shield appears light. Based on F-106, a weight increase is in order.	+ 5
Fire detection system on the F-106 is 7.5 lbs., if system is similar the XF-109 will need 60 lbs. vs. 19 lbs.	+ 41

AIR CONDITIONING AND ANTI-ICING

Based on studies of extending the F-106 to M = 2.3, it was estimated an increase in air conditioning unit and ducting would be necessary.	+ 40
Bell data indicates no anti-icing for air inlet ducts and ramps. The spec. implies there will be anti-iced if necessary. Convair feels it will be necessary.	+ 60
Based on F-106 data, the rain removal appears light.	+ 15
Anti-Ice should be 28 lbs. in lieu of 5 lbs. allowed based on F-106.	+ 23
TOTAL WEIGHT EMPTY CHANGE	1084

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GENERAL WEIGHT GROUP COMMENTS ON

WEIGHT INCREASES ON XF-109 - M = 2.3 AIRPLANE
(Cont'd.)

USEFUL LOAD CHANGE

Based on Falcon displacement data on F-106, it is estimated the weight for Armament provisions for 3 side-winders should be increased.	+ 36
Add global survival kit to Useful Load in lieu of Weight Empty.	+ 28
Reduce fuel and maintain maximum VTO Gross Weight of 26,200 lbs.	- 911
REVISE TAKE-OFF GROSS WEIGHT	26200

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XF-109

Additional Weights Required for the Acoustic Environment

Based on limited information available a check was made to determine an approximate weight penalty that would arise due to the Dynamic Group comments about acoustic treatment.

Basic Ground Rules

1. Weight per square foot enclosing cockpit area is to be 12 pounds.
2. Weight per square foot enclosing any electronic area is to be 6 pounds.
3. Weight per square foot throughout lower surface is to be 4 pounds.
4. Wing box structure where fuel is stored is assumed to be adequate at take-off with full power and at landing with reduced power.

Weights were derived by scaling a reduced drawing; therefore, are approximate. Column No. 1 represents weight increase to XF-109 using Bell's Estimated Weight and Balance Report 2000-942002 as a basis. Column No. 2 represents weight increase to XF-109 using weights as listed previously in this report.

	(1)	(2)
Forward Electronics @ 6#/Sq. ft.	+ 100	+ 85
Cockpit @ 12#/sq. ft.	+ 800	+ 570
Aft Electronics @ 6#/sq. ft.	+ 250	+ 220
Fuselage 238 to 652	+ 450	+ 350
Leading Edge	+ 20	+ 20
Trailing Edge	+ 80	+ 30
Horizontal Tail	+ 50	+ 25
Landing Gear Area	+ 100	+ 100
TOTAL WEIGHT PENALTY	+ 1850	+ 1400

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STRUCTURAL COMMENTS

The F-109 aircraft is designed for 6g load factors--hot--at the end of its thermal life. It is being designed to operate in a thermal environment for 3200 hours, which represents approximately 4000 flights of 43 minutes duration. The original intent was to design it for a Mach 2.5 (V_H = level flight high speed) and Mach 2.75 (V_L = limit dive speed). However, engine capability, it appears, will reduce the top speed to Mach 2.3 for V_H and Mach 2.5 for V_L.

The F-109 development was originally financed by the Bureau of Aeronautics; therefore, the structural design criteria is patterned after MIL-8629. However, the Navy ran out of funding and the Air Force took over and provided funds to complete Phase I study through mockup. It is expected, however, that the structural design criteria will not be changed.

The first aircraft to be designed and built will be limited to Mach 2 because of engine capability. However, the primary aircraft structure will be designed to the Mach 2.3 to 2.5 capability. The pilot's enclosure, the wing tip nacelle pods and engine ducts will be designed for Mach 2 and therefore will have to be redesigned when the more advanced engine capability is obtained.

Their present planning considers the use of X2020 and 2024-T86 aluminum alloy material. From tests which they have conducted on 2020 material they felt that in the temperature environment within which the aircraft will operate, the T86 is slightly better. However, approximately 75% of the primary structure will probably be designed by stiffness requirements and, therefore, X-2020 mater-

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ial becomes the better material due to its larger modulus of elasticity and also lighter unit weight. Very little is known about the formability of X-2020 material; however, it is believed to be comparable to 75ST or T-86. Availability of the material in sheet, extrusions or forgings should be determined to comply with manufacturing schedules. However, it is the writer's understanding that this is no problem. The A3J aircraft uses a considerable amount of this material in its manufacture.

The overall structural arrangement appears to lend itself to ease of manufacture. The wing arrangement is multispar, leading edge, trailing edge with spoiler speed brakes, no ailerons nor flaps and a few chordwise ribs. The structural box carries across the center line of the airplane. The fuselage appears to be very similar in construction detail to the F-102/106. They do not consider the use of structural doors and it is possible that a weight saving could be realized using structural doors at the expense of inspection and equipment removal time. The horizontal tail is an all movable surface with the actuator located forward. The vertical tail is an all movable surface with the actuator located aft. The wing tip pods which house two jet engine mounts to a pivot on the wing tip and is actuated by hydraulic cylinders for rotation from vertical to horizontal alignment. The nacelle is locked in the horizontal position. Material used in the nacelle is basically T-86 and X-2020 material, except in the aft section where titanium is used because of high engine heat. The nacelle is designed to be rotated up to a speed of 1.33 x V_s. The present critical position from their analysis is 60° off horizontal.

The main landing gear is located further aft of the C.G. than on present tricycle arrangements. This results in higher nose wheel loads. The gears are to be designed to 12' per second ultimate sink speed in the VTOL arrangement and 8' per second ultimate sink speed in the STOL or conventional landing arrangement. The estimated forward speed in a conventional landing is approximately 222 knots.

CONCLUSION

The structural criteria and structural arrangement appear to be satisfactory. The Structural Design Groups have been requested to evaluate the weights of their components and these weights will be included in the Weights Group comments. A comparison was made to F-104 aircraft weights and the structural weights of the F-109 are comparable. It is suggested in finalizing the detail specification that a statement be added allowing a ± 2% weight tolerance in the event weight empty has to be guaranteed. Because of the limited time and information with which to work, it is recommended that a more detailed weights study be undertaken to define the structural weights of wing, fuselage, landing gears, nacelles and tail structure. In order to do this, more information will have to be obtained from Bell Aircraft and it is the writer's understanding that a very complete loads report for the wing, body, tail, etc. will be available approximately March 15th. It is believed that with an immediate go-ahead, structural release could be started in approximately two months.

AIRCRAFT SYSTEM ARRANGEMENT COMMENTS

The XF-109 general arrangement of structure and systems for installations and maintenance is in most cases excellent. Areas of concern are:

1. Hydraulic driven fuel pumps are located in fuel tanks with associated hydraulic plumbing. (This is serious because of fuel contamination possibilities and difficulty in troubleshooting this type of system).

2. The armament configuration as related to extension system, bay doors design and general cavity area design should be thoroughly studied by Convair so that knowledge gained in F-106 program can be utilized.

3. The F-106 "B" ejection seat should be considered.

DEPT. 600 TOOLING COMMENTS

Production programming on the Bell XF-109 Prototypes is advanced to the stage that a reasonably sound evaluation of its potential as a production article could be made. Basic lines are released and master model work has commenced. A manufacturing breakdown has been negotiated and finalized by Tooling and Engineering. Preliminary planning is well underway and factory layouts have been made. Tool Design drawings of major assembly tools are complete to the degree that rough structure can be started.

A review of the materials slated for use on this airplane reveals nothing that should be a problem in manufacturing. The structure, for the most part, will be made up of 7075, 7178, or 2020 aluminum alloy. Although we have had no experience with 2020, a review of Alcoa literature indicates that it will form, machine, and join equally as well as 7075.

There is some titanium indicated for the Aft Fuselage Section and some stainless steel indicated for the engine pods. However, we have had successful experience with both of these materials.

There is some indication of a problem with regard to the necessity for extremely close alignment of the engine pods to insure the accuracy of the impingement of the center line of thrust of the four (4) pod engines. Further, there no doubt will be some problems due to the density of the airframe and to the number of engines with the associated systems involved. Sealing of fuel tanks may pose some problems.

In conclusion, it is our opinion that the XF-109 and F-109 program would be well suited for our facility. The materials and construction of this aircraft are well within our current capabilities both in know-how and facilities; and it is felt that no formal manufacturing development programs would be required prior to production. Further, a review of the procedures of Bell's production departments reveals that they are very similar to Convair's, and would be compatible to our procedures should a joint program develop.

FLIGHT TEST PROGRAM

The following Bell personnel met with Mr. Cheney to discuss Flight Test Program.

J. A. Cannon, Manager, Military Sales and Flight Test
A. W. Courtial, Chief of Flight Test
W. H. Ungerer, Chief of Flight Test Engineering
F. H. Flint, Flight Test Planning
R. May, Project Manager, Model 2000
F. J. Mambretti, Chief, Weapon System Section
D. C. Wright, Project Manager, Model 2000

The F-102, F-106, and proposed F-106C flight test programs were presented to give the Bell personnel our experience in accelerated development programs for the Air Force. Messrs. Cannon and May then presented their version of an accelerated flight test program for the F-109. This consisted of five prototype aircraft delivered about one per month followed by no aircraft for about nine months, then a production aircraft test program of five aircraft delivered three per month. During this discussion it developed that an immediate objective was to organize an accelerated development flight test program for use with an overall manufacturing schedule for placing the F-109 into tactical operation.

By pooling our knowledge a program was developed consisting of ten aircraft with limited engines and standard aircraft electronics, cockpit, and wiring, followed by twenty-three production version aircraft to confirm final design and provide nine aircraft for Category II testing. The proposed program is attached. This program is basically similar to the F-106A program aircraft assignments except for two additional for stability and control and four additional for electronics and fire control. Delivery dates of one per month were recommended from the second through the seventeenth aircraft. For the eleventh through seventeenth a modification period of from three month to one month is provided for incorporating final production configuration and systems. This follows the concept outlined in the flight test recommended for the F-106C Program. The date for delivery of fifteen aircraft per month for the F-109 is programmed for three years and one month following the initial flight of the first prototype. The comparative period on the F-106A is two years and seven months. The proposed F-109 program provides more lead time of Category I testing to production build-up than exists for our current F-106 Program.

This program was to be used by Bell personnel as part of a presentation at the Pentagon on Monday, 16 February, to support the production build-up possibilities for the F-109 to Wing Strength by 1964.

The above program was made with no consultation with the other Convair engineers. As soon as practical the details that they learned during their discussion should be coordinated with this proposed program to determine possible improvements. A brief description of the test stands planned by Bell was obtained. This indicated that their test stands were primarily aimed at the development of the prototype configuration and did not provide for similar tests of the tactical production systems. An entirely new static test stand program should be developed using Convair's experience from the Model 8. Two major pre-flight tests not included were the full scale tests of engine inlet characteristics and air conditioning system for cockpit and electronic cooling. The system tests currently planned by Bell are presented in the attached table.

The flight test of the X-13 and X-14 have been extremely meager. They primarily only confirmed that the concept of vertical takeoff with jets is feasible. Practically no instrumentation was used.

The existing flight test organization at Bell Aircraft is relatively small. Under Mr. Ungerer, Chief of Flight Test Engineering, are six Flight Test Engineers of the planning-operational-analysis type and four instrumentation engineers of operational type. In the Flight Test Operational Group there are seven pilots and four engineers of the flight engineer-crew equipment type. These personnel have background in tests of the X-1, X-2, and X-5, plus utilization during Rascal missile tests of the following aircraft: F-80, F-89, B-17, B-36, B-47, and B-50. For operations at Edwards on the "X" aircraft they have utilized groups of fifty to sixty personnel. For operation at Holloman on the Rascal missile they have used a group of 200 to 300 personnel. Some of their experienced flight test personnel are currently being utilized by other groups in the Bell Engineering Department.

Action which should occur to further develop a flight test program are:

1) Review of the proposed flight test program by planning personnel of Engineering Flight Test

2) Review of proposed flight test program by other engineering sections.

3) Development of a Convair proposed program for static and systems tests which should precede and be coordinated with the flight test program.

FT. P. 2

CONVAIR (SAN DIEGO)
A DIVISION OF GENERAL DYNAMICS CORPORATION
ENGINEERING FLIGHT TEST

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XF-109 & JF-109 USAF V
ACCELERATED DEVELOPMEN
(AS PROPOSED AT BELL-CONVAIR C

CONFIGURATION			AIRCRAFT	FLIGHT TEST PROGRAM ASSIGNMENT	FLIGHT TEST PROGRAM MONTHS	TEST AGENCY	TEST FACILIT
ENGINES ①	ARMAMENT	ELECTRONIC					
—	—	②	RB-66	DEVELOPMENT - ELECTRONIC & COMPUTER			
—	—	—	XF-109 ST	STATIC TEST			
YJ85-5		③	XF-109 S/N 1	VTOL REGIME TO ABOVE V_S & VTOL ENGINE OPER.			
J85-5		"	2	PRELIM. STABILITY & CONTROL (EVALUATION)			
"		"	3	PROPULSION / PHOTO SUPPORT			
"		"	4	LOADS - AIR LOAD DISTRIBUTION			
"		"	5	AIRCRAFT SYSTEMS			
"		④	6	CNI, DOPPLER & COMPUTER			
"		③	7	STABILITY DERIVATIVES / NASA			
"	X	③	8	ARMAMENT SEPARATION & BALLISTICS			
"		⑤	9	FIRE CONTROL			
"	X	③	10	ARMAMENT SEPARATION & BALLISTICS			
J85 PHASE I	X	COMPLETE SYSTEM	JF-109 S/N 11	SYSTEMS COMPLETE - FULLY EQUIPPED			
"			12	STABILITY & CONTROL			
"			13	PROPULSION (PERFORMANCE)			
"			14	LOADS - AERODYNAMIC HEAT DISTRIB. & STRUCTURAL DEMONSTRATION			
"			15	COMPUTER & DOPPLER - INERTIAL			
"			16	PHASE IV PERFORMANCE (CATEGORY II)			
"			17	CNI / INTERCEPT ENVIRONMENT			
"			18	FIRE CONTROL - AIR-TO-AIR (A-A)			
"			19	" " - AIR-TO-GROUND			
"	X		20	WEAPON DELIVERY - A-A / PRODUCTION SUPPORT		JOINT CATEGORY I	
"	X		21	" " - BOMB			
"	X		22	" " - AIR-TO-SURFACE MISSILE			
"	X		23	ENVIRONMENT			
J85 PHASE II			24	ENGINES			
"	X		25	CATEGORY II			
"	X		26				
"	X		27				
"	X		28				
"	X		29				
"	X		30				
"	X		31				
"	X		32				
"	X		33				

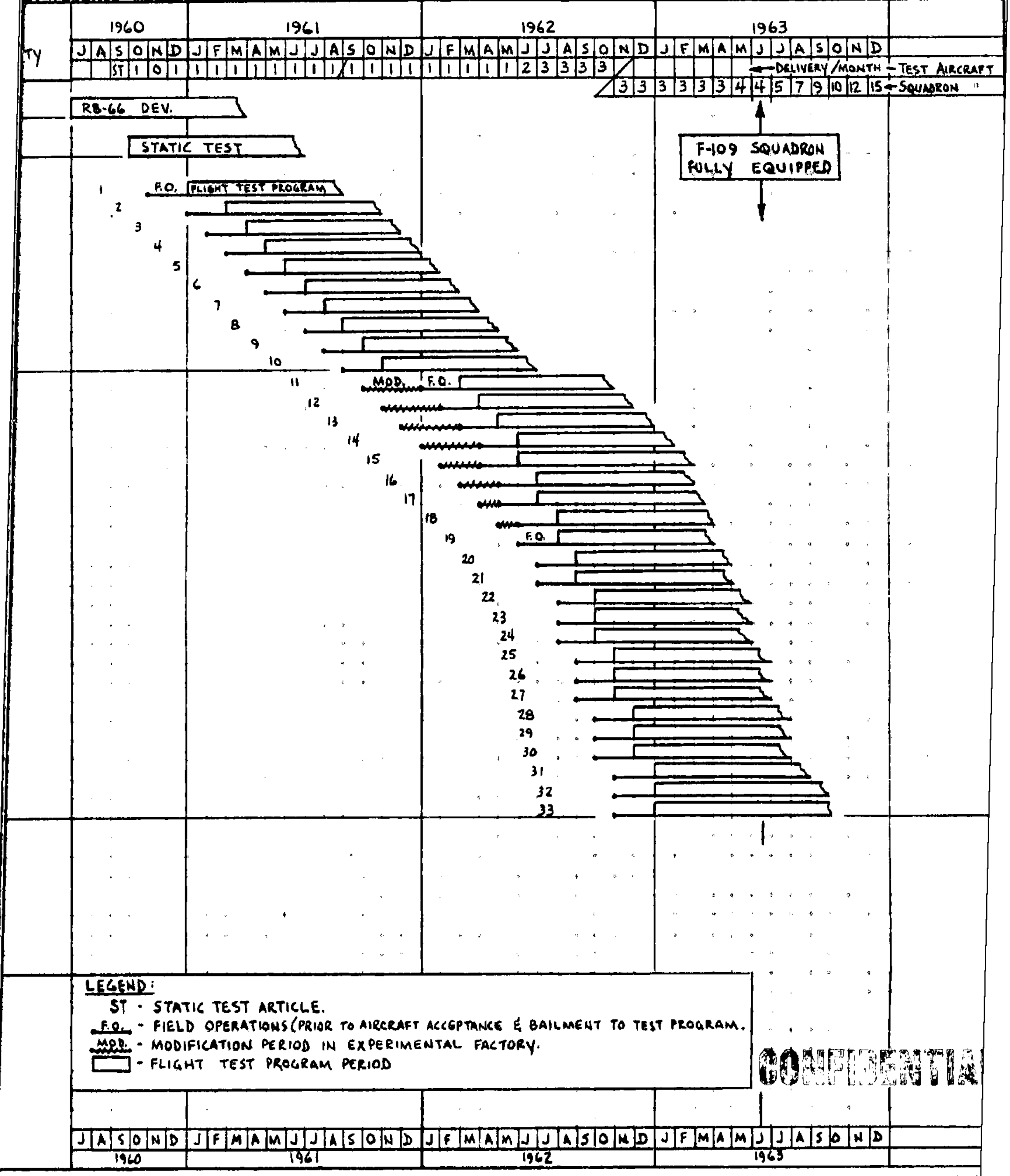
NOTES:

ENGINE CONFIGURATION:
① - ENGINE TYPE AVAILABILITY NOT CORRELATED WITH PROGRAM.
ARMAMENT CONFIGURATION:
X - ARMAMENT CONFIG. TO BE INSTALLED.
ELECTRONIC CONFIGURATION:
② - DEVELOPMENT - CNI, COMPUTER, & FIRE CONTROL.
③ - CONVENTIONAL AIRCRAFT.
④ - PROTOTYPE - CNI COMPUTER.
⑤ - " - FIRE CONTROL & COMPUTER.

TOL TACTICAL FIGHTER
NT FLIGHT TEST PROGRAM
CONFERENCE HELD ON 13 FEB. 1959)

REFERENCES:
BELL CHART No. S-117855
" " " S-117725-2 DTD 2/13/59

EFT-P-GEN-22
17 FEB. 1959



LEGEND:
ST - STATIC TEST ARTICLE.
F.O. - FIELD OPERATIONS (PRIOR TO AIRCRAFT ACCEPTANCE & BAILMENT TO TEST PROGRAM.
MOD. - MODIFICATION PERIOD IN EXPERIMENTAL FACTORY.
▭ - FLIGHT TEST PROGRAM PERIOD

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D-188A PHASE II TEST SCHEDULE
(Tentative)

		Category I Tests	Engine System	Fuel System	Flight Control	Hydraulics	Electrical
I.	ABM Release	-	5-1-59	5-1-59	5-1-59	5-1-59	5-1-59
II.	50 Per Cent Release	-	7-15-59	8-15-59	9-15-59	8-15-59	8-15-59
III.	100 Per Cent Release	-	9-1-59	10-1-59	11-1-59	10-1-59	10-1-59
IV.	Start Release - Test Procedures	-	12-1-59	2-1-59	12-1-59	12-1-59	1-1-60
V.	Finish Release - Test Procedures	-	9-1-60	9-1-60	7-1-60	5-1-60	9-1-60
VI.	Start Manufacturing Responsibility	2-1-59	7-1-59	8-1-59	4-1-59	6-1-59	11-1-59
VII.	Sub-System Testing	-	-	-	1-1-60	11-1-59	-
VIII.	Start Final Tests	-	2-1-60*	4-1-60	2-1-60	3-1-60	3-1-60
IX.	Complete Final Tests	3-1-60	11-1-60	11-1-60	11-1-60	8-1-60	11-1-60
X.	Standby Period	-	(11-1-60 (3-1-61	(11-1-60 (3-1-61	(11-1-60 (3-1-61	(8-1-60 (3-1-61	(11-1-60 (3-1-61

* NOTE: October 1, 1959, subject to engine availability.

F1. P4

CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION

SAN DIEGO

UNCLASSIFIED

REPORT ZP-8-040
DATE 30 April 1956
MODEL 8

[redacted]

TITLE

STUDY

ONE MB-1 ROCKET AND FOUR FALCON MISSILES IN THE F-102A AIRPLANE

ZP-8-040

PREPARED BY R. C. Lingley
R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert

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INTRODUCTION

This study shows the changes required to revise the basic F-102A armament from six Falcon missiles and (36) 2.00" rockets to four Falcon missiles and one MB-1 rocket.

The study is based upon a change that would be scheduled into the production line, however, it is entirely feasible to incorporate a similar design on a retrofit basis that could be accomplished in the re-cycling program.

Convair has installed an MB-1 ejection system in the center bay of an F-102A test airplane. Ground tests of this configuration have been completed using a simulated MB-1 weight to check out the ejection mechanism. This airplane is now at Holloman Air Force Base for flight evaluation. In these tests, inert MB-1 weapons will be used to check the predicted trajectory during ejection at various speeds and altitudes.* Firing of live MB-1 weapons with dummy warheads will follow.

Since the intent of these tests is to evaluate the ejection system for the MB-1 installation in the F-102A airplane, it will minimize the amount of additional testing required for the production airplanes.

* These tests are tentatively scheduled to start during the second week in May, 1956.

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SUMMARY

Conversion of the F-102A airplanes to carry one MB-1 rocket and four Falcon missiles can be accomplished with minor changes to the structure and revisions to the armament and related systems.

The existing four Falcon missiles and their displacing gear in the outboard missile bays will be unchanged.

Rockets and their provisions will be removed from the missile bay doors. The center missile bay doors will be made shorter and revised to accommodate the MB-1 rocket. The forward end of the center missile bay will be faired over creating a compartment for future equipment growth.

To minimize changes to the pneumatic and electrical systems, it is assumed that all doors will be opened during each armament cycle. (Further studies may prove the desirability of opening the bays individually to accommodate the particular armament selected. In this case, the outboard "flipper" doors must be sequenced to open simultaneously with the center bay doors in order to relieve pressure in the outboard bays. This would cause additional complications to the pneumatic and electrical systems.)

With the proposed armament load, the airplane gross weight will be 150 pounds less than the present tactical airplane. There will be a slight aft C.G. shift which will remain within acceptable limits.

The change in gross weight and the C.G. shift will have a negligible effect on performance.

[redacted]

DISCUSSION

The following changes must be made to provide for one MB-1 rocket in the F-102A airplane while retaining four Falcon missiles and removing provisions for (36) 2.00" rockets:

A. General (Ref. Dwg. SD-56-08036)

1. Replace (2) center Falcons with (1) MB-1 rocket.
2. Remove provisions for (36) 2.00" rockets.
3. Add shorter center bay doors.
4. Fair over forward end of center bay and provide access into this compartment.
5. Revise structure to provide for ejection loads.
6. Revise pneumatic, electronic and electrical systems to accommodate the above changes

B. Armament Group

1. Remove (2) Falcon missile displacing gear from center missile bay.
2. Remove provisions for rockets. (The existing outboard doors may be retained by removing rocket tubes and blast pans.)
3. Add ejection mechanism and provisions for (1) MB-1 rocket in the center missile bay.
4. Add shorter center bay doors to accommodate the MB-1 rocket.
5. Modify the missile bay beams to accommodate relocation. of center door actuators and removal of trunnion supports for the center missiles.

[redacted]

C. Fuselage Group

1. Revise structure to provide for ejection loads of the MB-1 rocket. The following frames are effected: Stations 310.25, 323.39, 329.62, 349.75, 370.75 and 377.75.

2. Add structure at forward end of center missile bay between stations 217 and 273. Add partition between missile bay beams at station 273. Provide access door into this compartment and provide for marker beacon antenna in this area.

D. Mechanism Group

1. Revise center bay door actuator installation to provide for shorter center bay doors.

E. Pneumatic Group

1. Remove two center missile displacement gear actuators and revise pneumatic system accordingly.

2. Revise pneumatic system to provide for revision of door actuators on shorter center bay door.

3. Relocate (4) air flasks to center bay compartment.

Note: Assume all six doors will operate simultaneously.

F. Electronics Group

1. Remove provisions for two center missiles and (36) 2.00" rockets.

2. Add provisions for (1) MB-1 rocket.

3. Add rocket umbilical plug for the MB-1.

4. Remove approximately (21) units and (1) rack from the MG-10 system

5. Add (7) units and (1) rack to the MG-10 system.

[redacted]

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6. Relocate marker beacon antenna from center bay door to forward end of bay.

G. Electrical Group

1. Delete rocket wiring and provisions.
2. Delete harnesses and provisions for two center Falcon missiles.
3. Provide harnesses and provisions for one MB-1 rocket.
4. Revise armament control panel in cockpit.
5. Revise miscellaneous harnesses, junction boxes, relay boxes, switches and circuit breakers to accommodate above changes.

H. Controls Group

1. Revise elevon control cable and relocate turnbuckles in center missile bay.

See the following pages for weight summary, C. G. envelope and a drawing of the proposed installation.

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WEIGHT SUMMARY

F-102A AIRPLANE WITH (1) MB-1 & (4) FALCONS

Wing Group	3,012
Tail Group	427
Body Group	3,520
Alighting Gear Group	1,045
Surface Controls Group	440
Engine Section	39
Propulsion Group	(6,441)
Engine Installation	5,015
Air Induction System	724
Exhaust System	133
Cooling System	17
Lubricating System	27
Fuel System	417
Engine Controls	21
Starting System	87
Fixed Equipment Group	(4,090)
Instruments and Navigational Equipment Group	143
Hydraulic and Pneumatic Group	319
Electrical Group	575
Electronics Group	2,165
Armament Group	403
Furnishings and Equipment Group	206
Air Conditioning and Anti-Icing Group	224
Auxiliary Gear Group	52
WEIGHT EMPTY	19,014
USEFUL LOAD	(8,720)
Pilot, Survival Kit and Parachute	248
Fuel - Unusable (JP-4) 23.7 Gallons	154
Fuel - Usable (JP-4) 1,050 Gallons	6,825
Oil - Trapped and engine	46
MB-1 Rocket (1)	791
Falcon Missile GAR 1-A and/or 1-C (4)	508
Missile Launchers (4)	148
NORMAL TAKE-OFF GROSS WEIGHT	27,734*

*Existing F-102A Gross Weight is 27,884 lbs. based on current status of weight and balance report No. 27 [illegible] dated 1 Feb. 1956.

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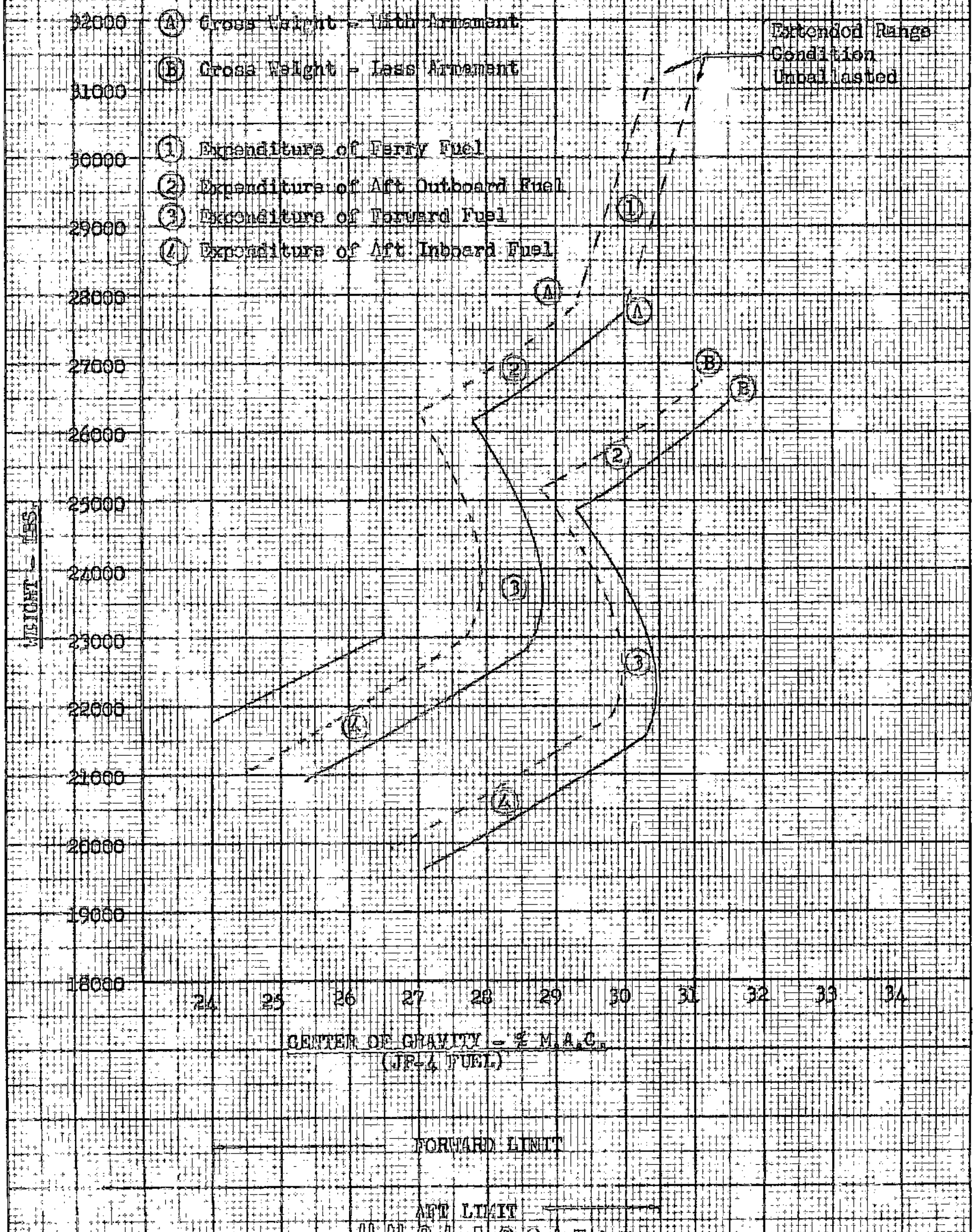
F-102A TACTICAL AIRPLANE WITH ONE (1) MB-1 ROCKET & FOUR (4) FALCON MISSILES

NOTE: Similar information for the F-102A Tactical Airplane (6 Falcon Missiles & 36 2.00 inch Rockets) is shown by the dashed line curves.



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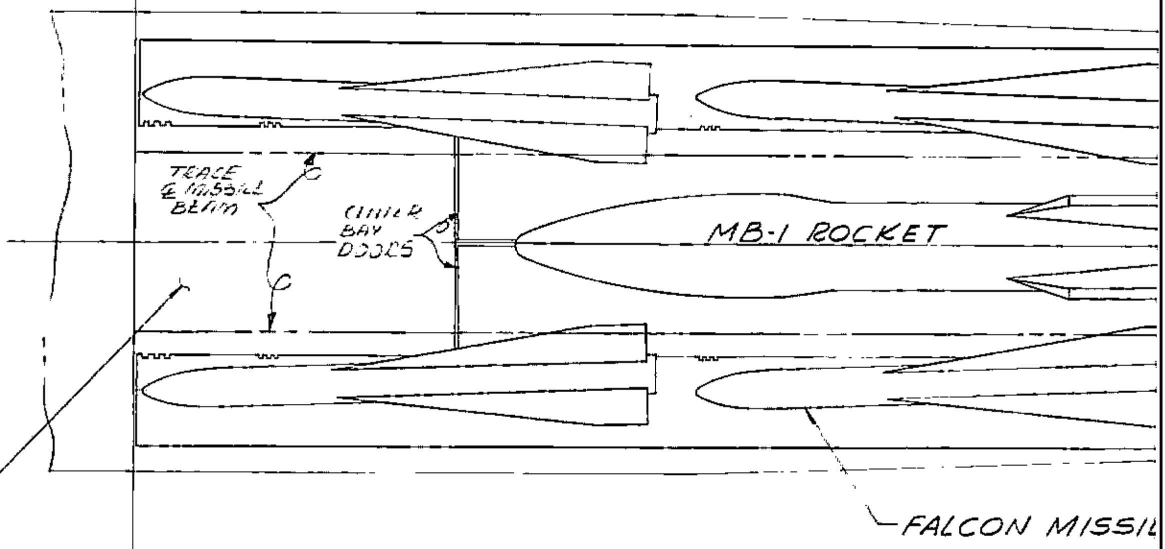
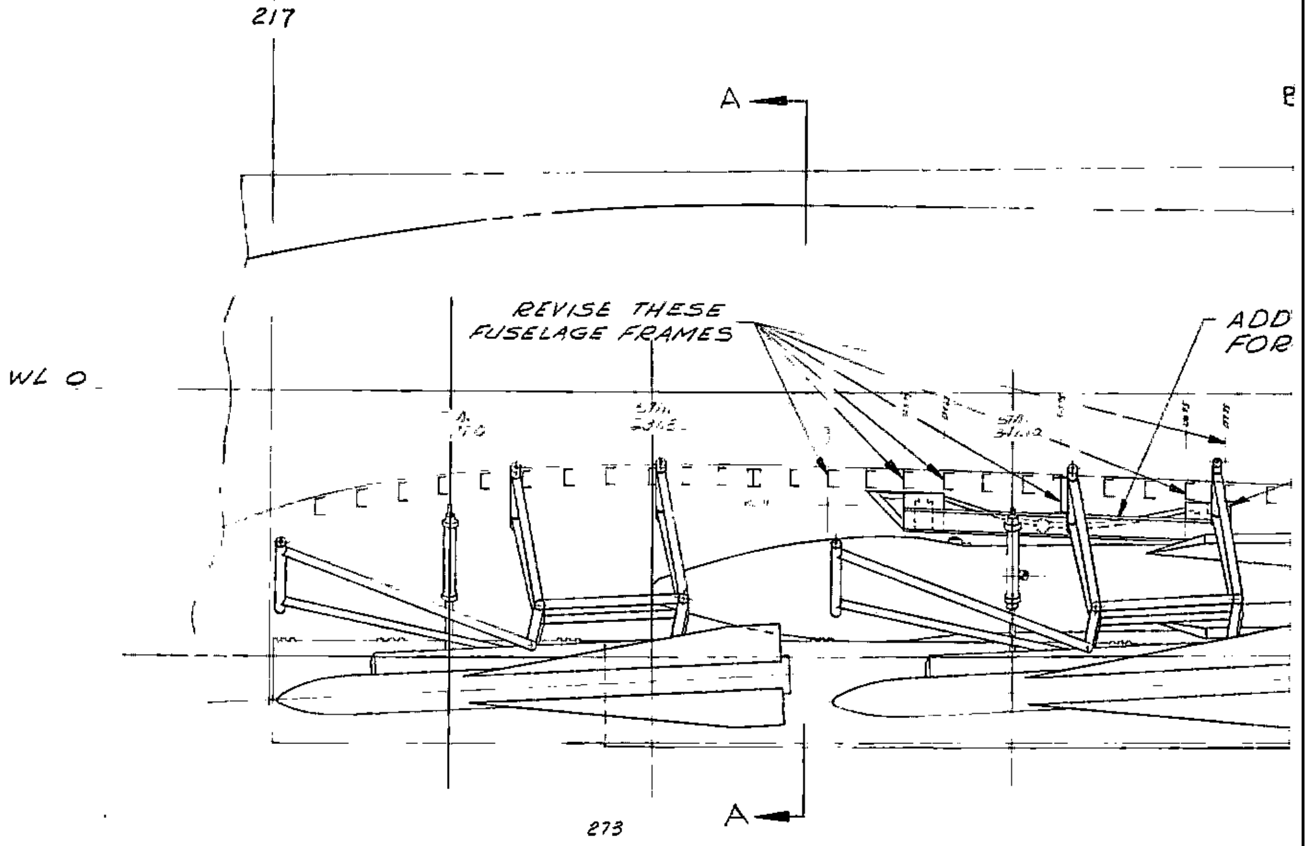


—FAIR OVER FORWARD END OF CENTER BAY. 217 TO 273. (PROVIDE ACCESS DOOR)

ADD PARTITION IN CENTER BAY AT STATION 273.

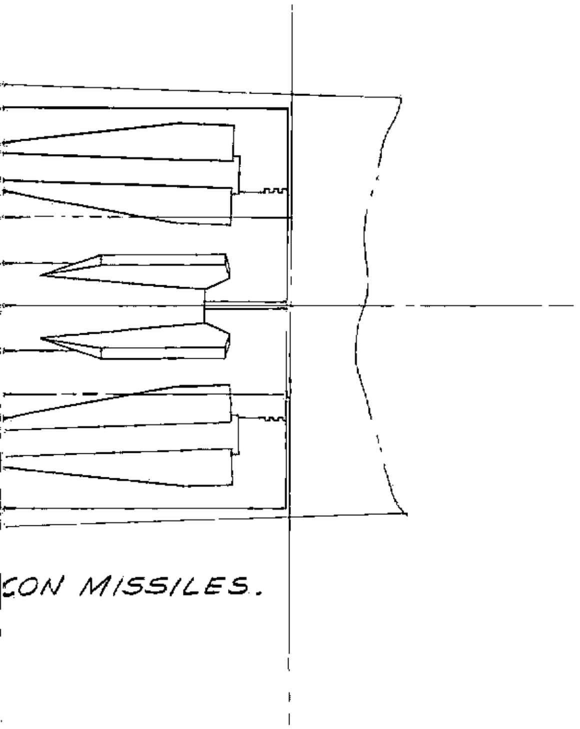
PLAN VIEW



SIDE VIEW

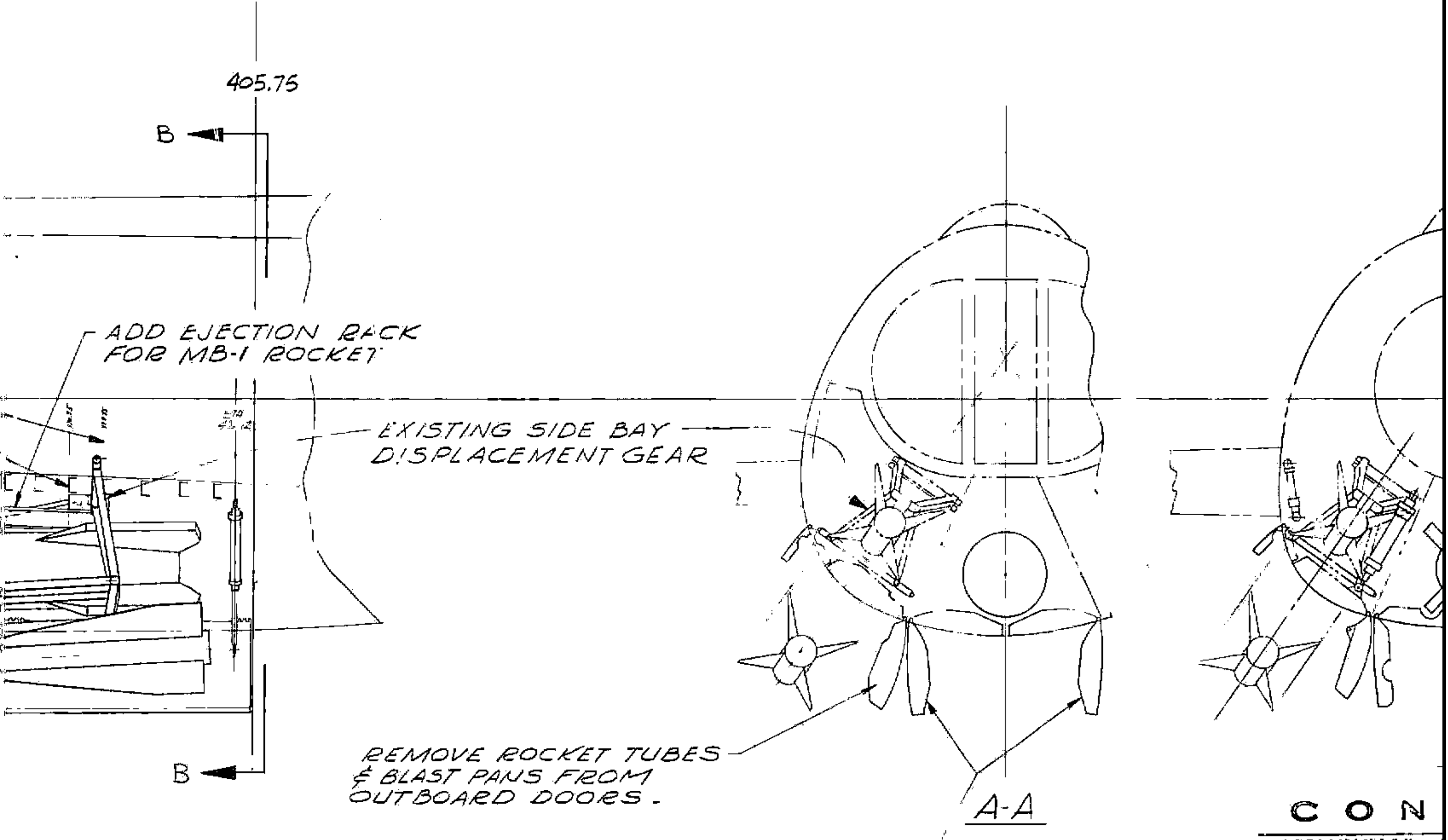
MISSILES IN FIRING POSITION

MB-1 ROCKET IN STOWED POSITION



CON MISSILES.

NOTE: REMOVE (2) FALCON MISSILES
& DISPLACEMENT GEAR FROM
CENTER MISSILE BAY.



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TION

DESIGN NEW CENTER BAY DOORS-
STA 273 TO 405.75.
MOVE FWD. DOOR ACTUATORS AFT.

REF. REPORT ZP-8-040.

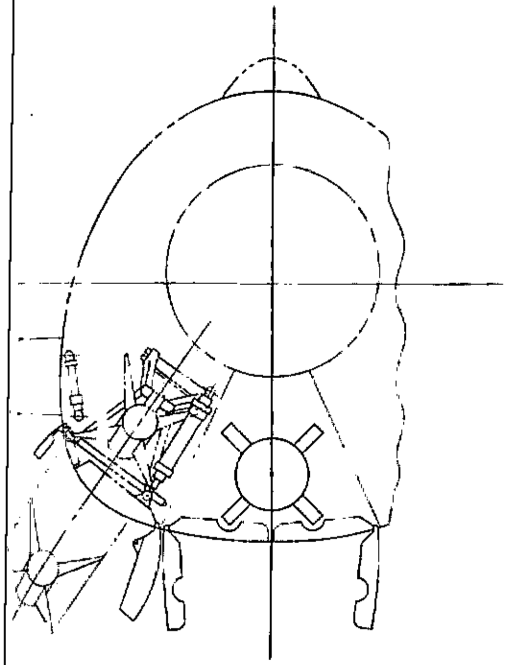
C O N

PRELIMINARY

STUDY-(1) M
(4) F
IN TH

E BUGGERIO DIETZ

ZP-8-040
page 8



B-B

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C O N V

PRELIMINARY

STUDY-(1) MB-1 ROCKET &
(4) FALCON MISSILES
IN THE F-102A AIRPLANE

DIETZ 1/10 26 APRIL 56

SD-56-08036

ZR-8-024

ZP-8-024

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

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REPORT ZP-8-024
DATE 26 July 1955
MODEL 8

TITLE

STATUS REPORT - PROPOSED F-102 C AND F-102D VERSIONS OF F-102A

ZP-8-024

26 JULY 1955

PREPARED BY H. E. Dickard
H. E. Dickard

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert

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Drawing No. SD-55-08068 8

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INTRODUCTION:

This report is a brief resume of the status of the F-102C and F-102D versions of the F-102A to date. A requirement for these versions is indicated in Reference (1).

The F-102C is identified with an armament load of (1) Ding Dong plus (2) Falcons and the F-102D is identified with (2) Ding Dongs and (40) 2" rockets.

The feasibility of, and a basis for further engineering effort for, the F-102C and F-102D projects depends largely upon the outcome of the firing tests scheduled for the dummy F-102A fuselage and test airplanes 1781 (fixed pylon Ding Dong mount), 1797 (Ding Dong extension launching system), and 1806 (Ding Dong ejection launching system).

SUMMARY:

The major findings noted to date are:

A. Fuselage ballast would be required with either the F-102C or F-102D.

B. The maximum take-off weight thus far studied (28,336 lb. for the F-102D) would not appreciably affect performance.

C. Revisions to the MG-10 electronic system for the Ding Dongs and I. R. Falcons, if used, would not change the volume requirements or weight of the airplane to any extent.

[redacted]

GENERAL:

The studies of SD-55-08057 and 08058 show a few suggested armament arrangements for the F-102C and D. The launching methods shown for the Ding Dong missile are basically the same as being incorporated in the experimental airplanes 1797 (extension) and 1806 (ejection). Study 2 of SD-55-08057 and Study 3 of SD-55-08058 would appear to constitute minimum changes to the present F-102A missile bay configuration. An alternate launching method which combines extension and ejection is shown on SD-55-08058.

The Falcon missiles may be launched by the displacing gear presently used on the F-102A or by a lighter ejection system now being considered. The ejection system would require ballast in addition to that shown below.

WEIGHTS:

Two preliminary weight studies of typical arrangements are tabulated below. Corresponding c. g. curves are shown on following pages.

	F-102C	F-102 D
Armament	(1) Ding Dong (2) Falcons	(2) Ding Dongs (40) 2" Rockets
Location	D. D., Center Bay Aft. -Falcons, Outb'd. Fwd.	D. D., Side Bay Aft. -Rockets, Short missile bay doors
T. O. Weight, Lbs.	27,332	28,336
Ballast, Lbs.	527 @ Station 200	642 @ Station 200

It should be noted that increases in the weight of electronic equipment will be reflected in a decrease in ballast and, conversely, that decreases in electronic equipment would increase ballast. Thus, changes in electronics would not appreciably affect the take-off weight of the airplane.

[redacted]

ELECTRONICS:

The revisions to the MG-10 electronic system for the F-102C and D will be principally involved with missile auxiliaries.

The anticipated effect on weights is noted above and no space problems are expected due to electronic changes.

RECOMMENDATIONS:

Based on this preliminary analysis of the problems involved in the F-102C and D configurations, it is recommended that further study be undertaken as follows:

1. Define electronic components for each armament arrangement as accurately as possible.
2. Determine how ballast may be installed in fuselage and its effects on the structure.
3. Consider, insofar as possible, the detail structural changes and mechanisms required to install the F-102C and D armament. This would, of course, involve alternate studies until a preferred method of launching the missiles is found.
4. Check for any requirements peculiar to the arrangement involved which may alter any of the airplane's systems or equipment beyond electronics.
5. Consider rearrangement of any fixed weight items which may reduce ballast.
6. Revise weights.
7. Estimate effects of armament configurations on dynamics of the airplane.

[redacted]

REFERENCES

1. Memo, R. P. White to W. W. Fox, Trip Report - Meeting held at ARDC in Baltimore, Md., on 22 June, 1955. Dated 24 June, 1955.

2. Drawing No. 8-05072, Studies - Ding Dong Installations

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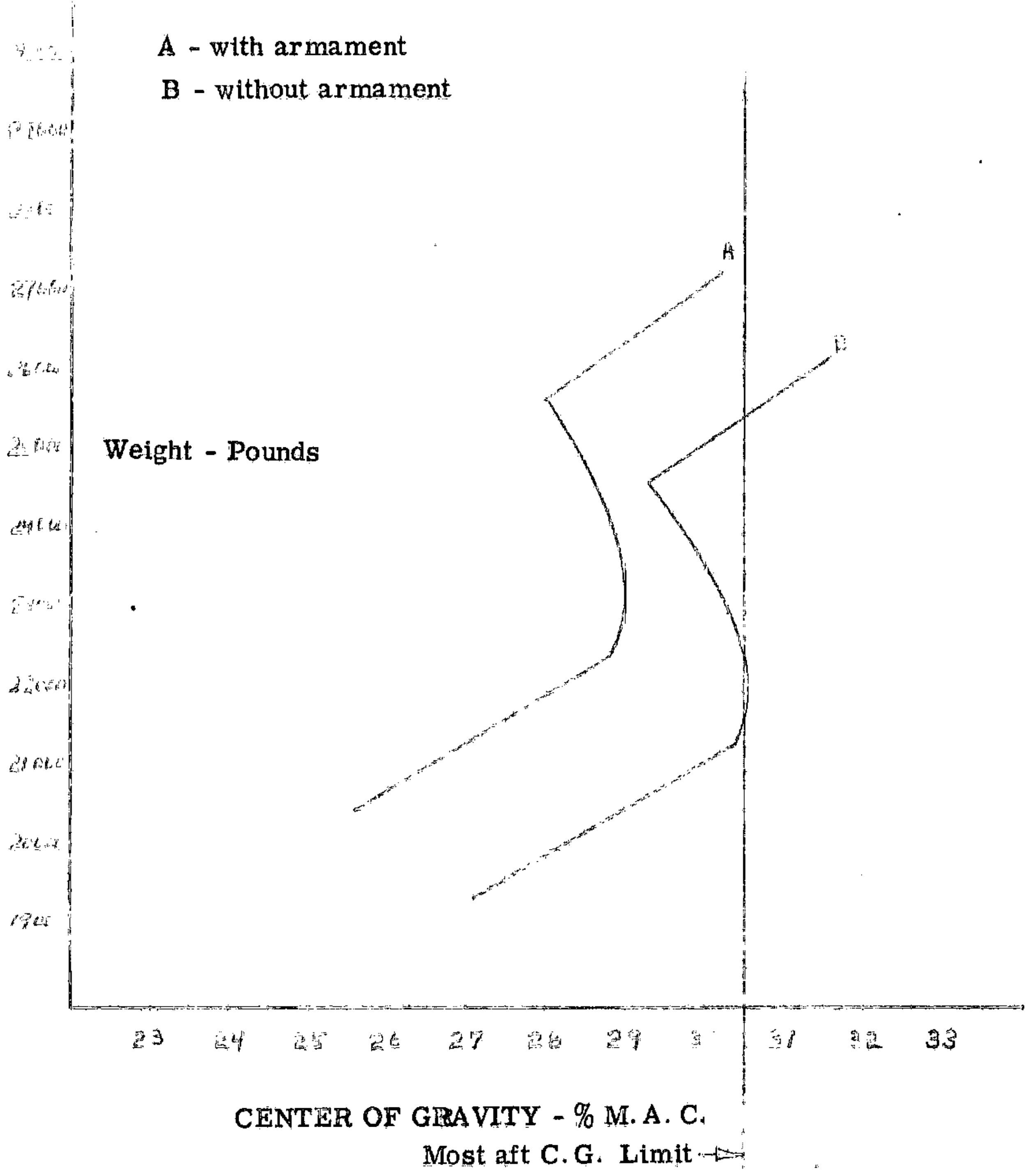
C.G. ENVELOPE - F-102C (1) DING DONG + (2) FALCONS

[redacted]



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C.G. ENVELOPE - F-102C (1) DING DONG + (2) FALCONS

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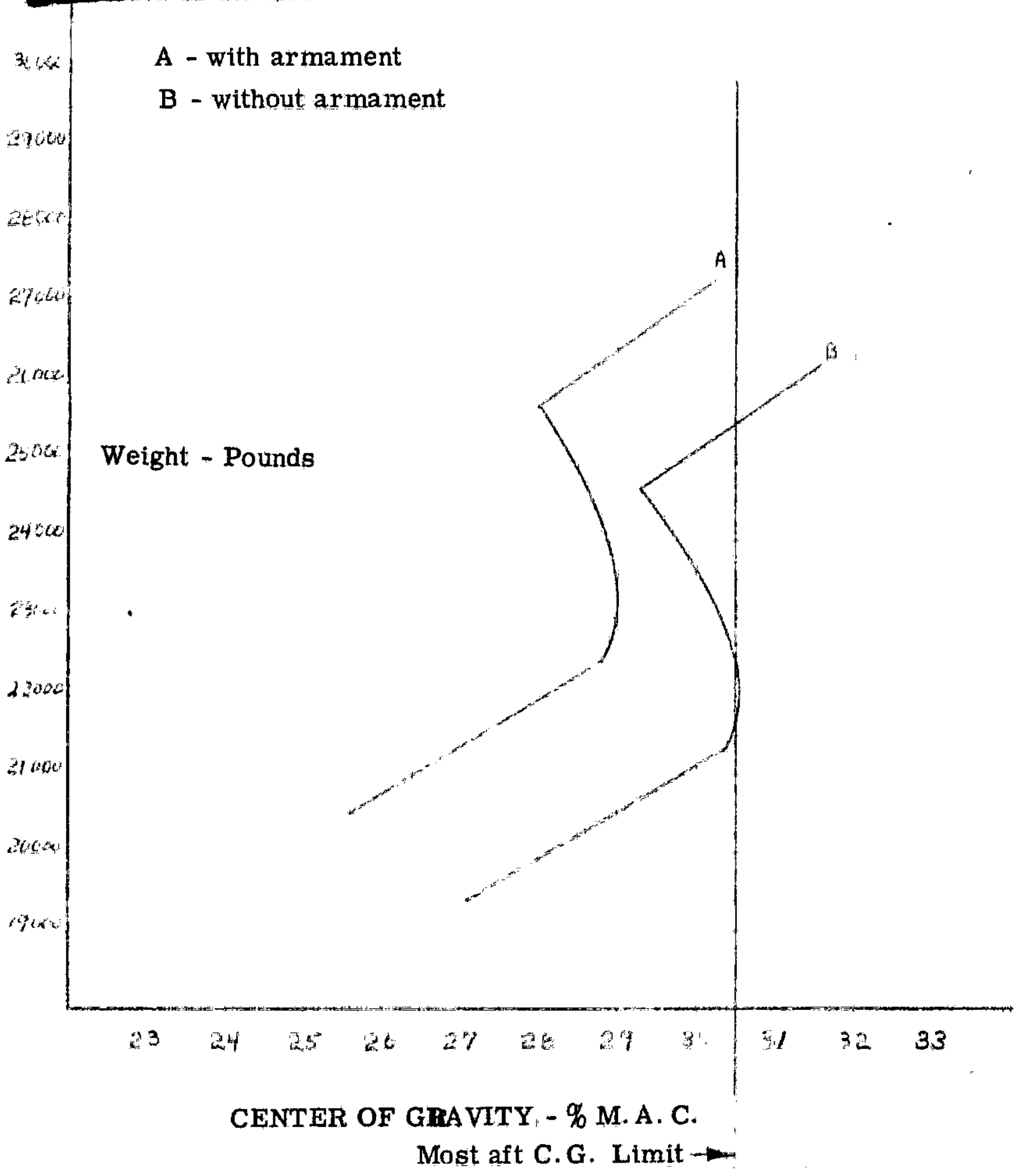

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C.G. ENVELOPE - F-102D (2) DING DONGS + (40) 2" ROCKETS

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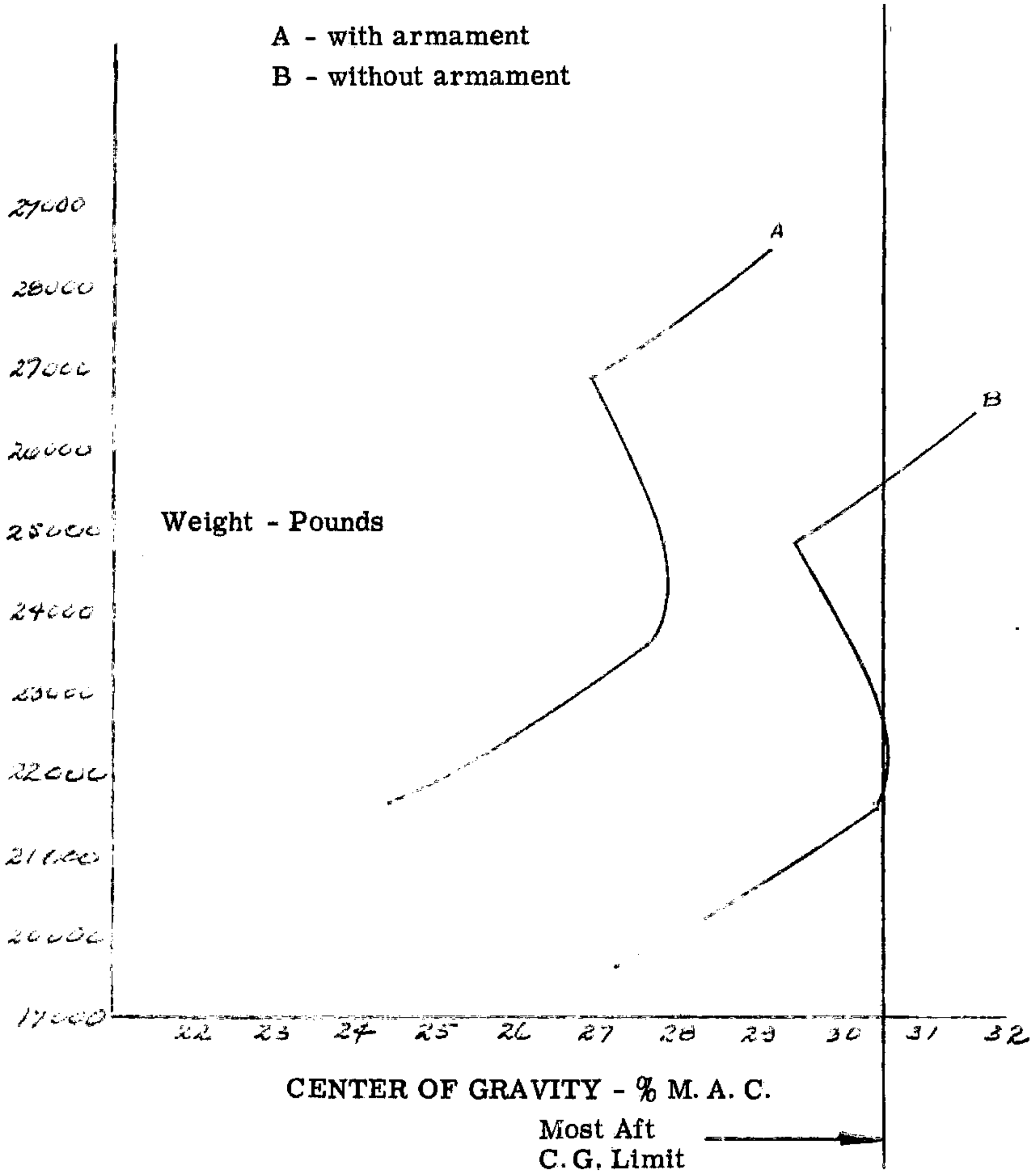

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① EXTENSION SYSTEM (DING DONG)

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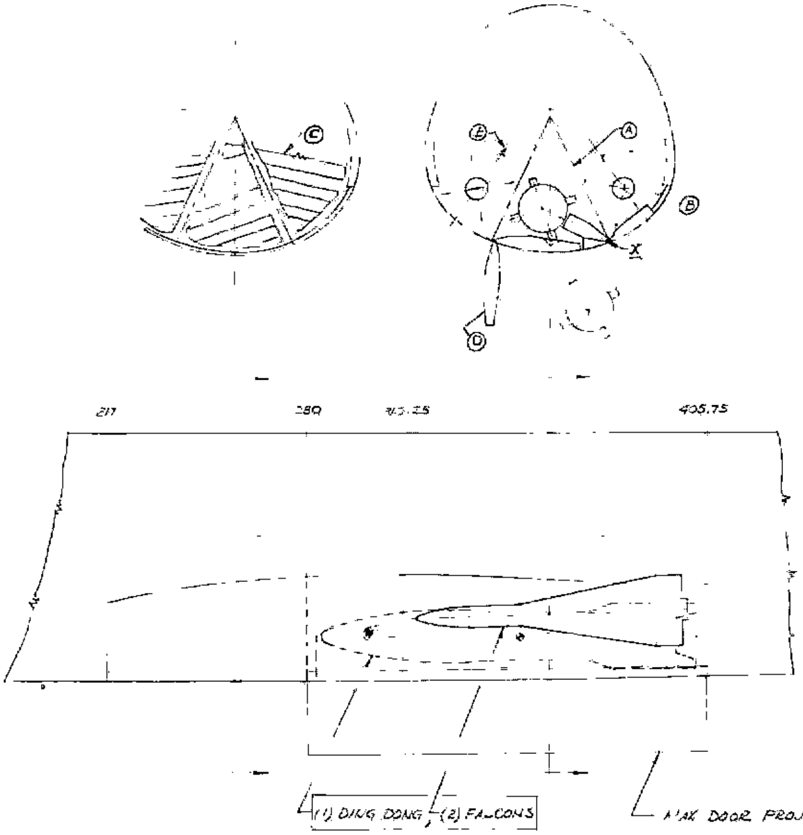

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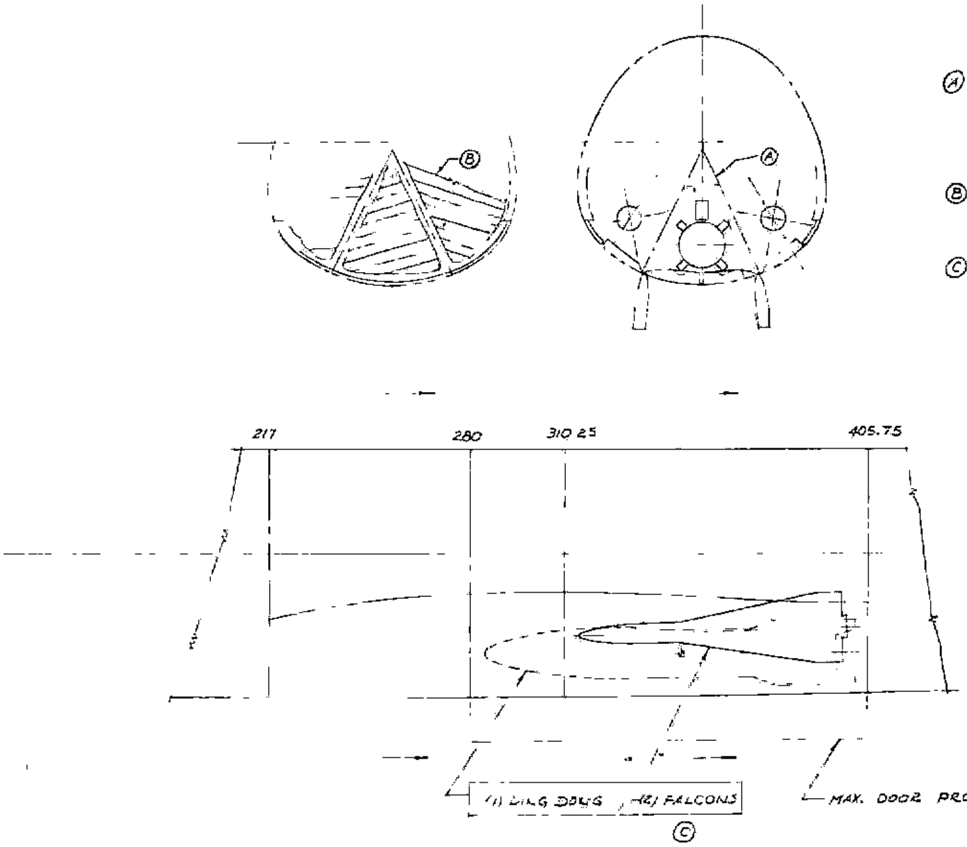
② EJECTION SYSTEM (DING DONG)



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ECT TO WT & BALANCE
IREMENTS (STA 217 TO ≈280
DOOR
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AS F-102A, OR AN EJECTION
M MAY BE USED
FALCONS MAY ALSO BE LOCATED
E FWD. BAY (217 TO 310.25)

BEAMS SAME AS F-102A
LOCAL MODIFICATIONS
ON SYSTEM — DOORS
Y SAME AS F-102A EXCEPT

SPACE AVAILABLE
O WT & BALANCE
(STA 217 TO ≈280)

CONS MAY ALSO BE LOCATED
BAY (217 TO 310.25)

REF DWG 8-05072

[redacted]

CONVAIR

PRELIMINARY DESIGN DRAWING

ARMAMENT STUDIES - F-102C
- (1) DING DONG - 2, FALCONS
INTERNALLY STOWED

CHECKED

CONVAIR—SAN DIEGO, CALIFORNIA 50-55-08057

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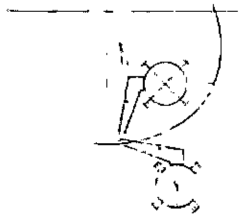
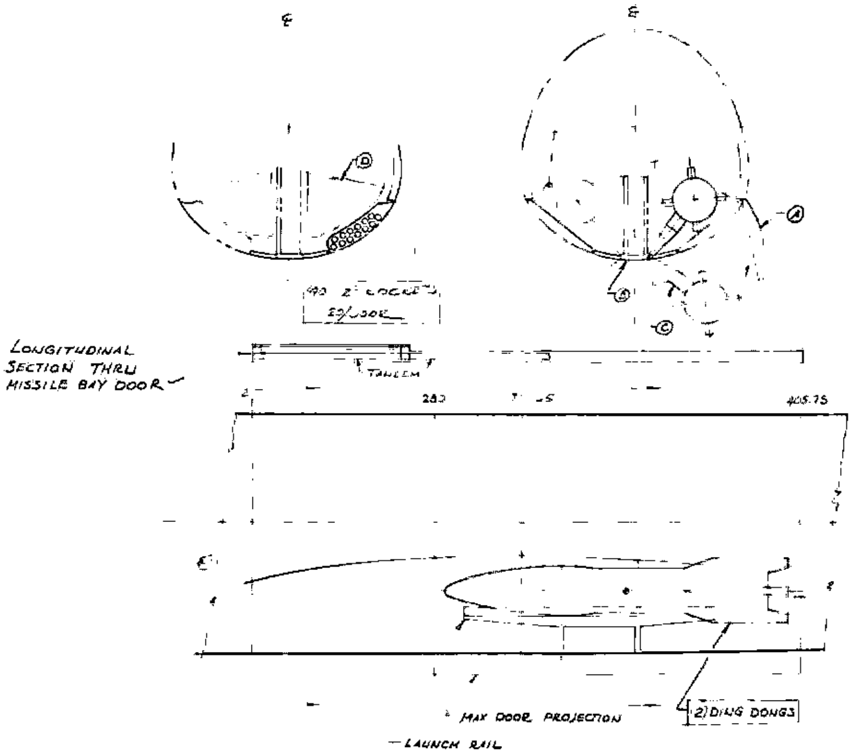
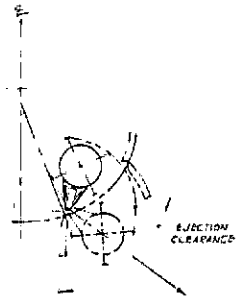
① EXTENSION SYSTEM (DING



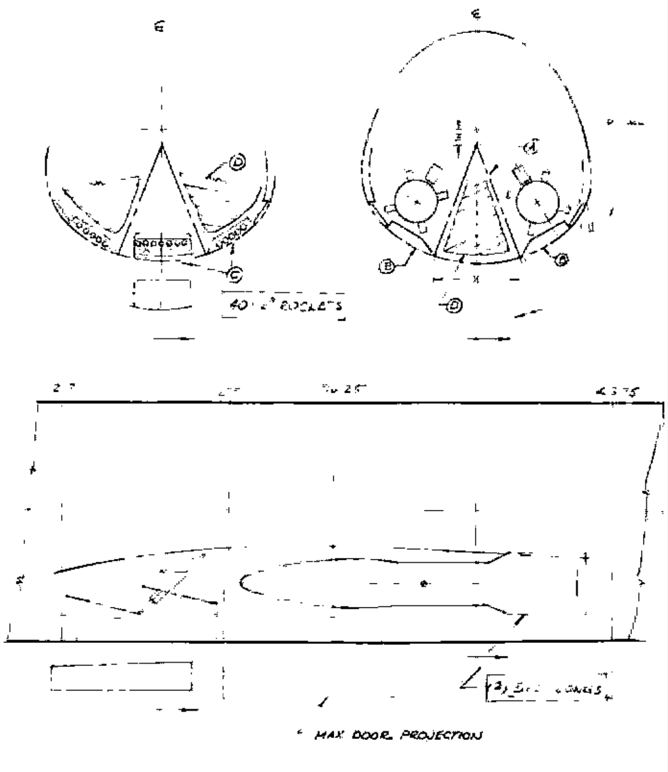
ALTERNATE EXTENSION METHOD



② EJECTION SYSTEM (DING DONGS)
(PRE-SPIN ARRANGEMENT)



ALTERNATE EJECTION METHOD — MISSILE IS EXTENDED, THEN EJECTED. MISSILE BAY BEAMS RELOCATED AS IN SCHEME ②



[redacted]

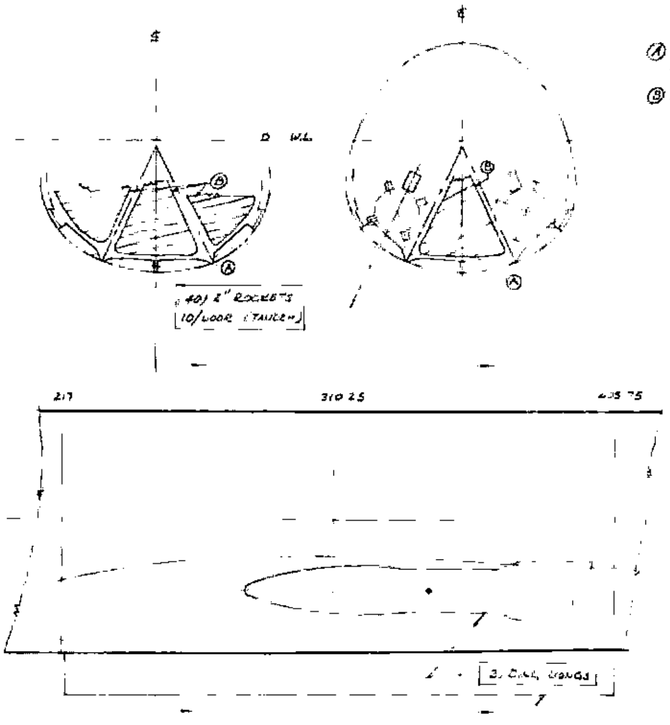
NGS)

Ⓐ NEW MISSILE BAY DOORS
Ⓑ NEW CENTER BEAM STRUCTURE
REPLACES F-102A BEAMS
Ⓒ DING-DONG EXTENSION SYSTEM
Ⓓ ADDITIONAL SPACE AVAILABLE STA 217-290
—SUBJECT TO WT. & BALANCE
REQUIREMENTS

③ EJECTION SYSTEM (DING DONGS)

(NON-SPIN)

Ⓐ MISSILE BAY BEAMS RELOCATED TO PROVIDE CLEARANCE FOR PRE-SPIN DING DONG EJECTION (DIMENSION 'X' IS DECREASED, DIMENSION 'Z' IS DECREASED WITH RESPECT TO F-102A
Ⓑ DOORS REVISED IN ACCORDANCE WITH Ⓐ
Ⓒ ROCKETS IN PACK(S) AND/OR DOORS
Ⓓ ADDITIONAL SPACE AVAILABLE
—SUBJECT TO WT. & BALANCE REQ'MENTS
(STA 217→275 IN OUTB'D BAYS —
CENTER BAY DEPENDS ON ROCKET
PACK ARRANGEMENT)



Ⓐ MISSILE BAY & DOORS ESSENTIALLY THE SAME AS F-102A
Ⓑ ADDITIONAL SPACE AVAILABLE
- SUBJECT TO WT. & BALANCE REQ'MENTS

— MAX DOOR PROJECTION

8-05072

PAGE 8
ZP-8-024

[redacted]

CONVAIR

PRELIMINARY DESIGN DRAWING

ARMAMENT STUDIES – F-102 D
+ 2 DING DONGS + 40 2" ROCKETS
INTERNALLY STOWED

DRAWN	CHECKED	[illegible]	[illegible]	DATE [illegible]

CONVAIR–SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION

SD-55-08058

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-043
DATE 20 July 1956
[redacted]DEL

UNCLASSIFIED

TITLE
STUDY
INSTALLATION OF THE GAR-X MISSILE IN THE F-106A AIRPLANE

ZP-8-043
20 July 1956

Dick Lingley
PREPARED BY R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert

Classification Canceled
USAF 1st Ind. (AFMDC-3E/
Maj. Fristoe/MT/[illegible]1843) 15AU660
Authorized by:
Date
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FORM 1812A-4

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INTRODUCTION

This preliminary study of the installation of the GAR-X missile in the F-106A airplane has been prepared at the request of the Air Force.

The existing F-106A airplane is armed with one MB-1 rocket and four Falcon missiles. This study is based upon replacing the MB-1 with the GAR-X missile and retaining the four Falcons. It also assumes that the GAR-X will be displaced by an ejection system similar to the existing MB-1 method, which has shown satisfactory results in Convair tests to date. Other methods of displacing the GAR-X would have to be studied at a later date if certain limitations on the weapon preclude the use of an ejection system.

Since the anticipated operational date for the GAR-X is 1959 it would be compatible with the F-106A production schedule; however, it would be desirable to incorporate the required changes into the production line as soon as possible to minimize the quantity of airplanes that will have to be modified in a re-cycling program. The nature of the required changes make it a more difficult job to accomplish on a retrofit basis.

It is suggested that consideration be given to a load of (1) GAR-X and (2) Falcons for the retrofit airplanes only. This would greatly reduce the cost of modification on the airplanes that will have been completed before the "one-plus-four" can be incorporated into the production line. It would also insure early delivery of tactical airplanes to Air Force units completely equipped to carry the GAR-X weapon and two Falcon missiles.

Operations Analysis studies show that the possible armament combinations are in the following order of preference:

1. One GAR-X + four Falcons (2 Radar and 2 I. R.)
2. One GAR-X + two Falcons (either radar or I. R.)
3. Two GAR-X + no Falcons

The weight and balance information in this report is based on the 1 + 4 version only.

[redacted]

SUMMARY

Conversion of the F-106A airplane to carry the GAR-X weapon can be accomplished on a retrofit basis with minor modifications if the change is based on a load of one GAR-X and two Falcon missiles. This change would be confined to the area of the ejection rack only.

For the production airplanes (or in a retrofit program) a more extensive modification will be required to provide for one GAR-X and four Falcons.

The existing forward Falcon displacement gear must be removed due to the length of the GAR-X. This gear will be replaced with two individual units to provide for the forward Falcons. In addition, the air conditioning ducts and harnesses must be relocated in the forward missile bay area. Fuselage structure must be modified to accommodate relocated trunnion points, actuators, etc. in addition to the modified ejection mechanism required by the GAR-X. These modifications will be very similar to those now being designed for the F-106B (Tandem) with the exception of the electronics rack in the forward missile bay.

There will be no change required on the missile bay doors or operating mechanism.

Minor revisions will be required on the MA-1 fire control system to provide for the change from the MB-1 rocket to the GAR-X missile. It is estimated that these changes will be confined to the missile auxiliaries and associated circuitry only. Approximately two units will be removed and replaced with four.

There will be no performance degradation on the F-106A airplane when installing the GAR-X missile due to the basic Convair concept of carrying all armament internally.

For the 1 + 4 version there will be a slight weight reduction and the C. G. travel will remain within acceptable limits.

[redacted]

DISCUSSION

The following brief description of the installation of the GAR-X missile in the F-106A airplane is based upon replacing the existing MB-1 rocket with the GAR-X while retaining the four Falcon missiles. Only the ejection system for displacing the GAR-X has been considered in this study.

Further studies must be made to determine the precise detail changes required. The following is a general description only. (Reference drawing SD-56-08053).

1. Remove the existing Falcon missile displacing gear from the forward missile bay.
2. Remove the single actuator for the forward missile gear.
3. Remove the trunnion support structure in the forward missile bay.
4. Install separate displacement gear with individual actuators for the two forward Falcons. (use the Tandem F-106B gear if possible).
5. Provide trunnion support structure for the new missile gear in the forward bay.
6. Revise structure to accommodate the two actuators of the forward Falcon gear.
7. Relocate air conditioning ducts and electrical harnesses to the center area of the forward missile bay (similar to the F-106B).
8. Redesign the rudder cable pulley bracket at station 280.
9. Re-route the throttle teleflex control between sta 280 and 316 to clear the new Falcon gear actuator.

[redacted]

10. Remove the MB-1 ejection mechanism and modify structure accordingly.

11. Provide new ejection mechanism for the GAR-X. (Note:- the ejection points for GAR-X are not compatible with MB-1).

12. Revise structure to provide for the relocated ejection mechanism and associated ejection loads,

13. Revise the pneumatic system to provide for the two new forward Falcon gear actuators and the revised ejection mechanism for the GAR-X.

14. Revise the MA-1 fire control system to provide for the GAR-X in lieu of the MB-1. Preliminary estimates indicate that two units in the missile auxiliary system will be replaced with four units.

15. Provide miscellaneous changes in the electrical system to accommodate the GAR-X missile.

See the following pages for the weight summary, C. G. curve, and drawing of the proposed installation.

[redacted]

PRELIMINARY WEIGHT STUDY

	WEIGHT
F-106A with MB-1 Type Rocket and (4) GAR-3/-4 Missiles	33100 *
Changes:	(- 25)
(2) Single vs. (1) Missile Displacement Gear in Fwd. Bay	(+ 126)
Displacement Gear and Actuating System	+ 76
Missile Bay Structure	+ 40
Electronic Harnesses	+ 10
GAR-X vs. MB-1 Type Rocket, Including Installation	(- 151)
Weapon (619# vs. 815#)	- 196
** Ejector System	---
Wiring	+ 15
MA-1 System	+ 15
Miscellaneous [redacted]	+ 15
F-106A with GAR-X Weapon and (4) GAR-3/-4 Missiles	33075

* Gross Weight as of 15 July 1956
** Assumed same as MB-1 Type Rocket Ejector System

[redacted]

COMPARATIVE GROUP WEIGHT STATEMENTS
F-106A
WITH (4) GAR-3/-4 MISSILES AND WITH MB-1 TYPE ROCKET OR GAR-X GUIDED WEAPON

	F-106A (BASIC) *	F-106A WITH GAR-X **
Wing Group	3231	3231
Tail Group	636	636
Body Group	4656	4696
Alighting Gear Group	1193	1193
Surface Controls Group	437	437
Engine Section	37	37
Propulsion Group	(7978)	(7978)
Engine Installation	5911	5911
Air Induction System	888	888
Exhaust System	262	262
Cooling System	46	46
Lubricating System	50	50
Fuel System	724	724
Engine Controls	26	26
Starting System	71	71
Fixed Equipment Group	(4952)	(5083)
Instruments and Navigational Equipment Group	134	134
Hydraulic and Pneumatic Group	381	381
Electrical Group	547	547
Electronics Group	2543	2583
Armament Group	684	760
Furnishings and Equipment Group	264	264
Air Conditioning and Anti-Icing Group	346	361
Auxiliary Gear Group	53	53
WEIGHT EMPTY	23120	23291
Useful Load	(9980)	(9784)
Pilot, Survival Kit, and Parachute	270	270
Fuel - Unusable (JP-4) 36 Gals.	234	234
Fuel - Usable (JP-4) 1240 Gals.	8060	8060
Oil - Trapped and Engine	60	60
MB-1 Type Rocket	815	
GAR-X Weapon		619
Ejector Cartridge	1	1
GAR-3/-4 Missile (4)	540	540
NORMAL TAKE-OFF GROSS WEIGHT	33100	33075

* Based on Convair Weight and Balance Status Report dated 15 July 1956

** Preliminary Estimate

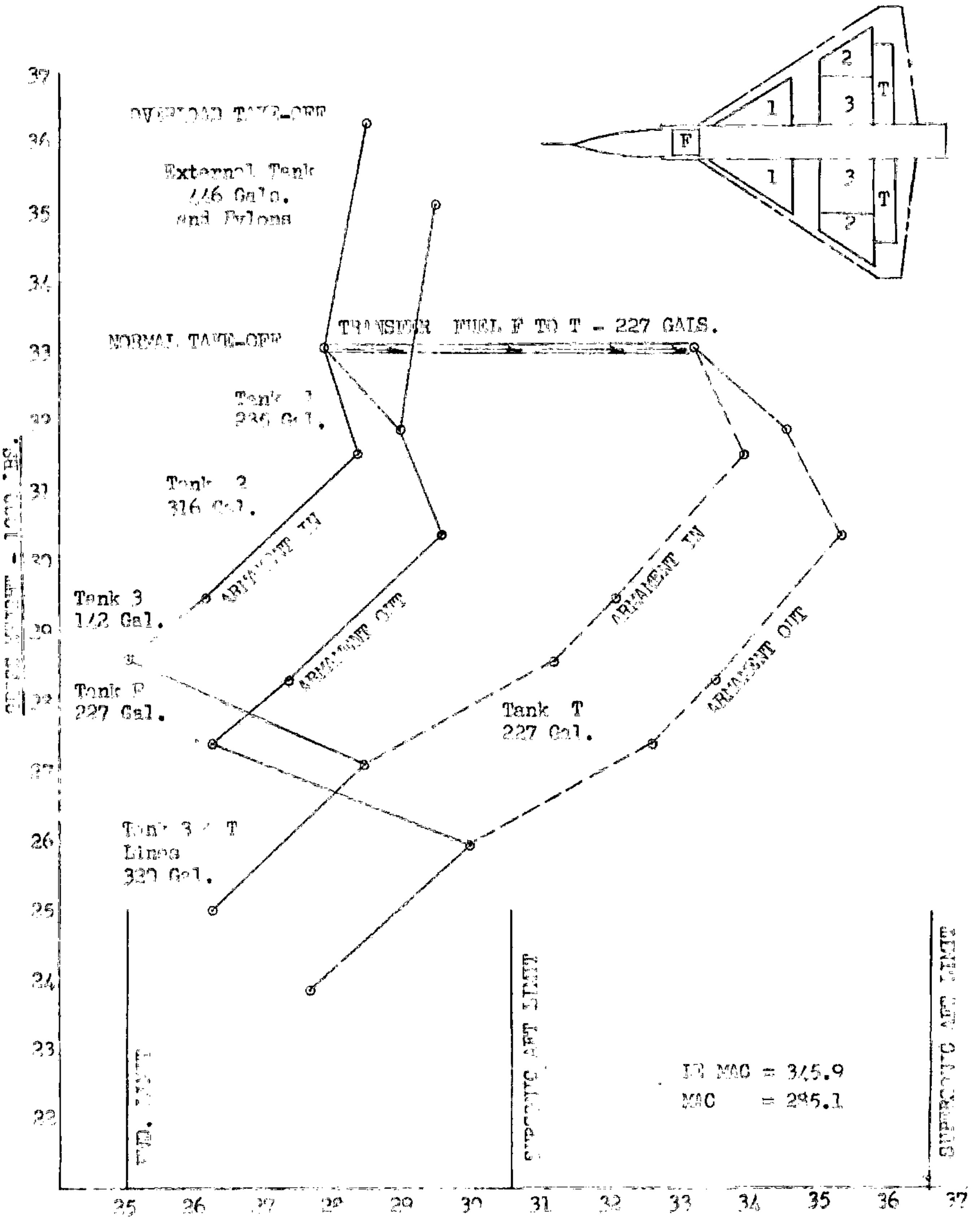
[redacted]

F-106A WITH GAR-X WEAPON AND (4) GAR-3/-4 MISSILES

PRELIMINARY CENTER OF GRAVITY ENVELOPE

6° CRUISE ATTITUDE

JP-4 FUEL



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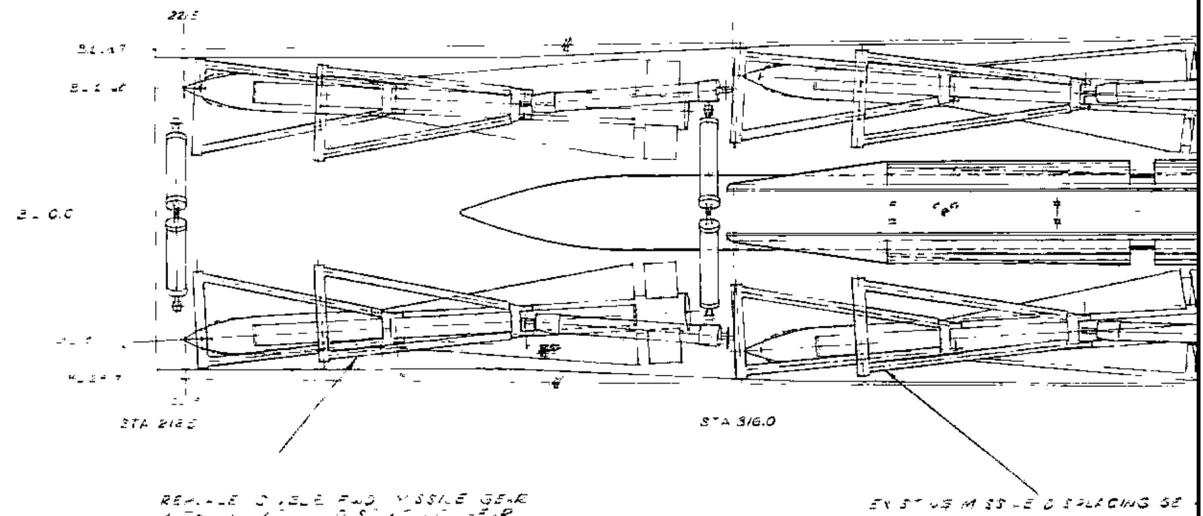

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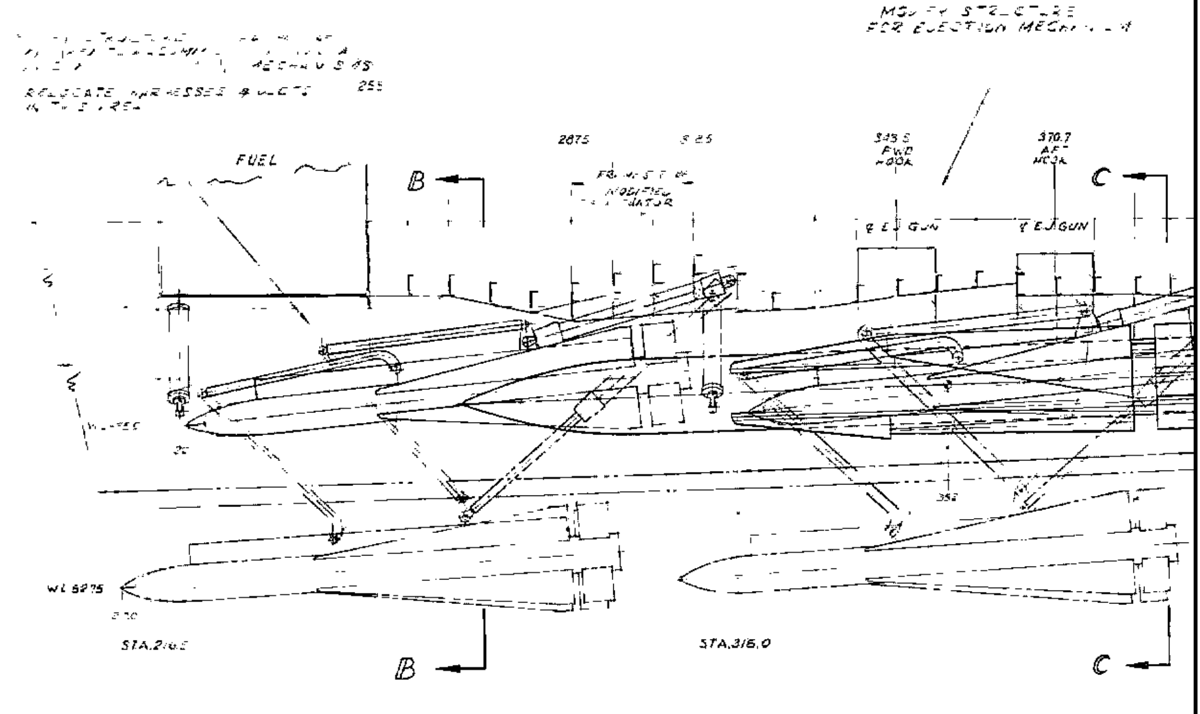

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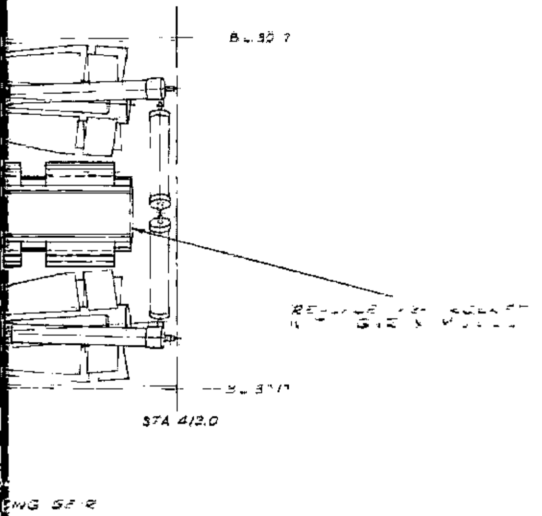
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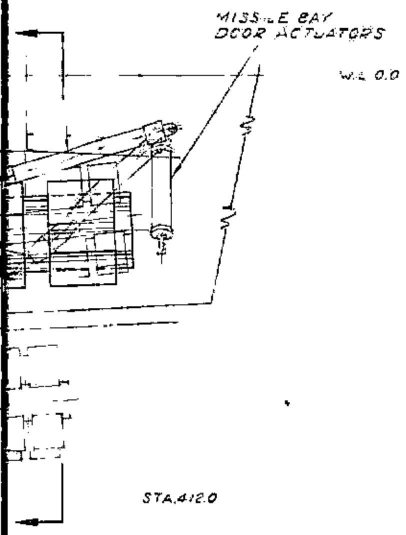
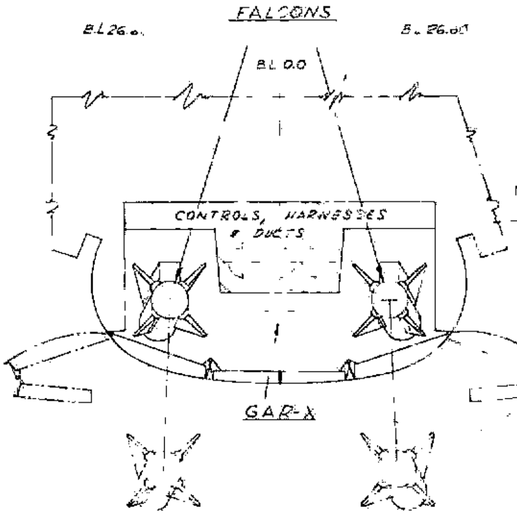
NG GEAR


MISSILE BAY
DOOR ACTUATORS
W.L. 0.0
STA.412.0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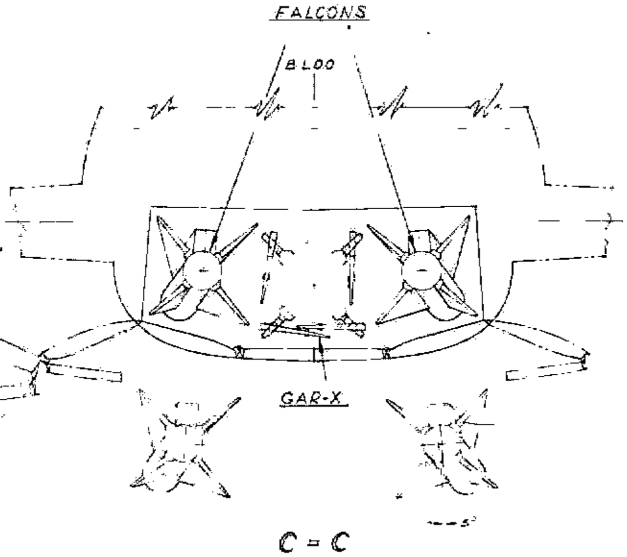
CONTROLS, HARNESSES
& DUCTS
GAR-X

B – B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5°

C – C

Use as + Orig.

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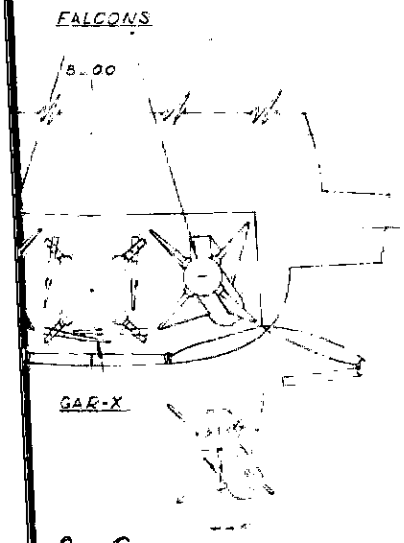

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CONVAIR
PRELIMINARY DESIGN DRAWING

STUDY
(1) GAR-X + (4) FALCON MISSILES
F-106A AIRPLANE

BY | CHECKED | APPROVED | SCALE 1/10 | DATE 7-15-56

CONVAIR—SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION
SD-56-08053

ZP-8-029

ZP-8-029

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

[redacted]

REPORT ZP-8-029
DATE 26 Oct. 1955
MODEL 8

UNCLASSIFIED

TITLE

J-75 ENGINE WITH LONG AFTERBURNER EFFECT ON F-102 B AIRPLANE

ZP-8-029

26 October 1955

PREPARED BY R. C. Lingley
R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A R Lambert
A. R. Lambert

NO. OF PAGES

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REVISIONS

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FORM 1812 A-4

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EFFECT ON F-102B AIRPLANE

Introduction:

This study has been made at the request of the Air Force to determine the penalties involved when installing the longer J-75 engine in the F-102B airplane.

Summary:

It is not recommended that the long afterburner be incorporated into the F-102B since the performance penalties and design changes coupled with the obvious increased costs and schedule slippages do not warrant the change.

Discussion:

The effect on the airplane is as follows:

1. Physical Changes - (Ref. Dwg. SD-55-08074). Since it is not possible to move the engine forward due to the location of the main landing gear bulkhead, all the changes are confined to that portion of the fuselage from Station 630 aft.

   The engine shroud is increased 20.65 inches in length.

   The end of the tail cone is moved aft a corresponding 20.65 inches.

   It is not possible to "straight line" the tail cone from the canted bulkhead to the new aft end because the engine shroud clearance must be maintained. Hence, new rear end lines must be faired from Station 630 to the aft end.

   The increased diameter of the lines at the speed brake intersection cuts area from the lower portion of the speed brakes. This area must be replaced near the aft end of the speed brakes causing the center of pressure to move aft with an increase in actuator loads.

   With the tail cone moved aft 20.65 inches, the airplane touchdown angle is decreased one degree. (From 17° - 30' to 16° -30'.)

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[redacted]

UNCLASSIFIED

[redacted]

2. Weight and Balance Changes (Ref. Dwg. SD-55-08074). The long afterburner adds 144 pounds to the airplane weight empty and is distributed as follows:

Power Plant Group	
Engine	+75 lbs.
Shroud	+33 lbs.
Fuselage Group	
Tail Cone	+31 lbs.
Speed brakes	+ 5 lbs.
Total Weight Empty	+144 lbs.

This causes an aft C. G. shift of .66% M.A.C.

To maintain the balance within limits, an additional 207 pounds of ballast must be added for a total gross weight increase of 351 pounds.

To eliminate the ballast, the wing would have to shift aft 3 to 4 inches. This would cause extensive changes to the airplane which are not outlined in this report due to the limited time available.

A detail analysis of the effect of moving the wing would have to be the subject of a future study.

3. Performance Penalties - The loss of one degree of touchdown angle due to the longer tail cone would increase the landing and take-off speed 5 knots.

No change in high speed performance is caused by the extended tail cone lines.

The 351 pounds increase in gross weight will decrease the subsonic and supersonic ceiling 260 feet.

This increased gross weight will also decrease the combat radius by 7 nautical miles.

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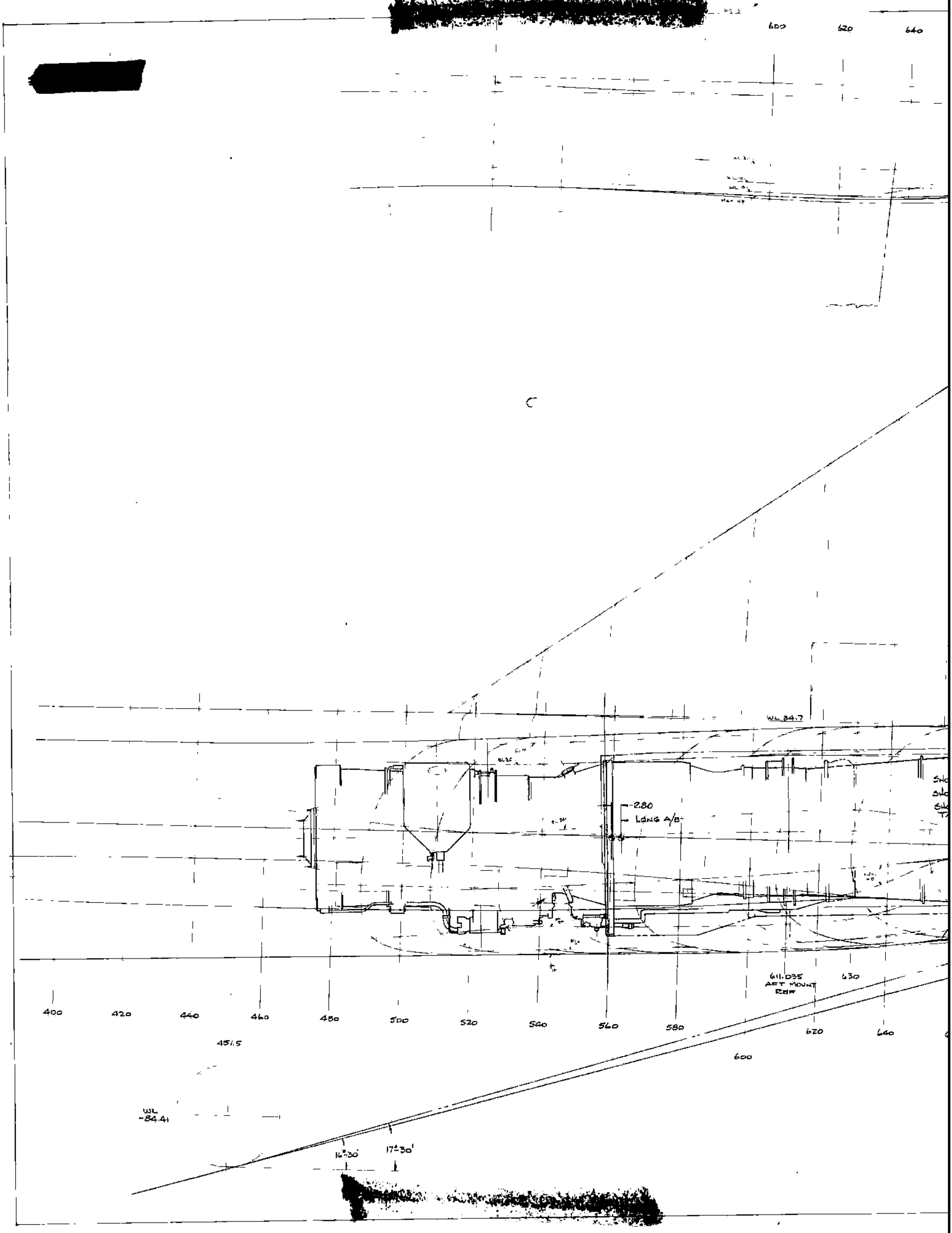

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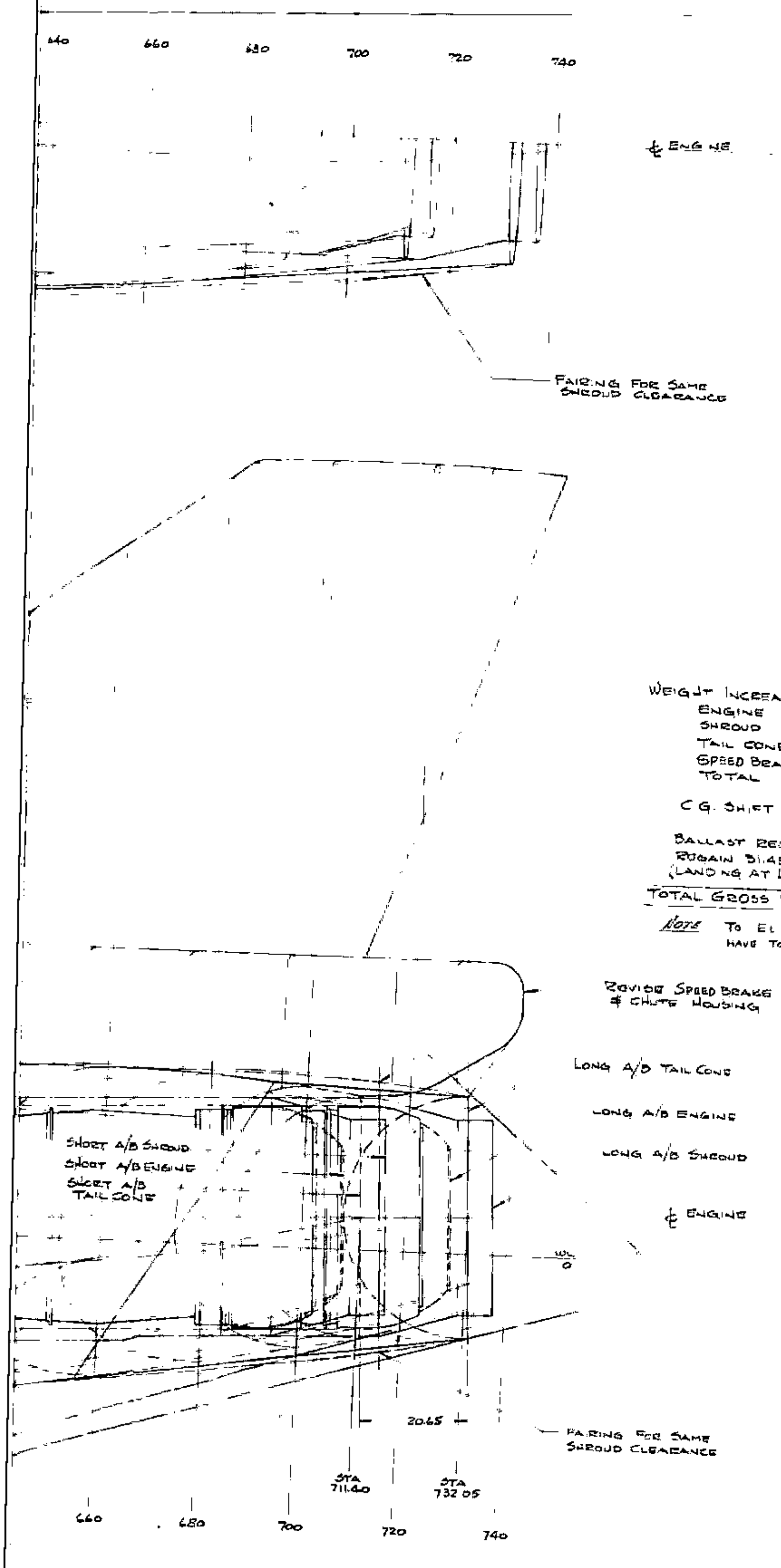

WEIGHT INCREASE FOR J75 LONG A/B ENGINE

ENGINE	75.0 LBS.
SHROUD	33.0 LBS.
TAIL CONE	31.0 LBS
SPEED BRAKE	5.0 LBS
TOTAL	144.0 LBS
C.G. SHIFT AFT	.66%
BALLAST REQUIRED TO REGAIN 31.43% M.A.C. (LANDING AT LOW GROSS WT)	207.0 LBS
TOTAL GROSS WEIGHT INCREASE	351.0 LBS.

NOTE TO ELIMINATE BALLAST, WING WOULD HAVE TO MOVE AFT APPROX. 5½ INCHES.

[redacted]

CONVAIR

PRELIMINARY DESIGN DRAWING

STUDY - J.75 ENGINE with
LONG AFTERBURNER
F-102B AIRPLANE

RUDIS	CHECKED	SCALE TENTH	DATE 10/24/56
AIR-SAN	OF GEN	CORP	SD-55-08074

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CONSOLIDATED VULTEE AIRCRAFT CORPORATION
ENGINEERING DEPARTMENT . . SAN DIEGO, CALIFORNIA

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Herewith Copy No. _______ of Convair Secret Report No. ZP-8-026, "A Study of the F-102B as a Reconnaissance Airplane", dated September 15, 1955.

Please sign this receipt and return to Preliminary Design Section, San Diego Engineering Department.

CONVAIR, A DIVISION OF GENERAL DYNAMICS CORPORATION

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FORM 22-A

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-026
DATE Sept. 15, 1955
MODEL F-102B
Revised 16 Jan. 1956

[redacted]

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TITLE

A Study of the F-102B

as a

Reconnaissance Airplane

Classification Changed to

UNCLASSIFIED

Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by: Dept. Date
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PREPARED BY

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert

NO. OF PAGES

NO. OF DIAGRAMS

REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED
1	1-16-56		Rewritten to conform with drawing changes	1 and 1 thru 17
2	1-16-56		Drawings revised	SD-55-08064, 08067, 08070, 08071
3	"		Drawings SD-55-08083, 08082, 08085 replace respectively	SD-55-08065, 08068, 08069

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7. Strike and Bomb Damage
Assessment Installation 4
8. Night Photo Reconnaissance Installation 4
9. Ferret Reconnaissance Installation 5
10. Radar Charting (all weather) Installation 7
11. Synoptic Weather Reconnaissance Installation 8
12. Pod Mounted Gun Installation 9
13. Weight Summary 10
14. Drawings 11
15. Performance Summary and Mission Profiles . . . . . . Appendix A

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F102B RECONNAISSANCE AIRPLANE

INTRODUCTION

This report presents some brief exploratory studies of Reconnaissance versions of the USAF F102 Interceptor, now in production. The purpose of the report is twofold. It is to invite the attention of Air Force personnel to the performance of the F102 when equipped as a reconnaissance airplane, and to provide a basis for discussions to develop the most suitable reconnaissance arrangements.

The F102 Program includes a plan of continuing development. F102A airplanes are presently in production. The F102B production will follow. This type will incorporate more effective electronic fire control equipment and armament and will have a more powerful engine. Its performance will be greatly advanced.

The first flight of the F102B series is scheduled for December 1956. The production build up of F102B airplanes will start in August of 1958.

If reconnaissance airplanes are desired on an earlier schedule, the installations shown herein can be made in the F102A type airplanes. The range of the F102A and F102B is the same, the superior performance of the F102B in other respects is the principle difference between the two. The performance of the F102A was shown in Convair Report No. ZP-8-007

The report is in two sections. The first consists of Descriptions and Drawings of various reconnaissance versions of the F102B. The second section shows the Performance of the Day Photographic Reconnaissance version of the airplane.

It is believed that the photo reconnaissance versions of the F102B airplane studied herein will provide a means of attaining the normal photographic reconnaissance objectives; operational intelligence, strike assessment, terrain intelligence and technical intelligence. The radar reconnaissance version will permit attaining these same objectives during weather when the use of photographic equipment is not feasible. Operational and technical information concerning enemy defenses will be obtained by the ferret version of the airplane. A synoptic weather version will assist in obtaining weather data.

The fuselage structure varies between several of the equipment installations. Actually the percentage change in fuselage structure between them is quite small though at first glance it may appear as if a new fuselage has been used.

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GENERAL ARRANGEMENT

The F102 Interceptor is an all weather supersonic fighter designed to destroy enemy bombers. Its conversion to a reconnaissance airplane is easy due to the favorable location of electronic fire control equipment and armament. Drawing SD55-08071 shows the General Arrangement of the Reconnaissance F102B. The nose shape ahead of the cockpit is only slightly different from that of the F102B Interceptor. Otherwise the external dimensions are the same.

FUSELAGE SPACE DIAGRAM

Drawing SD55-08084 is a fuselage space diagram of the F102B. The cross-hatched areas indicate the spaces occupied in the Interceptor by electronics equipment and armament. In the Reconnaissance versions of the airplane these spaces are available for the installation of cameras and other reconnaissance equipment or fuel.

CAMERA AND CAMERA SYSTEM DEVELOPMENT

The cameras described in the following pages and shown on the installation drawings are "off the shelf" equipment. This equipment was used principally to show the space available. In almost every instance there is being developed an equivalent piece of equipment both lighter and better suited to the purpose. In addition, the new equipment is being designed to facilitate picture correlation. This is being done by printing correlation data either in the corner or on the edge of the picture.

New camera equipment being developed is being done on a system basis so that the camera control system is very much simplified over most of those now in use.

Convair has reasonable assurance that the camera combinations shown or any other combination of equivalent sizes and not more numerous can be developed into an integrated operational system for installations by 1958.

DAY PHOTO RECONNAISSANCE INSTALLATION

Two arrangements of the cameras in the Day Photographic version of the airplane are shown on drawing SD-55-08064. In the arrangement on the right, there are three camera groups and a T.V. viewfinder.

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DAY PHOTO RECONNAISSANCE INSTALLATION (cont.)

The first group consists of three 6" focal length cameras at stations 1 and 2 in a tri camera arrangement. Three KA-8 cameras used in this group will provide 542 nautical miles of stereo coverage each, when photos are taken at 10,000 feet altitude. The scale will be 1:20000 on a 9 x 9 format. The cycle rate of 6 cycles per second is ample for taking overlapping (60%) pictures down to 500' altitude.

The second group consisting of two K-38 cameras is in a split vertical arrangement. With 24 inch focal length lenses, a scale of 1:20000 will be obtained at 40,000 feet altitude.. Using standard A-8 magazines each camera will permit 256 nautical miles of stereo coverage. The cycle time of 1-1/2 seconds is adequate for taking overlapping pictures down to 14,000 feet at the maximum speed of the airplane.

At Station 5 is a K-38 camera with a 36" focal length lens. This camera can be used for more detail or for area coverage at very high altitudes, say above 50,000 feet altitude. At 50,000 feet, the scale would be better than 1:20,000. With a standard A-8 magazine, (260 exposures) it will be possible to get 214 nautical miles of stereo coverage at 50,000 feet with this camera.

Total stereo coverage for the three camera groups is over 1000 nautical miles. It is 800 nautical miles with the two forward camera groups.

The television viewfinder at Station "O" permits the pilot to align the cameras over a specific target or put the airplane on a given track.

As indicated on the drawing, other cameras may be used instead of those described above.

In this and all other arrangements of reconnaissance equipment studied a common set of communications, navigation and landing equipment has been used. The items included are:

Communication Set	AN/ARC-34
Homing adapter	AN/ARA-25
Radio Navigator Set (Tacan)	AN/ARN-21
I.L.S. Glide Slope and Localizer Receiver	AN/ARN-31
IFF Transponder	AN/APX-6
Air Data Computer	
Inertial Navigation System	

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DAY PHOTO RECONNAISSANCE (cont.)

There are several air data computers available. However, it is expected that some alterations to any of them will be required depending on the particular reconnaissance equipment finally chosen for installation.

The inertial navigation system will probably be one of three now being developed.

The alternate installation on the left of the drawing shows a short focal length stabilized vertical camera at station 1. The tri-camera installation is at stations 2 and 3. Station 4 has a KA-2 or a K17c 12" lens camera in a fixed vertical mount. Station 5 is the same as in the installation described above.

This alternate arrangement will give the same lineal coverage but 1:20,000 scale pictures will be available from 20,000 feet altitude instead of from 40,000 feet. In addition, a stabilized short, focal length vertical as well as a stabilized long focal length vertical camera is provided.

STRIKE AND BOMB DAMAGE ASSESSMENT INSTALLATION

The arrangement of cameras on drawing SD55-08067, for Bomb Damage Assessment is the same as that for Day Photo Reconnaissance with two exceptions. The tri-camera group of 6 inch KA-8 cameras is replaced by a single forward obique K-38 camera and the split vertical group is replaced by a set of two K-38 cameras mounted so that pictures can be taken to the side.

The forward looking camera is fitted with a 36 inch lens. The two K-38 cameras looking to the side have 24"lenses. They are arranged so that if both are trained to the same side their images will overlap. Their field of view extends from almost vertical nearly out to the horizon. Thus anything that can be seen visually to the side will come into the camera's view.

Flying at 20,000 feet will produce a picture with a minimum scale of 1:10,000 in this side looking group. The lineal coverage available with the standard A-8 magazine is 128 miles.

NIGHT PHOTO RECONNAISSANCE INSTALLATION

An arrangement of equipment for night reconnaissance is shown on Drawing SD55-08082. Forward of the cockpit are a Trip Control Unit and Flash detector and two camera groups. The first camera is a vertical K-43 with a 6 inch focal length lens. Immediately aft of it are two K-43 cameras with 6 in. lenses in a split vertical arrangement.

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NIGHT PHOTO RECONNAISSANCE INSTALLATION (Cont.)

Aft of the cockpit is a stabilized vertical K-47 camera with a 24 inch lens. Aft of this camera in an armament bay are mounted eight M-120 Photo Flash Bombs on a retractible rack and 260 rounds of M112 cartridges in fixed racks.

The single vertical K-43 camera can be used with either the flash bombs at intermediate altitudes or the flash cartridges in the higher part of the low altitude range.

At extremely low altitudes the split vertical K-43 cameras would give more width coverage.

At 3000 feet altitude the single K-43 camera in conjunction with the 260 flash cartridges can obtain a charted strip 38.5 nautical miles by .37 n. miles wide. With a flash projector such as the Edgerton Electronic type even more linear coverage would be available.

Using the K-47 camera in conjunction with the M120 Photo Flash Bombs, pictures with a scale of 10,000 are obtained at 20,000 feet altitude.

This arrangement of cameras gives considerable flexibility in the altitude at which night photography may be accomplished.

FERRET RECONNAISSANCE INSTALLATION

The Ferret system is shown on Drawing SD 55-08083. It consists of a group of forward antennae, a group of aft antennae and corresponding groups of airborne detecting, analyzing and recording equipment. The forward antennae are in the nose radome and aft of them are the electronic units that they serve. Aft of the cockpit are the second set of electronic components and immediately aft of them in the under side of the fuselage are the aft antennae. The Communication, Navigation and Landing equipment are above the aft ferret electronic components in the same position they occupy in the Photographic Reconnaissance versions of the airplane.

The ferret system obtains data for both operational and technical intelligence uses. The whole system includes a Ground Installed Reconnaissance Data Handling System in addition to the airborne systems mentioned above. These systems are being developed by Melpar, Incorporated of Alexandria, Va.

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The Electromagnetic Reconnaissance System provides the function of detecting and analyzing repetitive pulse-type electro-magnetic radiations. The antenna units located in the nose radome and the bay aft of the aft electronics compartment intercept essentially all pulse-type electromagnetic radiation which illuminates the aircraft in two 120 degree sectors centered abeam the aircraft from the horizontal to 45 degrees below the horizontal. These antennas, in conjunction with the analyzer and programmer units located in the forward and aft electronics compartments, provide the analysis of frequency and direction of the intercepted signal.

The frequency of received signals is determined instantaneously with a resolution of ± 5 percent in the range of 30 to 40,000 magacycles. The Electromagnetic Reconnaissance System, in conjunction with the Ground Data Handling system is capable of determining the location of enemy radars on any frequency from 60 to 31,500 magacycles.

The analysis of frequency and direction provides the following parameters which are recorded in digital form on magnetic tape: frequency, direction, pulse width, pulse-repetition frequency, and time of intercept. In addition, a video recorder located in the aft electronics compartment provides a photograph of a video pulse presentation of the intercepted signal. The Electromagnetic Reconnaissance System is capable of recording 70,000 such intercepts per mission. The airborne intercept analysis and recording of one signal is automatically accomplished in 0.1 second or less.

The Electromagnetic Reconnaissance System is designed as nearly as possible to have an equal probability of intercepting the various type transmissions such as search and tracking signals. System philosophy and reliability considerations are based on the following in their order of importance:

(1) Detection (frequency determination)

(2) Function of intercepted signal

(3) Coverage (location of source)

Although the location of the signal source is last in the order of importance, most radars can be located within a circle of equal probability having a radius of 20 miles or 20 percent of the range, whichever is the greater, provided sufficient intercepts are made.

At the time of each analysis of frequency and direction, the Recording System records navigational and atmospheric data along with the common element, time.

In the post flight phase of a mission, the Ground-Installed Reconnaissance Data-Handling System, a wholly automatic system, uses the common factor, time,

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to correlate the Electromagnetic Reconnaissance System data with the Recording System data. The data is reduced to graphic and printed forms suitable for human analysis. In this ground system, approximately 80 percent of the ultimate attainable data from a ferret reconnaissance mission can be obtained in about three hours time.

RADAR CHARTING (ALL WEATHER) INSTALLATION

For the equipment arrangement in this installation refer to Drawing No. SD-55-08085.

The high resolution reconnaissance APS-60 (XA-2) radar receiver-transmitter unit and antenna are mounted in the nose radome. The remainder of the AN/APS-60 (XA-2) radar units, with the exception of the pilot's radar indicator, are located in the compartment directly behind the radome. An auxiliary indicator and its associated camera for obtaining a permanent record of the reconnaissance data observed on the pilot's indicator is shown in the aft electronics compartment.

Also in the aft electronics compartment is shown the space which is available for mounting the "black boxes" of an AN/APQ-56 sidelooking reconnaissance radar and its associated recording equipment. The AN/APQ-56 antenna location is shown in the compartment aft of the aft electronics compartment in the bottom of the fuselage. The bottom of the fuselage is made of radome type materials replacing the armament bay doors of the interceptor.

The AN/APS-60 (SARTACK) radar is being developed by Convair, San Diego, under the direction of the Aerial Reconnaissance Laboratory of the Wright Air Development Center.

This radar has been designed for all-weather around-the-clock detection of both moving and fixed ground targets. Low and medium altitude surveillance of interdiction targets by a tactical aircraft, using this high resolution radar, supplies an important new reconnaissance capability.

The AN/APS-60 radar is a 60 K_w peak power output K_u - band radar which operates in the frequency band of 16,500 ± 200 megacycles per second. This radar, utilizing a roll, pitch, and yaw stabilized assembly which combines the receiver-transmitter and antenna, is estimated to be capable of the following performance at altitudes of 20,000 to 30,000 feet:

Land Mapping	40 to 50 miles
Scattered Targets (Isolated)	100 miles

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RADAR CHARTING (ALL WEATHER) INSTALLATION (Cont.)

Cities	125 miles
Large Cities	150 miles
Highway Traffic (Airborne Moving Target Indication)	20 miles

The AN/APQ-56 side-looking reconnaissance radar, developed by Westinghouse, is utilized in this study to provide a radar system capable of obtaining a maximum amount of high definition radar information during a radar reconnaissance mission. The AN/APQ-56 is capable of obtaining this information at night or other times when the earth may be obscured by smoke, haze, clouds, or other similar atmospheric conditions.

This radar operating at 30 K_w peak power output in the K_a frequency band uses back-to-back antenna arrays approximately ten feet in length. The radar provides ground strip mapping up to approximately 45,000 feet in altitude with a coverage of 10 to 20 miles on either side of the aircraft.

The AN/APQ-56 radar mapping information is recorded on film for analysis at ground stations.

The combination of the AN/APS-60 and AN/APQ-56 radars should provide excellent radar reconnaissance coverage.

SYNOPTIC WEATHER RECONNAISSANCE INSTALLATION

For the equipment arrangement used in this installation, refer to Drawing No. SD-55-08070.

Antenna assemblies of the AN/APS-42A Weather Radar and a Cloud Base and Top Radar (CB&T) are located in the nose radome of the aircraft. Other "black boxes" of these radars are located in the forward electronics compartment, the aft electronics compartment, and in the belly compartment aft of the aft electronics compartment. An APS-42A indicator is located in the cockpit to provide visual flight information to the pilot. A recording system consisting of a strip recorder, a power supply, and an air data recorder are located in the aft electronics compartment. Space provisions have been retained in this compartment for installation of equipment for transmission of radar weather data back to the home base during flight when this equipment becomes available. Space provisions have also been set aside in the compartment aft of the aft electronics compartment for installation of receiving equipment, radiosonde type transmitter packages, and radiation level measuring equipment.

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SYNOPTIC WEATHER RECONNAISSANCE INSTALLATION (cont.)

The AN/APS-42A Weather Radar, produced by Bendix, is an X-band radar which operates at a minimum peak power output of 40 K_w. The radar is used in this installation primarily for the location of light and heavy areas of turbulence within storms or cloud formations. The radar also provides a ground search function which is useful as a navigational aid.

The Cloud Base and Top Radar is used in this installation to obtain cloud density information above and below the aircraft flight path. The general dimensions and configurations of the radar components shown are Convair's best estimate. Convair found no company developing equipment for this purpose. However, the principles involved have been examined and it is reasonable that a number of subcontractors could be developed to produce the equipment.

The same is true of equipment for measuring pressure, temperature and humidity below the flight path of the aircraft. While radiosonde equipment utilizes the principles needed, a more rugged airborne transmitter unit would have to be developed to permit release at high speeds. In addition some experimental work on trajectories and radiation characteristics would be required.

POD MOUNTED GUN INSTALLATION

A study of a pod mounted 20 mm T-160 gun installation is shown on Drawing SD55-08086. This pod would be a temporary installation to give the airplane a fighting capability during the early stages of a conflict before air superiority is gained.

Two installations would be mounted, one on either side of the airplane in place of the wing drop tanks. Provisions for 300 rounds of ammunition per gun are made.

The optical sighting system already in the airplane with little modification together with a small computer, an infra red adapter and an infra red search head will be used for fire control.

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F-102 RECONNAISSANCE AIRPLANE

VII GROUP WEIGHT SUMMARY

BASIC DATA & ENEMY INTELLIGENCE CAMERA INSTALLATION
(REF. DRAWING SD-55-08064)

Wing Group		3295.0	
Tail Group		570.3	
Fuselage Group		4747.0	
Landing Gear		1406.8	
Surface Controls		443.5	
Nacelle		33.5	
Power Plant Group		8408.8	
Engine	6125.0		
Exhaust System	232.7		
Cooling System	71.8		
Air Induction System	878.6		
Lubricating System	35.2		
Fuel System	964.1		
Engine Controls	28.6		
Starting System	72.8		
Instruments and Navigation		137.9	
Hydraulic and Pneumatics		319.6	
Electrical System		487.9	
Electronics and Navigation		482.0	
Armament		88.0	
Furnishings		297.3	
Air Conditioning and Anti-Icing		355.8	
Auxiliary Gear		68.6	
Photographic Group		239.0	
TOTAL WEIGHT EMPTY			21,381.0
Crew		270.0	
Unusable Fuel		284.0	
Fuel - Wing & Fuselage		9574.0	
Fuel - Missile Bays		5990.0	
Engine and Unusable Oil		60.0	
Camera Equipment		953.0	
External Tanks and Pylons		693.0	
External Fuel - 1,100 Gals.		7150.0	
Missile Bay Tank		460.0	
TOTAL USEFUL LOAD			25,434.0
TAKE-OFF GROSS WEIGHT			46,815.0

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REF. IBM #2

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DRAWINGS

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REPORT ZP-8-026
DATE Sept. 15, 1955
MODEL F-102B

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by: Dept. Date
130-1 4-4-67

TITLE

APPENDIX "A"

Performance Summary & Mission Profiles

A Study of the F-102 B

as a

Reconnaissance Airplane

Report ZP-8-026

PREPARED BY Mario D'Annunzio
Mario D'Annunzio
M R Tyson
M. R. Tyson

CHECKED BY J. F.

GROUP Aerodynamics Group
Preliminary Design Group

REFERENCE

APPROVED BY D. H. Bennett
D. Bennett

NO. OF PAGES

NO. OF DIAGRAMS

REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED

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[redacted]

ANALYSIS
PREPARED BY
CHECKED BY
REVISED BY

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
(SAN DIEGO)

PAGE 1 App. "A"
REPORT NO. ZP-8-026
MODEL F-102B
DATE Sept. 15, 1955

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PERFORMANCE SUMMARY

P & W J75-JT4A-24 TURBOJET

BASIC OPERATIONAL DATA & ENEMY INTELLIGENCE CAMERA INSTALLATION
(REF. DWG. SD-55-08064)

Weights

Design Take-Off	46,750 lbs.
Maximum Fuel	22,714 lbs.
Operating Weight Empty*	23,343 lbs.
Maximum Combat**	33,850 lbs.
Landing***	25,268 lbs.

Speeds (At Combat Weight of 33,850 lbs.)

Maximum at 35,000 ft. with Max. Power	M = 2.00
Maximum at 35,000 ft. with Mil. Power	M = 1.01
Cruise Speed, Optimum (w/wout external fuel tanks)	531 knots

Climb (At Combat Weight of 33,850 lbs.)

Maximum Rate of Climb at Sea Level	35,300 fpm
Combat Ceiling (M = .94)	51,600 ft.
Take-Off Ground Roll at Design Take-Off Weight	3,440 ft.

Landing Ground Roll at Landing Weight

With drogue chute	2,300 ft.
No drogue chute	3,100 ft.

Engine Ratings (P & W Spec #2604)

Normal Rated Power	13,500 lbs.
Military Power	15,500 lbs.
Military plus Afterburner Power	23,500 lbs.

* Design T.O. weight less all fuel and external tanks.
** Max. weight over target (based on single stage mission).
*** Operating weight empty plus reserves (5% initial fuel plus 20 min. endurance at S.L.)

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FORM 1812-A-1

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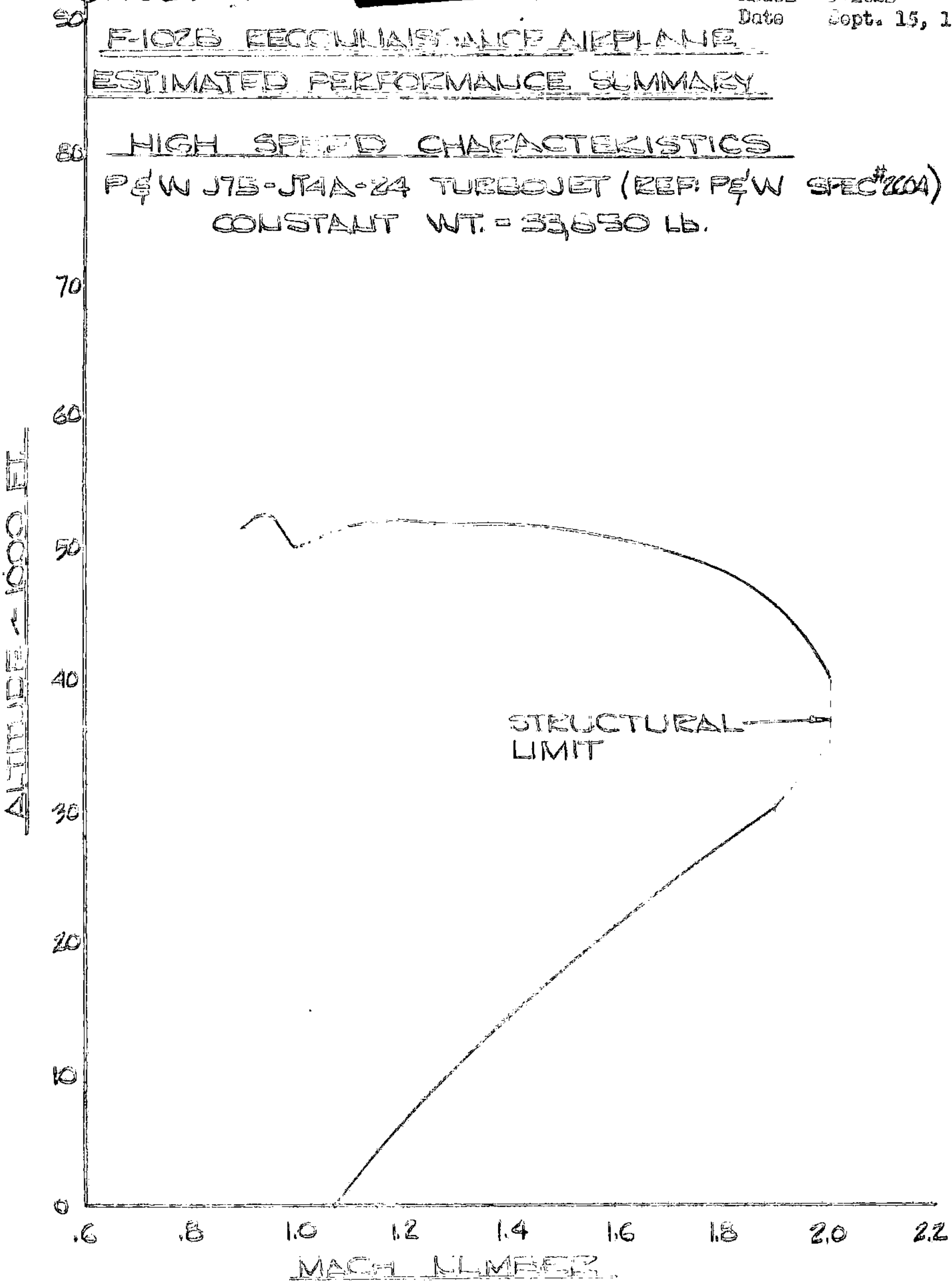
F-102B RECONNAISSANCE AIRPLANE
ESTIMATED PERFORMANCE SUMMARY

HIGH SPEED CHARACTERISTICS

P&W J75-JT4A-24 TURBOJET (REF: P&W SPEC #2604)
CONSTANT WT. = 33,650 Lb.



MRP [redacted] 9-95

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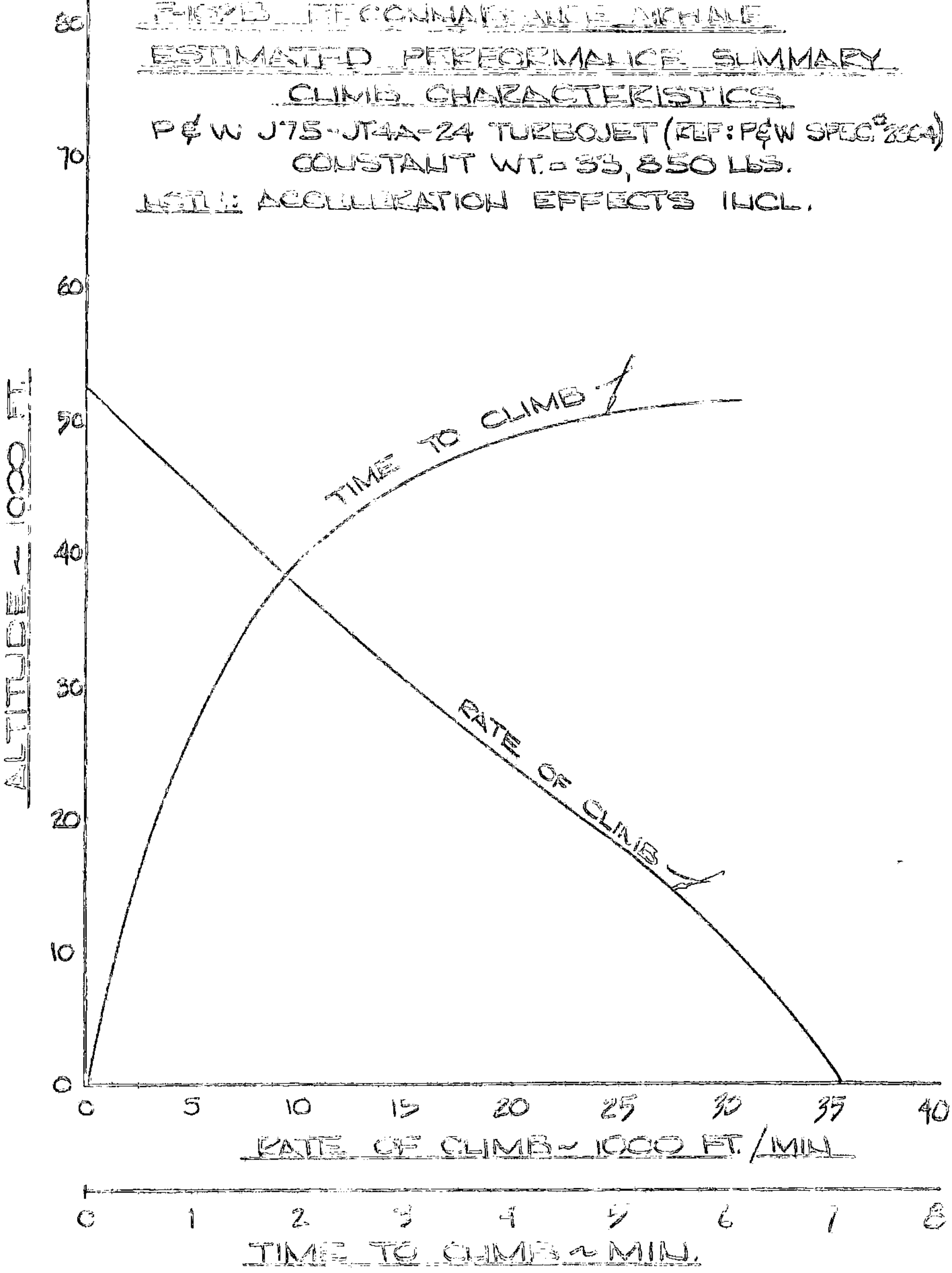

F-102B RECONNAISSANCE AIRPLANE
ESTIMATED PERFORMANCE SUMMARY
CLIMB CHARACTERISTICS

P&W J75-JT4A-24 TURBOJET (REF: P&W SPEC # 2364)
CONSTANT WT. = 33,850 LBS.
NOTE: ACCELERATION EFFECTS INCL.

MEF- [redacted] 5-55

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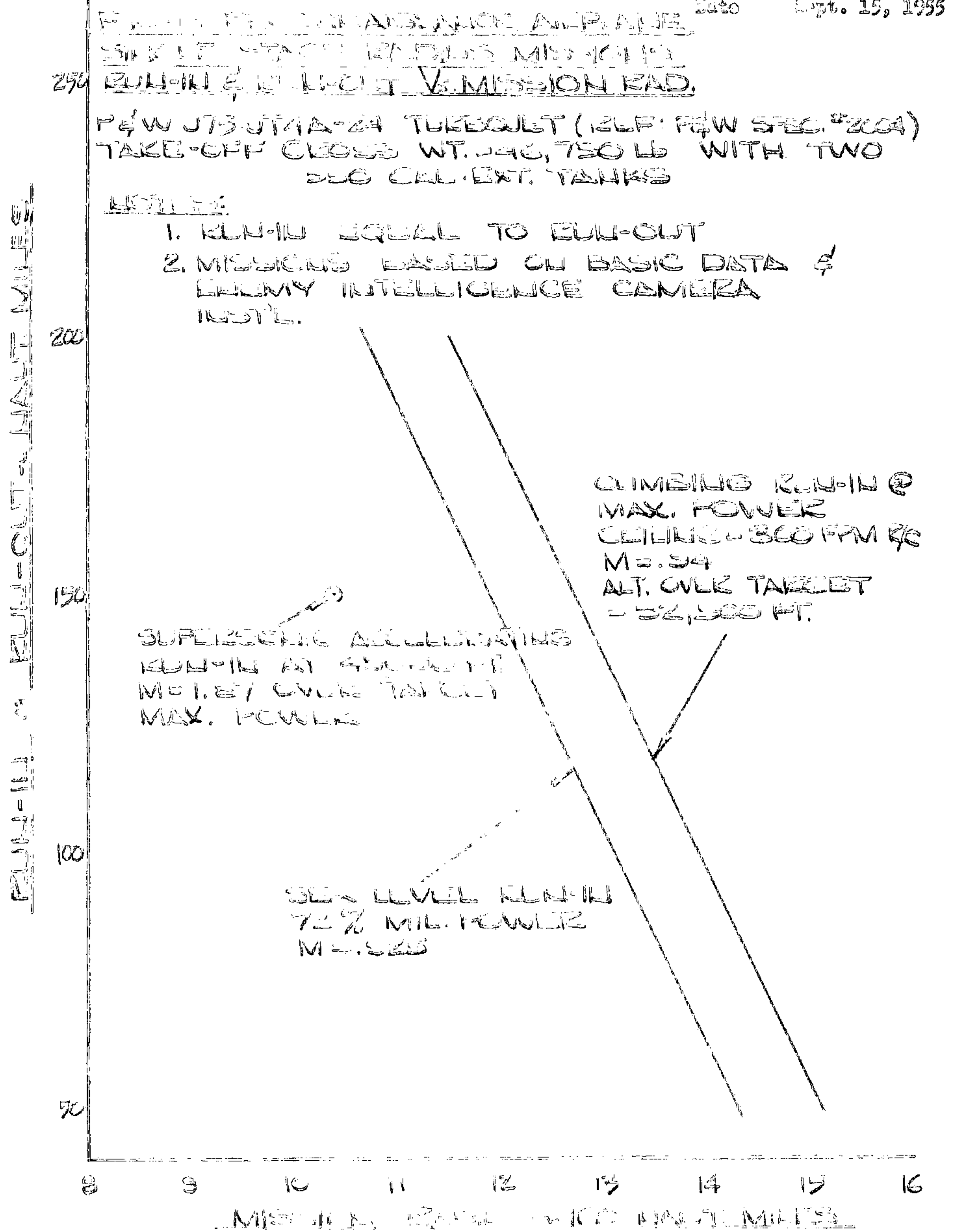

[illegible] RECONNAISSANCE AIRPLANE
SINGLE STAGE [illegible] MISSIONS
RUN-IN & RUN-OUT VS. MISSION RAD.

P&W J75-JT4A-24 TURBOJET (REF: P&W SPEC. #2004)
TAKE-OFF GROSS WT. 46,750 LB WITH TWO [illegible] GAL. EXT. TANKS

NOTES:

1. RUN-IN EQUAL TO RUN-OUT
2. MISSIONS BASED ON BASIC DATA & EMPTY INTELLIGENCE CAMERA INST'L.

[redacted] 2-55

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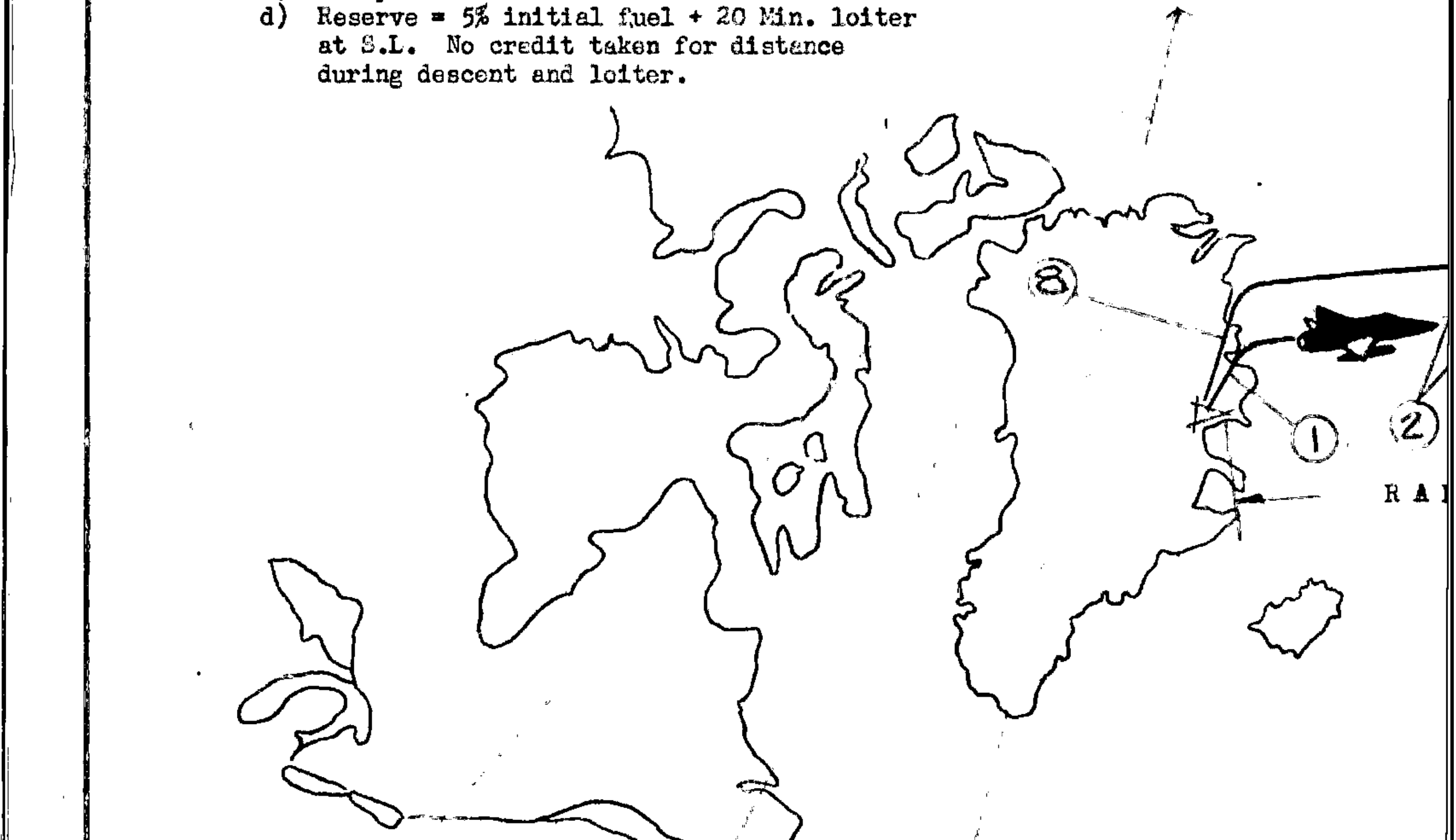
MISSION PROFILE - F-102B RECONN

SINGLE STAGE - SEA LEVEL ALTITUD

TAKE-OFF GROSS WEIGHT 46

a) Warm up, take-off and acceleration allowance = 5 Min. at NRP + 1 Min. A.B.
b) Optimum Cruise altitude.
c) Escape climb at Mil. Power.
d) Reserve = 5% initial fuel + 20 Min. loiter at S.L. No credit taken for distance during descent and loiter.



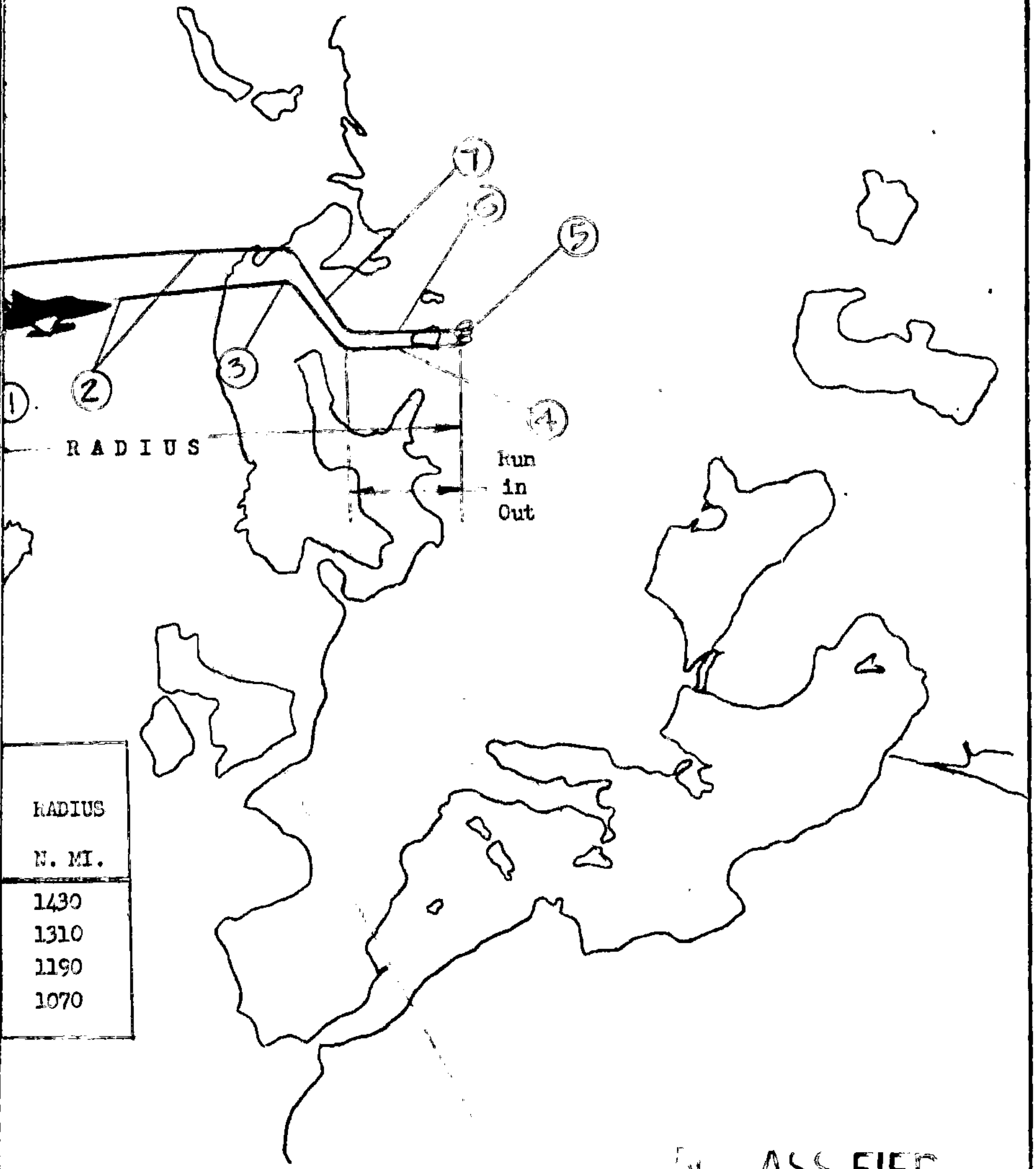
RUN-IN-OUT DIST. TO TARGET	RADIUS N. MI.
50	1430
100	1310
150	1190
200	1070

1. Climb at Mil. power to cruise altitude.
2. Cruise climb at M = 0.925.
3. Drop external tanks.
4. Run in at 0.925 M at S.L. at 72% Mil. power.
5. 5 Min. of evasive action over target.
6. Run-out at 0.925 M at S.L. at 72% Mil. power.
7. Climb at Mil. power.
8. Descent and loiter at S.L.

2B RECONNAISSANCE AIRPLANE

L ALTITUDE OVER THE TARGET

WEIGHT 46,750 LBS.



RADIUS N. MI.
1430
1310
1190
1070

UNCLASSIFIED

[redacted]

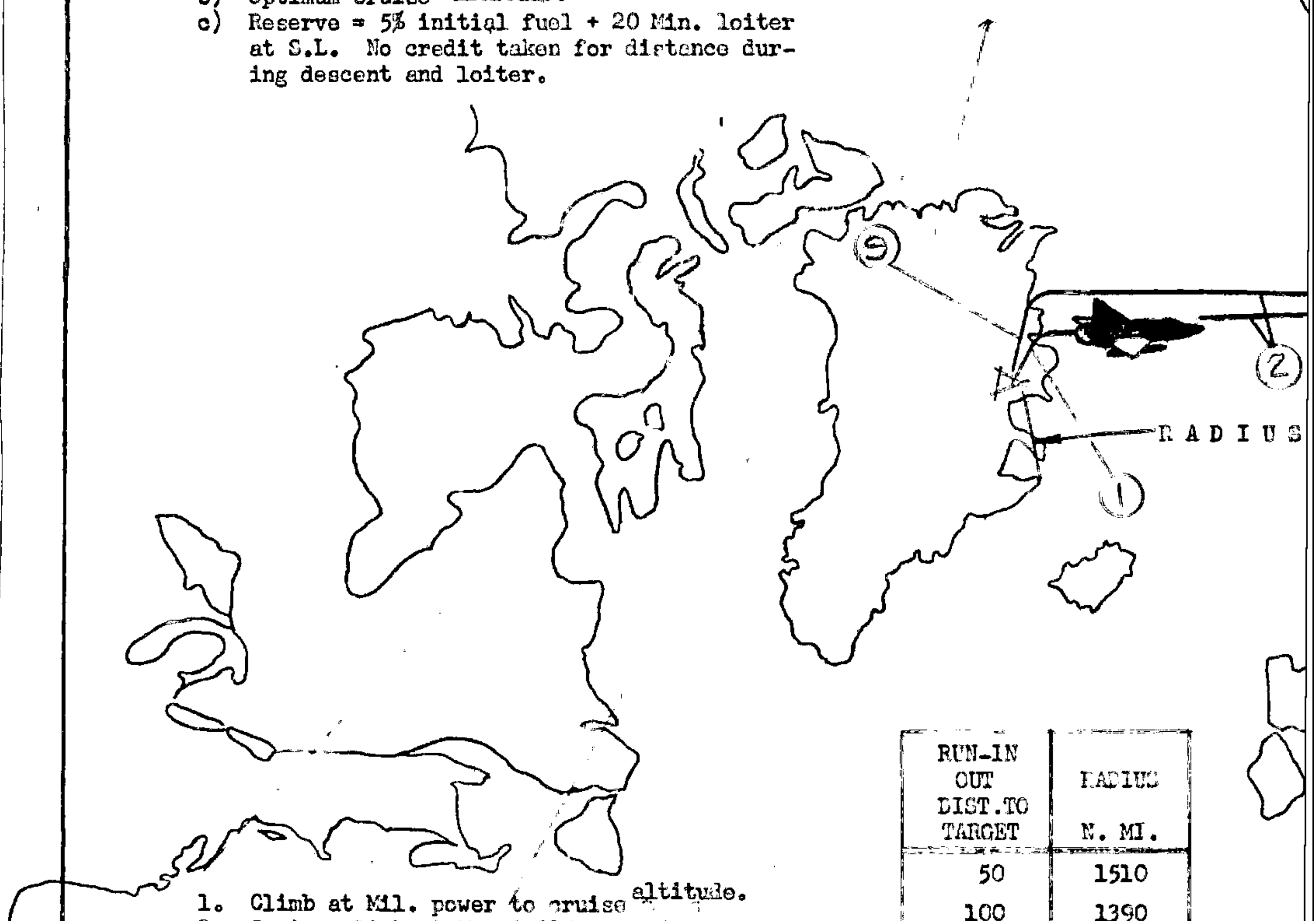
MISSION PROFILE - F-102B RECONNAISSAN

SINGLE STAGE - HIGH ALTITUDE MISS

TAKE-OFF GROSS WEIGHT 46,750 LBS

UNCLASSIFIED

a) Warm up, take-off and acceleration.
allowance = 5 Min. at NRP + 1 Min. A.B.
b) Optimum cruise altitude.
c) Reserve = 5% initial fuel + 20 Min. loiter at S.L. No credit taken for distance during descent and loiter.



RUN-IN OUT DIST. TO TARGET	RADIUS N. MI.
50	1510
100	1390
150	1270
200	1150

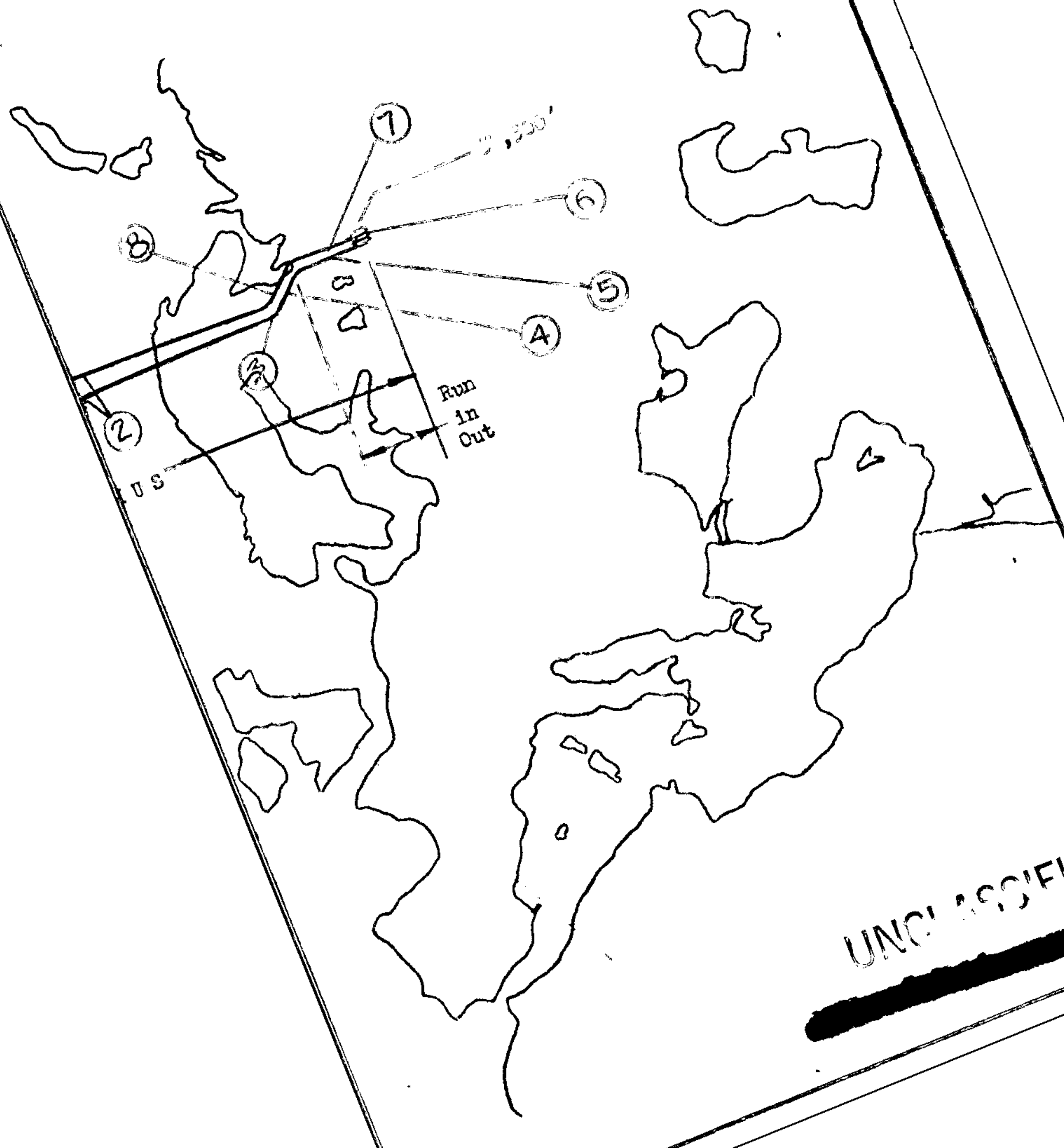
1. Climb at Mil. power to cruise altitude.
2. Cruise climb at M = 0.925.
3. Drop external tanks.
4. Climb at Mil. + A.B. power to Max. power cruise ceiling.
5. Run-in at 0.94M at Mil. + A.B. power at Max. power cruise ceiling.
6. 2 Min. evasive action over target.
7. Run-out at 0.94M at Mil. + A.B. power.
8. Descend to cruise altitude.
9. Descend and loiter at S.L.

Page 6 App. "A"
Report ZP-8-026
Model F-102B
Date Sept. 15, 1955

ISSANCE AIRPLANE

MISSION

0 LBS.



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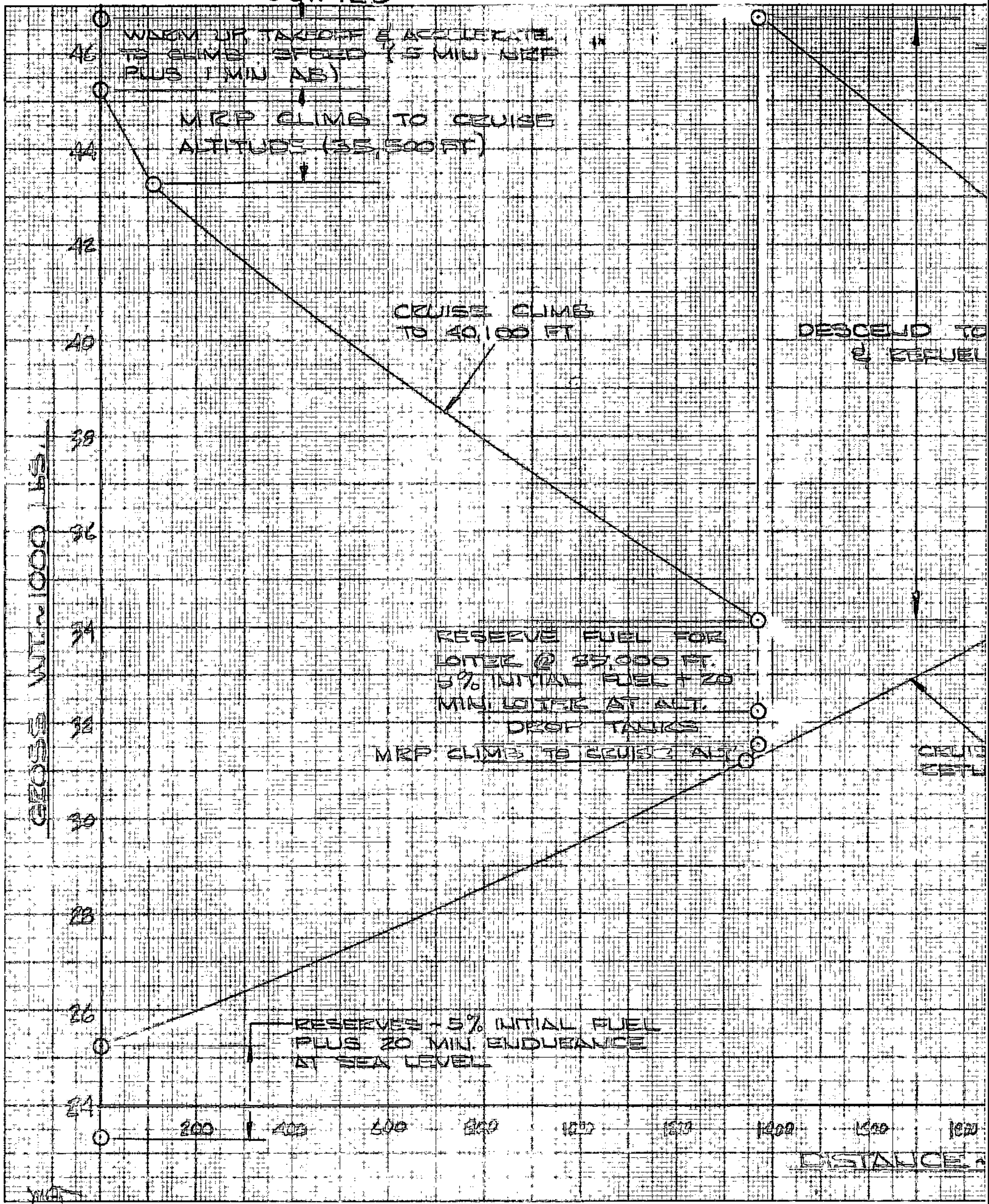
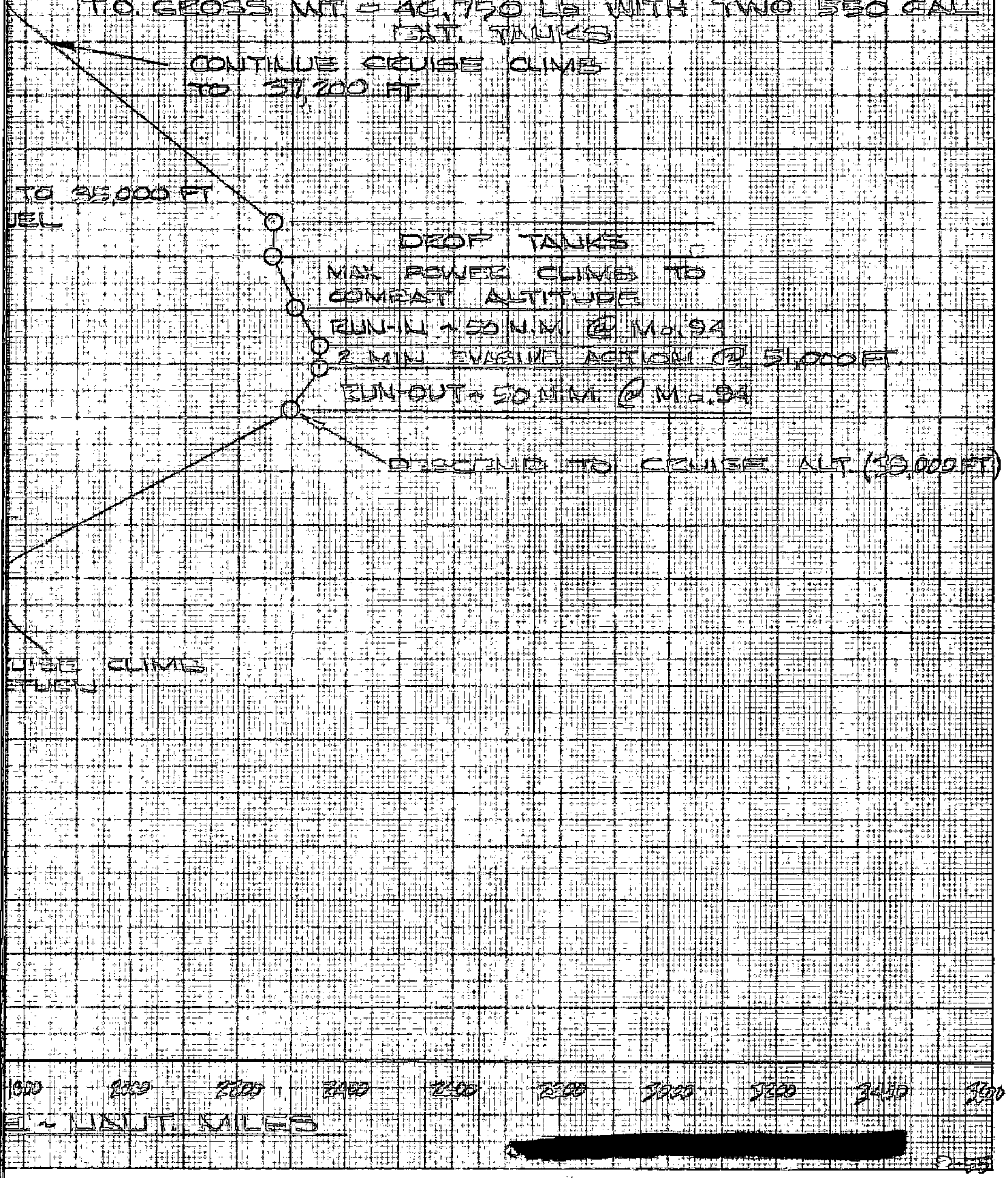


F-102B RECONNAISSANCE AIRPLANE
TWO STAGE RADIUS MISSION
GROSS WT. Vs DISTANCE

P & W J75-JT4A-24 TURBOJET (REF: P&W SPEC #2604)
T.O. GROSS WT – 46,750 LB WITH TWO 550 GAL EXT. TANKS



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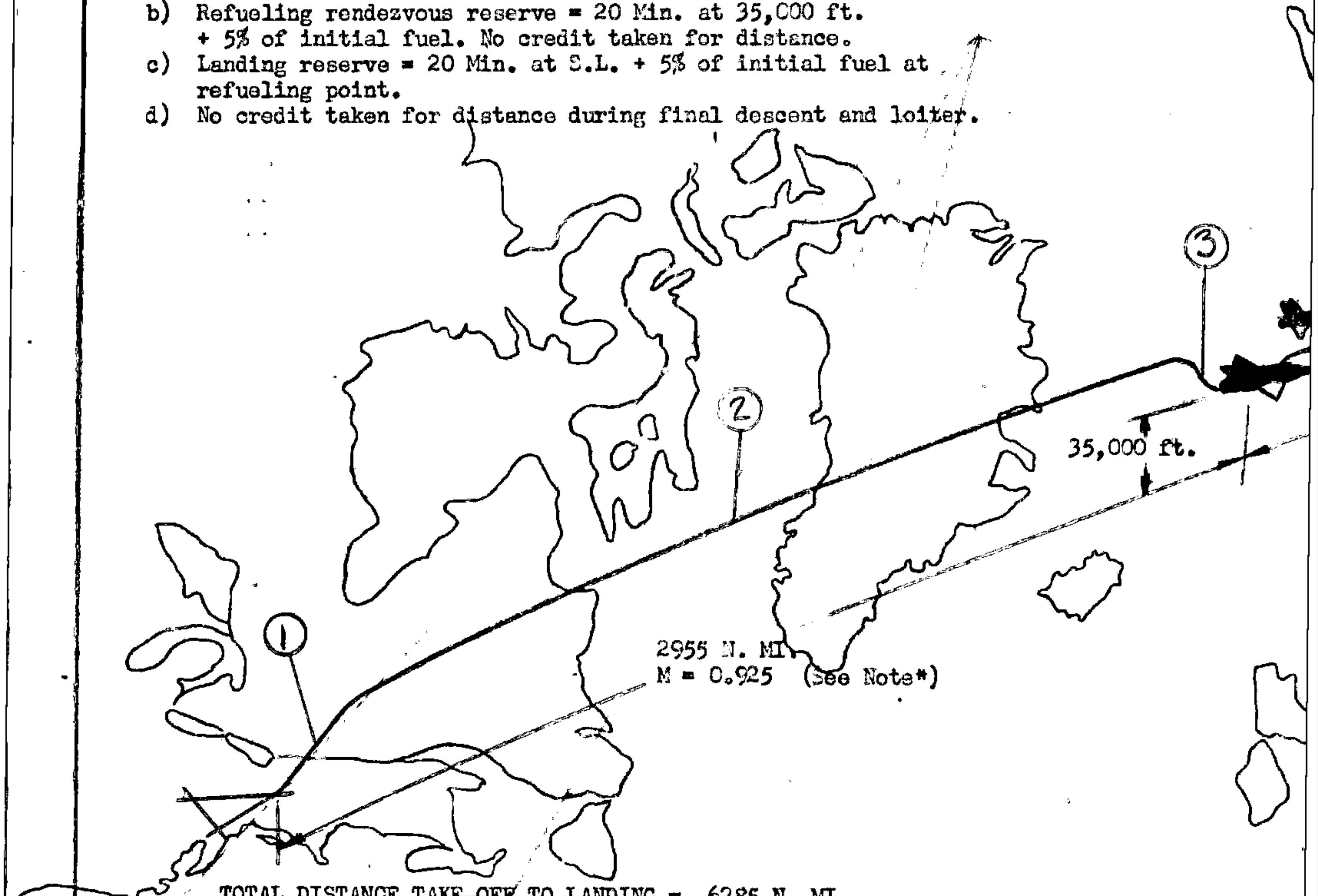
MISSION PROFILE - F-102B RECONNAISSA

TWO STAGE - LONG RANGE MISSION - ONE

WITH ALTERNATE BASE

TAKE-OFF GROSS WEIGHT 46,750 L

a) Warm up, take-off and acceleration allowance = 5 Min. at NRP + 1 Min. A.B.
b) Refueling rendezvous reserve = 20 Min. at 35,000 ft. + 5% of initial fuel. No credit taken for distance.
c) Landing reserve = 20 Min. at S.L. + 5% of initial fuel at refueling point.
d) No credit taken for distance during final descent and loiter.



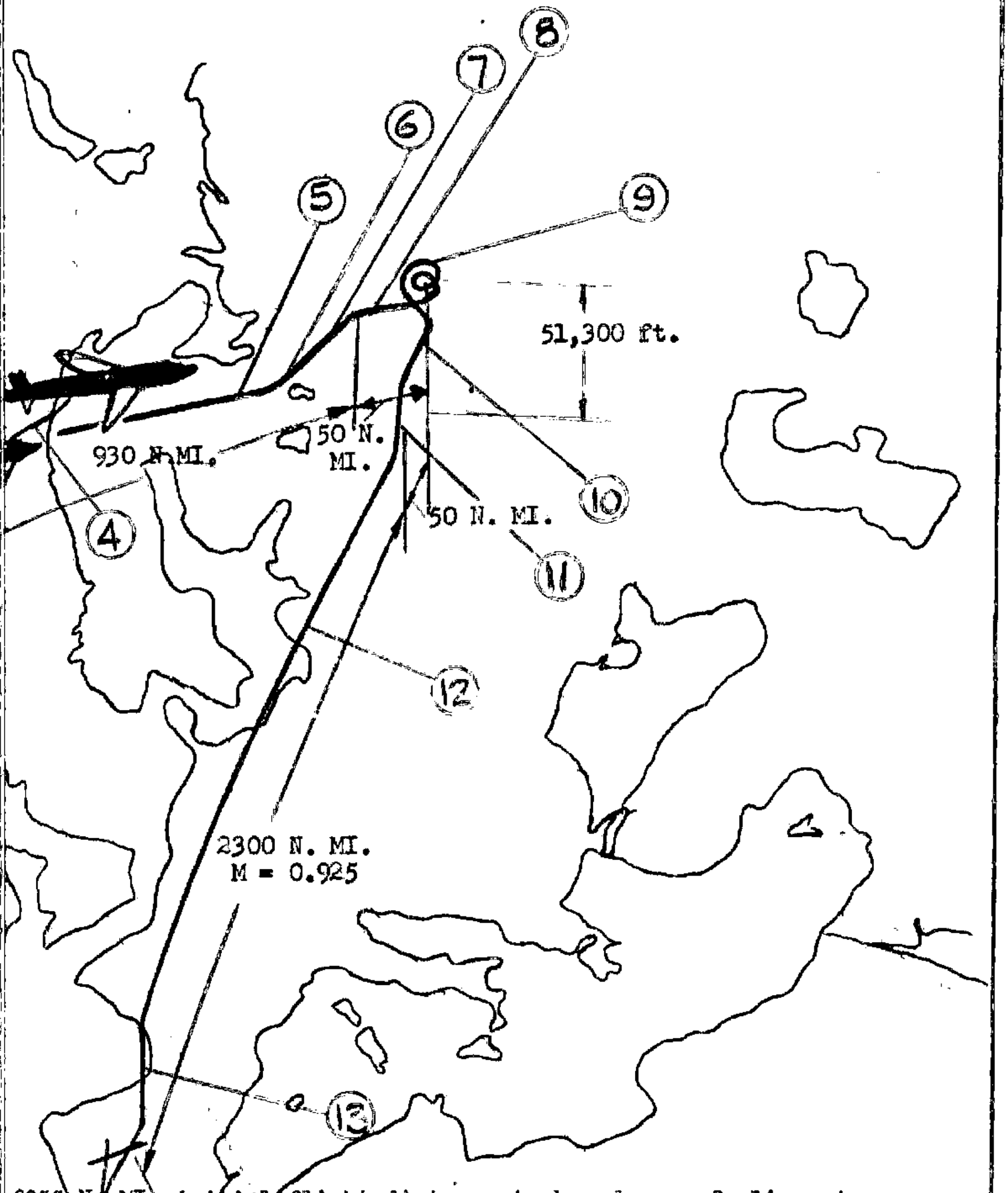
TOTAL DISTANCE TAKE-OFF TO LANDING = 6285 N. MI.

1. Climb at Mil. power to cruise altitude.
2. Cruise climb, at M = 0.925.
3. Descend to 35,000 ft. (refueling altitude).
4. Refuel to take-off gross weight (46,750 Lbs.)
5. Cruise climb, at M = 0.925.
6. Drop external tanks.
7. Climb at Mil. + A.B. power to max. power cruise ceiling.
8. Run-in 50 N. MI. at 0.94M at Mil. + A.B. power.
9. Two minutes evasive action over target.
10. Run-out 50 N. MI. at 0.94M at Mil. + A.B. power.
11. Descend to cruise altitude.
12. Cruise climb at M = 0.925.
13. Descend and loiter at S.L.

*Note: The 29
point where
20 Min. loit

This initia
made at the
loiter at 3
this operat

0 LBS.



2955 N. MI. initial flight dist nce is based on refueling at a re total fuel remaining in the airplane is 5% of the initial fuel plus oiter at 35,000 ft.

ial distance decreases to 1370 miles when the refueling contact is he "point of no return" and reserves of 5% initial fuel and 20 Min. 35,000 ft. retained. The total distance take-off to landing for ation would be 4700 N. MI. instead of 6285 N. MI.

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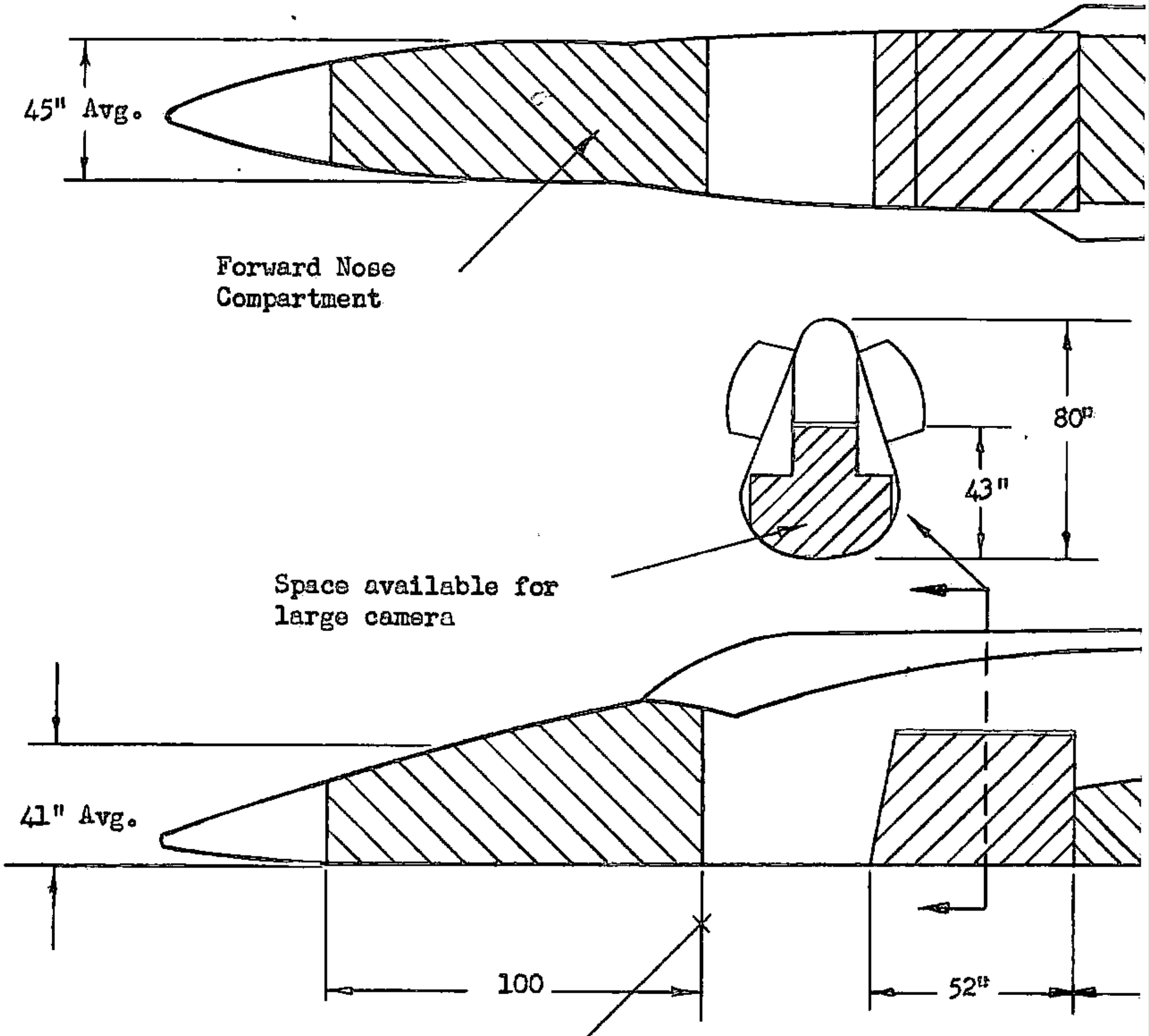

Forward Nose Compartment

Space available for large camera

Manufacturing Joint at this bulkhead

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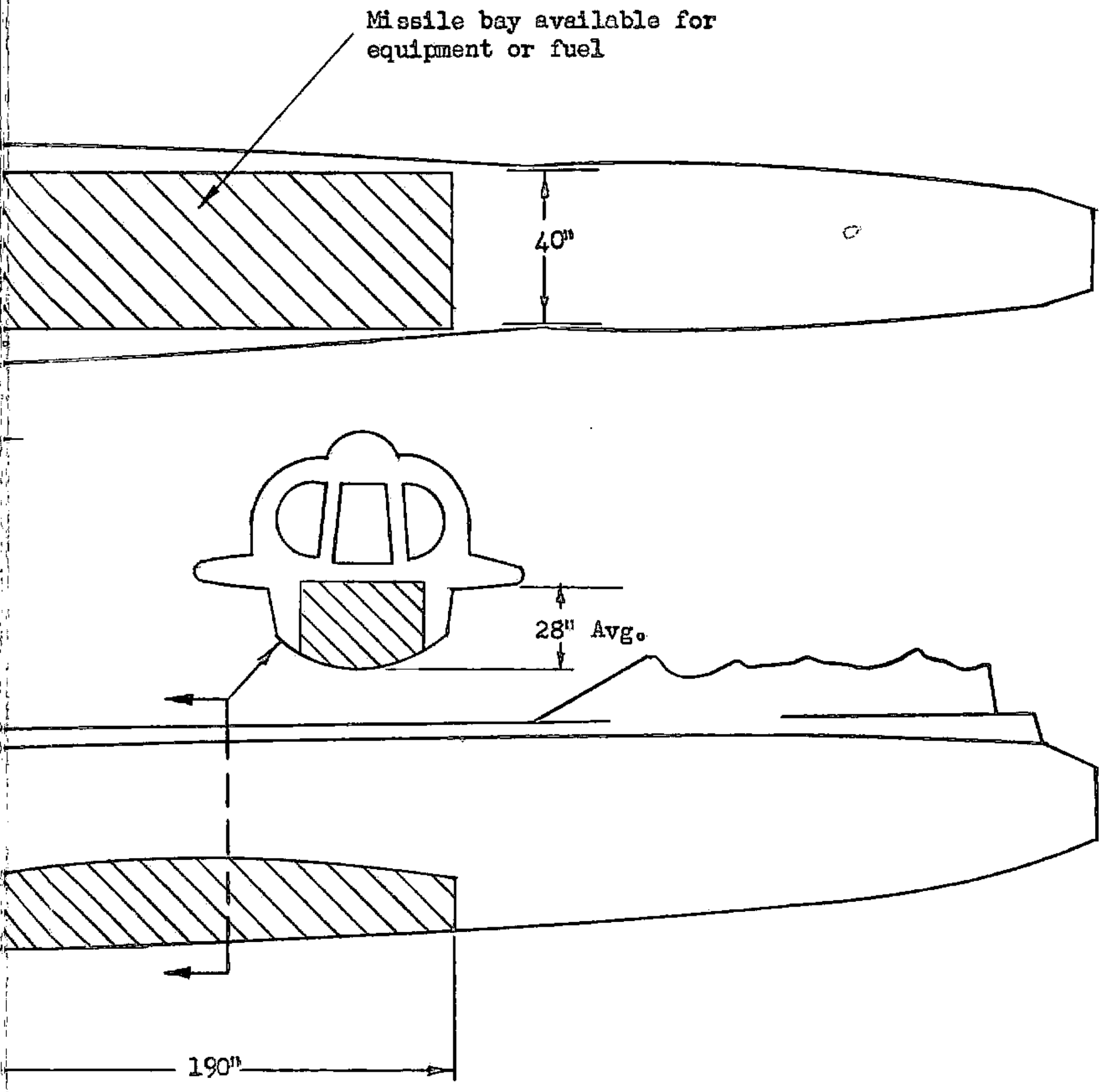

F-102B RECONNAISSANCE AIRPLANE SPACE DIAGRAM

Showing space available for various combinations and sizes of cameras

CONVAIR DRAWING
#SD 55 08084

ZP-8-037

Map To Report
Filed in Misc Tube

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CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION

SAN DIEGO

REPORT ZP-8-037
DATE 1 March 1956
MODEL 8

TITLE

STUDY
CARRIER BASED VERSION OF THE F-102B AIRPLANE (U)

ZP-8-037

PREPARED BY R. C. Lingley
R. C. Lingley
R. E. Craig
R. E. Craig, Aerodynamics

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A R Lambert
A. R. Lambert

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INTRODUCTION

The purpose of this report is to familiarize Navy personnel with the Air Force F-102A and F-102B airplanes and to present an investigation of the feasibility of adapting the F-102B airplane to carrier-based operations.

The report is in three sections:

I. Design and performance information on both the F-102A and F-102B airplanes.

II. A study of the Air Force version of the F-102B with minimum modifications for carrier suitability. Complete performance is included in this version.

III. A brief study of a Navy version of the F-102B incorporating changes to the electronic and weapon systems for internal stowage of Sparrow missiles. No performance is included in this section. For a more complete study of this airplane (including performance), see Convair secret report No. ZP-55-13006 dated 20 June, 1955. (Performance shown under Item II above may be used for approximation.)

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SUMMARY

This study indicates that a carrier-based version of the F-102B airplane is entirely practical and would provide the Navy with a Mach 2.0 airplane capable of attaining an altitude of 52,000 feet. By use of the "zoom technique", a usable altitude of 65,000 feet can be attained for sufficient time to make an attack. For information on the "zoom technique", see Convair's Secret Report No. ZO-P-020.

Since the F-102B is scheduled for early production (June 1957), it would assure delivery of Navy versions at an earlier date than attainable with a completely new design. The proven aerodynamic configuration of the F-102 airplanes would eliminate the costly delay in designing and testing a new design with unknown performance and systems characteristics.

The outstanding success of the F-102A airplane as a stable platform for launching six (6) internally stowed Falcon missiles and rockets cannot be over emphasized. Salvos of six (6) missiles and twenty-four (24) rockets have been fired at speeds up to Mach 1.20 and altitudes up to 53,000 feet. Rockets have been fired as low as 300 feet above the terrain and missiles as low as 500 feet. Both the 2" (T-214) rockets and the 2.75" FFAR rockets have been fired. In all of these tests not a single engine malfunction has occurred over a period of two years.

A package of 12 rockets has been designed and tested that mounts on the missile launchers. With six (6) of these packages, plus the twenty-four (24) rockets in the missile bay doors, a total of 96 2.75" rockets can be carried internally for an all-rocket configuration.

Successful electronically controlled attacks have been made using the Falcon guided missiles against QB-17 drones up to 28,500 feet. To obtain higher target altitudes it was necessary to use balloons. In a typical snap-up attack, a single Falcon was fired at 45,000 feet and cut the shroud lines of a balloon at 52,000 feet.

This study shows that the Navy version of the F-102B could take off from a carrier without afterburner with a full load of armament and internal fuel. However, it would be necessary to use afterburner power for take-off with external tanks in order to take full advantage of the airplane mission capability.

[redacted]

To improve landing visibility, the nose of the carrier version is modified to provide 20 degrees of down vision for the approach and landing condition. One of the outstanding features of the delta wing F-102 is the stability during slow speed approach attitudes. Reasonable landing speeds for the carrier approach condition are obtainable as shown in the performance section.

The Navy fighter mission specifies a "three-hour cycle" time including cruise out 150 nautical miles, combat twenty minutes (15 military + 5 afterburner) and cruise back. The carrier-based version of the F-102B with external tanks meets this requirement with a cycle time of two hours. By trading combat time for cycle time, it would be possible to obtain the "three-hour cycle" time with three minutes of combat time which is equivalent to a typical interceptor mission.

The F-102B airplane carries a mixed load consisting of four (4) Falcon guided missiles and one (1) MB-1 weapon. All armament is stowed internally. Alternate loads of six (6) Falcon missiles or two MB-1 weapons can be carried. The MA-1 fire control system is used.

With minor modification to the missile bay area, four Sparrow 1-A missiles can be carried. With this armament, an APQ-50 fire control system replaces the present MA-1 system.

In all of the configurations, two MB-1 weapons can be carried internally. These weapons are armed with an atomic warhead.

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SECTION I.

EXISTING F-102A & F-102B AIRPLANES

[redacted]

[illegible] INTERCEPTORS

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SECTION I.

DISCUSSION

This section presents design and performance information on the F-102A and F-102B airplanes. It is included in this report for reference to familiarize Navy personnel with the capabilities of these Air Force interceptors.

The purpose of these airplanes is to provide the Air Defense Command with supersonic weapons capable of all-weather interception of high speed targets.

The F-102A is armed with six internally stowed Falcon guided missiles with either a radar or infra-red target seeker in the nose. They are automatically fired by the Hughes MG-10 electronic system. The airplane is powered by a Pratt & Whitney J-57 engine and is capable of level flight speeds of Mach 1.18 at 35,000 feet and Mach 1.0 at 50,000 feet. Its combat ceiling is 53,000 feet. The first complete F-102A squadron will be activated in June of this year (1956).

The F-102B airplane is now in the design stage. Mockup inspection was completed in December of 1955. The first prototype will fly in December of this year (1956). First production units are scheduled for June, 1957.

The F-102B airplane is powered by a Pratt & Whitney J-75 engine and is capable of speeds up to Mach 2.0 (temperature limit) from 35,000 feet to 40,000 feet and Mach 1.79 at 50,000 feet. Its combat ceiling is 54,000 feet but it is capable of "zooming" to 74,000 feet before reaching the minimum speed limit established by engine flame-out. The maximum useful altitude is limited to 65,000 feet by equipment limits. At this altitude, it is possible to maintain level flight with a 1.2 g maneuvering capability for sufficient time to make an attack. For more detail information on the "zoom technique", see Convair Secret Report No. ZO-P-020.

This airplane is armed with a mixed load. It includes four Falcon guided missiles and one MB-1 weapon with an atomic warhead. These weapons are all stowed internally. Alternate loads include six Falcons or two MB-1 weapons.

[redacted]

All of the electronic and control functions required for the F-102B interceptor are performed by the Hughes MA-1 aircraft and Weapon Control System. A typical mission may be described as follows:

Immediately after take-off, the pilot turns control over to the MA-1 system which automatically navigates and flies the airplane out to meet the target.

The system flies on a collision course and automatically fires the armament pre-selected by the pilot and destroys the target.

The airplane is then automatically navigated to the return base area (any one of 40 destinations selected by the pilot) and flown down the landing approach course. The pilot takes over and lands the airplane.

The difference between the F-102A and F-102B airplanes is mainly in the internal changes such as electronics, engine and weapon arrangement. Externally, the most noticeable difference is in the aft location of the inlet ducts on the F-102B closer to the leading edge of the wing. This was done to save duct weight and to provide actuation space for a variable ramp inlet which improves the engine performance at high speed. Another external difference is the large fin on the F-102B necessary for the higher mach range.

In both airplanes the fuel is carried in integral wing tanks. However, on the F-102B a 233-gallon fuselage tank is incorporated just forward of the inlet ducts to provide for fuel transfer in order to move the airplane center of gravity aft during supersonic flight.

Since the aerodynamic center of pressure moves aft during acceleration from subsonic to supersonic speeds, the stability margins are increased resulting in high trim drag increments. Moving the airplane's center of gravity aft (by means of the fuel transfer system) reduces the stability margin, resulting in a decrease in trim drag.

The average center of gravity shift on the F-102B is approximately 6% and for each 1% of center of gravity shift, approximately 400 feet are gained in altitude for a given mach number.

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This fuel may be transferred forward again if the airplane is decelerated to subsonic speed.

On the following pages are weight and balance summaries, performance summary and curves, mission profiles and three-view and inboard arrangement drawings of the F-102A and F-102B airplanes.

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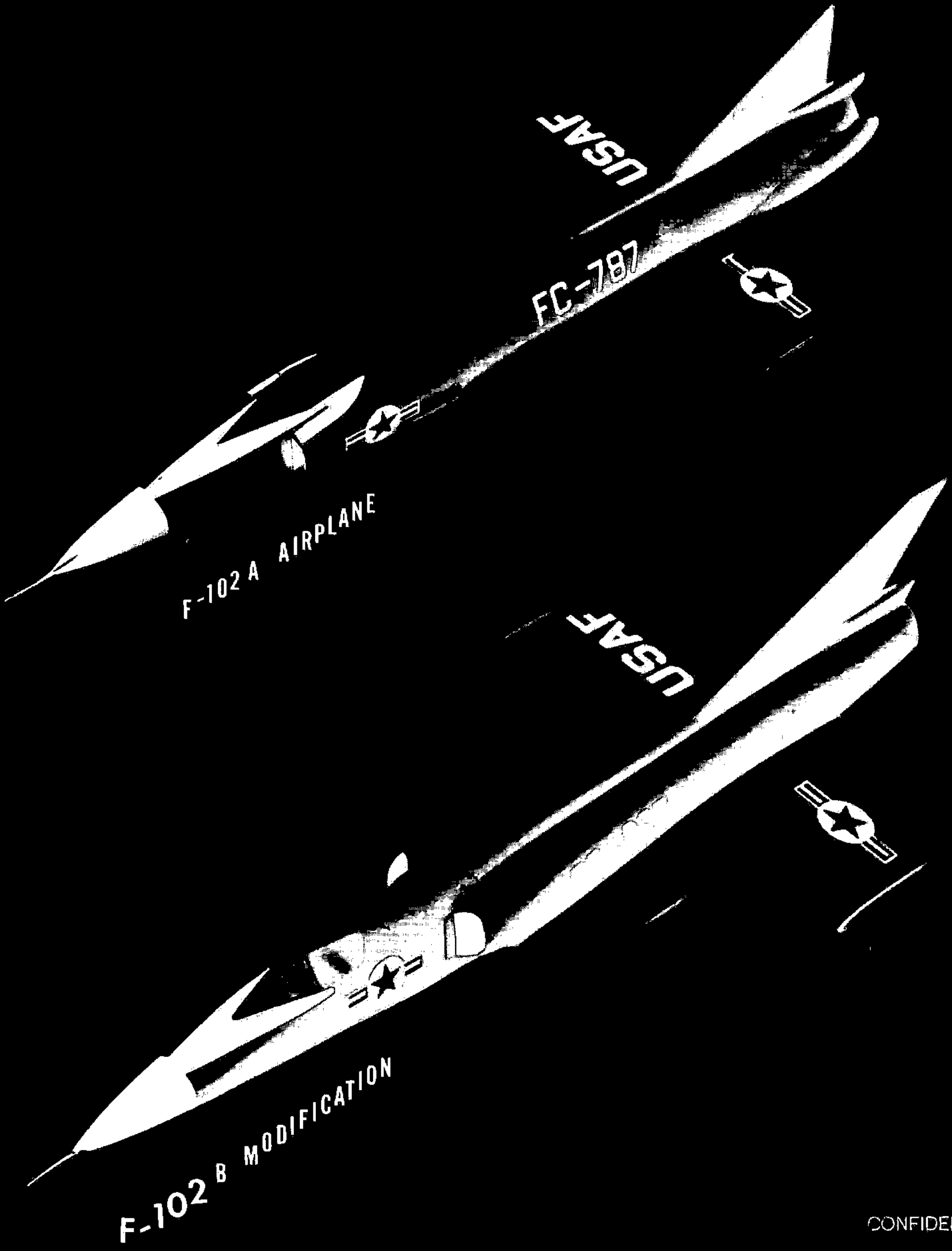

F-102 A AIRPLANE

F-102 B MODIFICATION

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GROUP WEIGHT STATEMENT
F-102A INTERCEPTOR

	F-102A* Interceptor	
Wing Group	3050	
Tail Group	438	
Body Group	3541	
Alighting Gear Group	1045	
Surface Controls Group	441	
Engine Section	39	
Propulsion Group	(6445)	
Engine Installation	5015	
Air Induction System	724	
Exhaust System	133	
Cooling System	17	
Lubricating System	27	
Fuel System	417	
Engine Controls	25	
Starting System	87	
Fixed Equipment Group	(4347)	
Instruments and Navigational Equipment Group	143	
Hydraulic and Pneumatic Group	319	
Electrical Group	575	
Electronics Group	2192	
Armament Group	617	
Furnishings and Equipment Group	229	
Air Conditioning and Anti-Icing Group	224	
Auxiliary Gear Group	48	
Unaccountable Weight	- 101	
WEIGHT EMPTY		19245
Useful Load		8639
Pilot, Survival Kit and Parachute	248	
Fuel-Unusable (JP-4)	154	
Fuel-Usuable (JP-4) at 6.5#/Gal.	6825	
Oil-Trapped and Engine	46	
Missiles 6	762	
Rockets 36-2.00"	382	
Missile Launchers	222	
NORMAL TAKEOFF GROSS WEIGHT		27884

* Based on the current status of Weight and Balance Status report No. 27, dated 1 February 1956.

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GROUP WEIGHT STATEMENTS

F-102B INTERCEPTOR *

Wing Group	3232	
Tail Group	634	
Body Group	4504	
Alighting Gear Group	1220	
Surface Controls Group	447	
Engine Section	37	
Propulsion Group	(8224)	
Engine Installation	6125	
Air Induction System	916	
Exhaust System	262	
Cooling System	45	
Lubricating System	52	
Fuel System	725	
Engine Controls	28	
Starting System	71	
Fixed Equipment Group	(4922)	
Instruments and Navigational Equipt. Group	121	
Hydraulic and Pneumatic Group	428	
Electrical Group	552	
Electronics Group	2493	
Armament Group	681	
Furnishings and Equipment Group	269	
Air Conditioning & Anti-Icing Group	320	
Auxiliary Gear Group	58	
WEIGHT EMPTY		23,220
Useful Load		9,980
Pilot, Survival Kit and Parachute	270	
Fuel-Unusable (JP-4) 36 Gals.	234	
Fuel-Usuable (JP-4) at 6.5#/Gal.	8060	
Oil-Trapped and Engine	60	
Rocket-Ding Dong (1) (MB-1 Weapon)	815	
Ding Dong Ejector Cartridge (5)	1	
Missiles - Falcon GAR-IA and/or -IC (4)	540	
NORMAL TAKFOFF GROSS WEIGHT		33,200

* Based on the current status of Monthly Weight and Balance Status Report No. 3, dated 25 January 1956.

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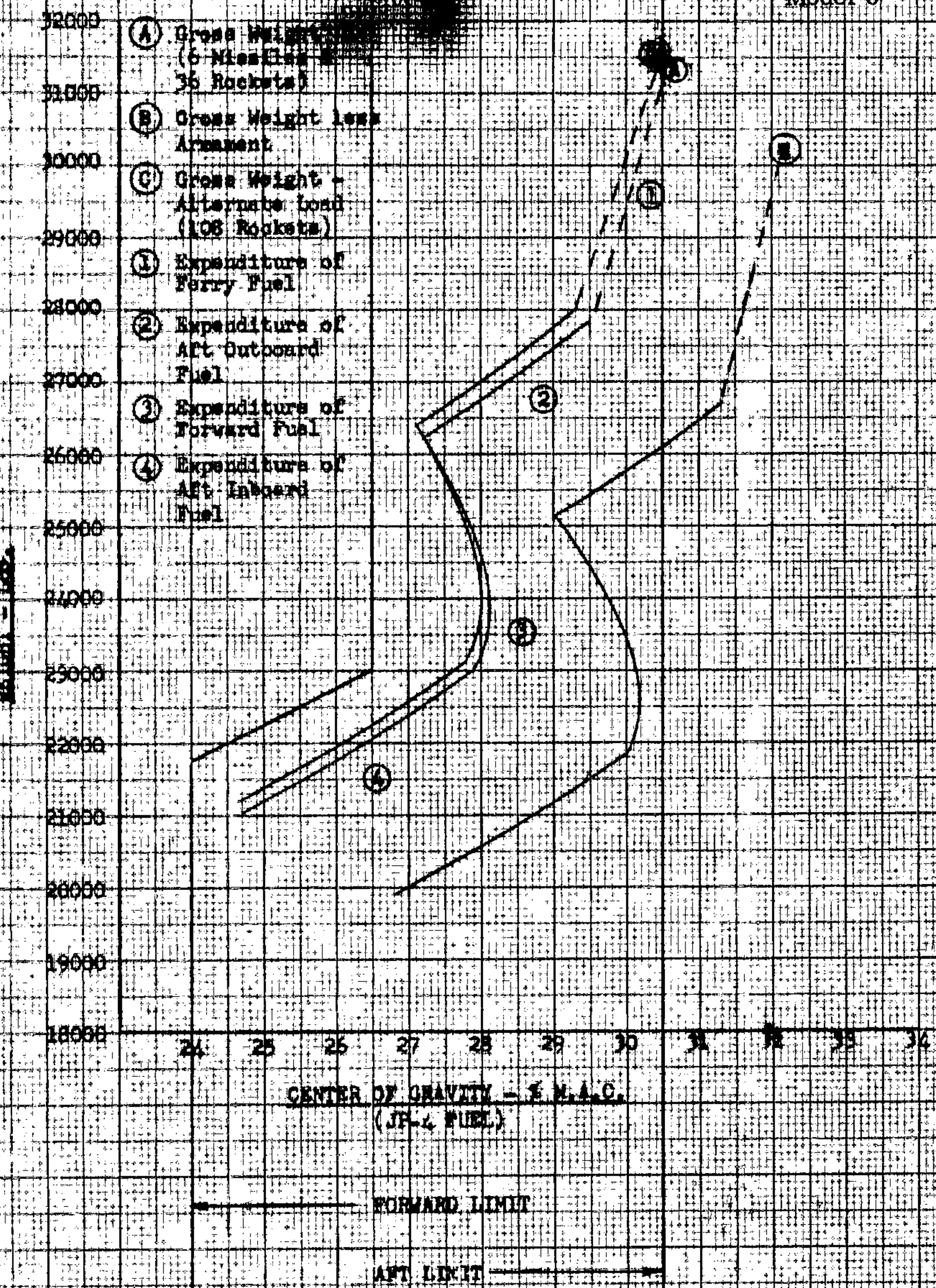
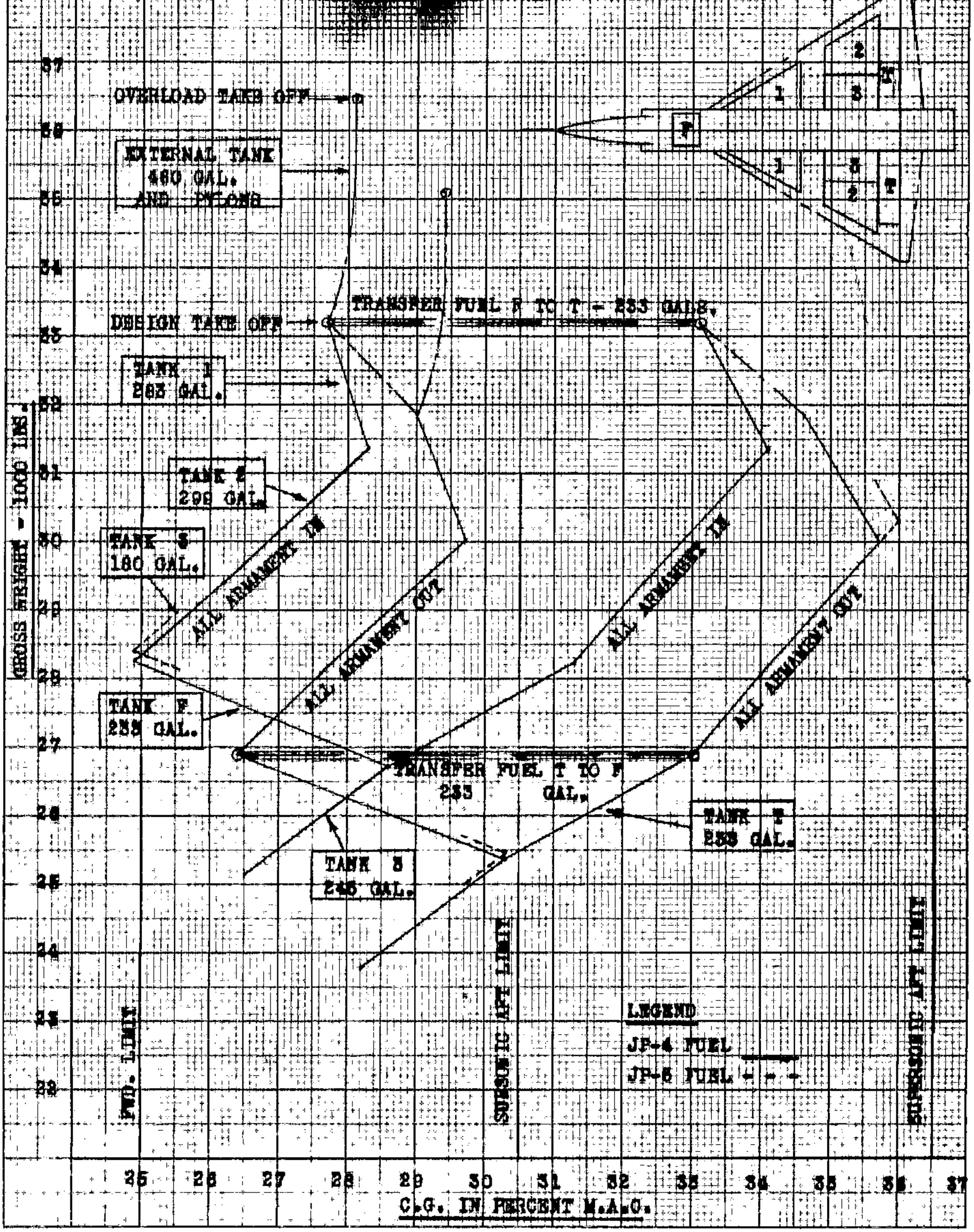
F-102A WEIGHT & BALANCE STATUS REPORT



K+E 10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO.
ALBANENE

F-102B WEIGHT & BALANCE STATUS REPORT
Dated 8 February 1956

[redacted]RAVITY ENVELOPE



K+E 10 X 10 TO THE ½ INCH
KEUFFEL & ESSER CO.
ALBANY, N.Y.
359T-11G

[redacted]

F-102A & F-102B

PERFORMANCE SUMMARY TABULATION

		F-102A	F-102B
Engine		P & W J57 - -P23	P & W J75 (JT4A-24)
Rated Static Sea Level Thrust (max.)	lbs.	16,000	23,500
T. O. Weight (full internal load)	lbs.	27,500 *	33,200
Combat Weight (a)	lbs.	24,867	30,370
Internal Fuel	lbs.	6,825	8,060
High Speed with Military plus Afterburner Power (b)			
35,000 Feet	Mach No.	1.18	2.00 (c)
50,000 Feet	Mach No.	1.00	1.79
Combat Ceiling with Military plus Afterburner Power (R/C= 500 ft./min. at Combat Wt.)	ft.	53,000	54,000
Time to Climb to Altitude with Military plus Afterburner Power (from start of take-off run)			
35,000 Feet	min.	3.7	2.5
50,000 Feet	min.	7.1	4.7
Combat Radius-Mil-C-5011A Area Intercept Mission (Combat 5 min. at max. Power at 50,000 feet)			
With internal fuel only	N. mi.	375	375
With 460 gal. ext. fuel	N. mi.	620	625

a) Weight at start of combat for Mil-C-5011A Point Intercept Mission.
b) Gross weight = weight upon reaching combat ceiling immediately following take-off without external tanks. (See plot given for carrier-based F-102B on page .)
c) Structural high speed of airplane.

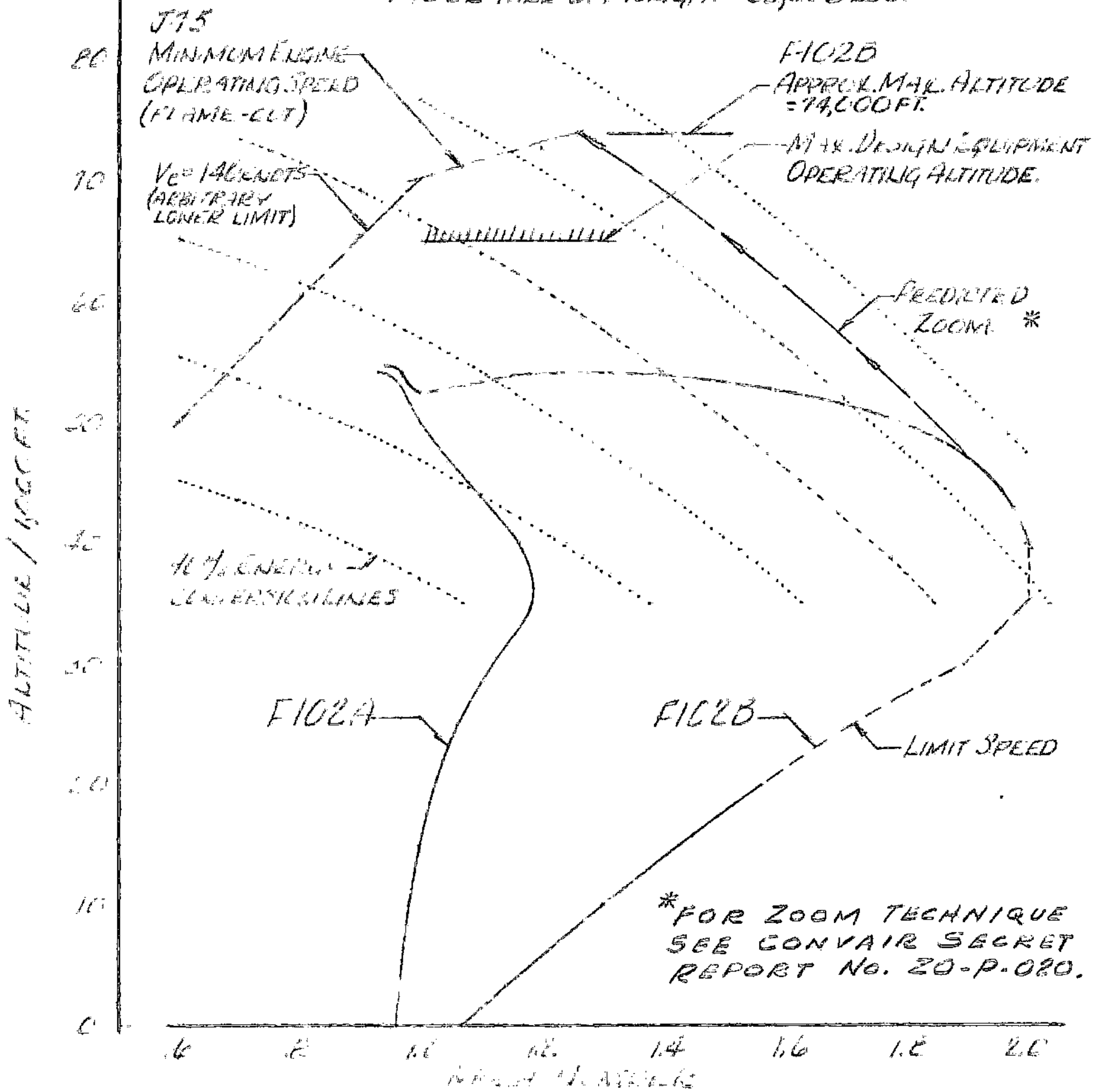
*Based on airplane spec. weight [redacted]

F102A & F102B

HIGH SPEED ENVELOPES

J-75 (JT4A 24) ENGINE, MILITARY + A.B. POWER

GROSS WEIGHT = WEIGHT UPON REACHING COMBAT CEILING
IMMEDIATELY FOLLOWING TAKE-OFF WITHOUT
EXTERNAL TANKS.
F102A TAKE-OFF WEIGHT = 27,500 LBS.
F102B TAKE-OFF WEIGHT = 33,200 LBS.



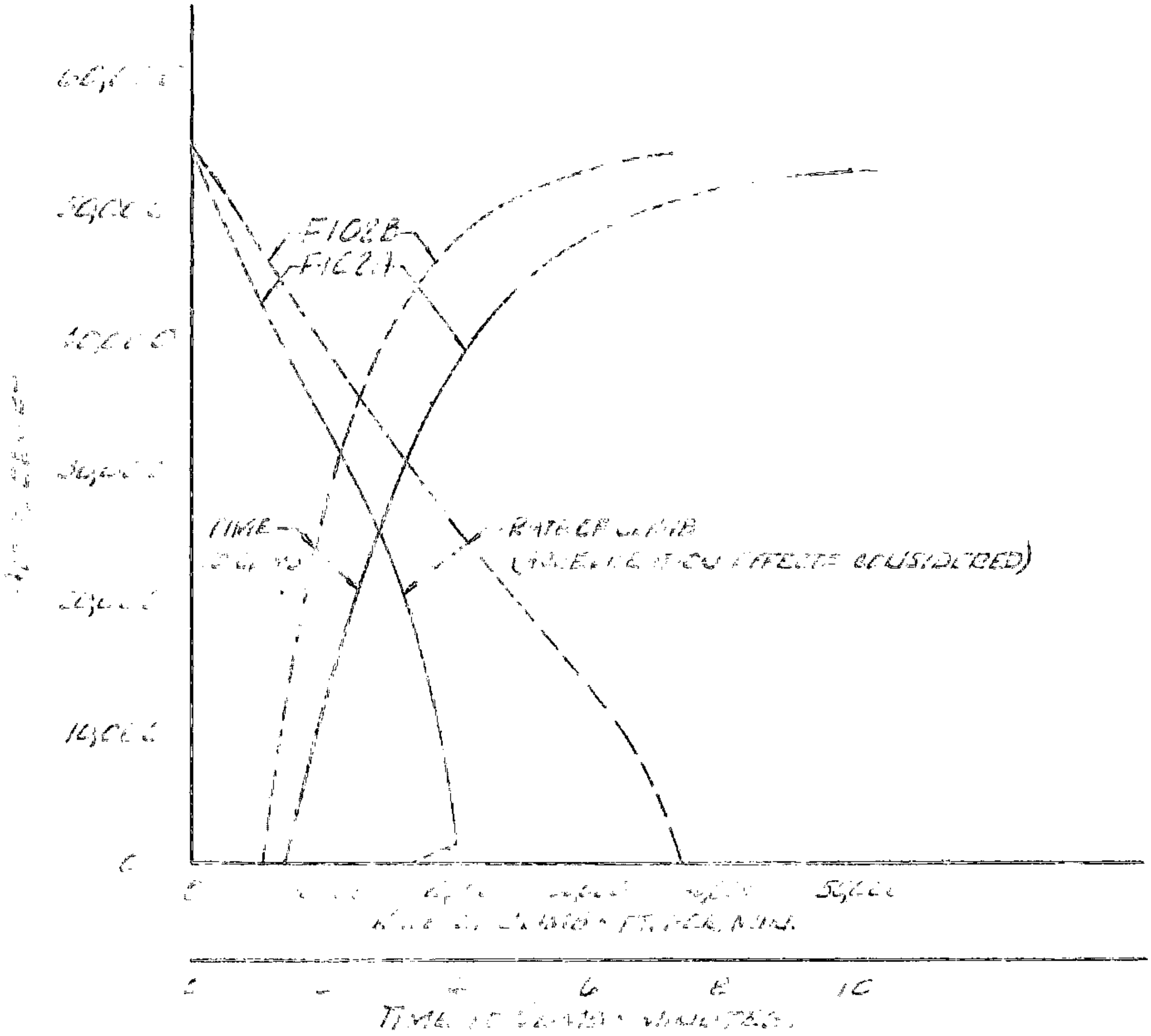
*FOR ZOOM TECHNIQUE SEE CONVAIR SECRET REPORT No. ZO-P-020.

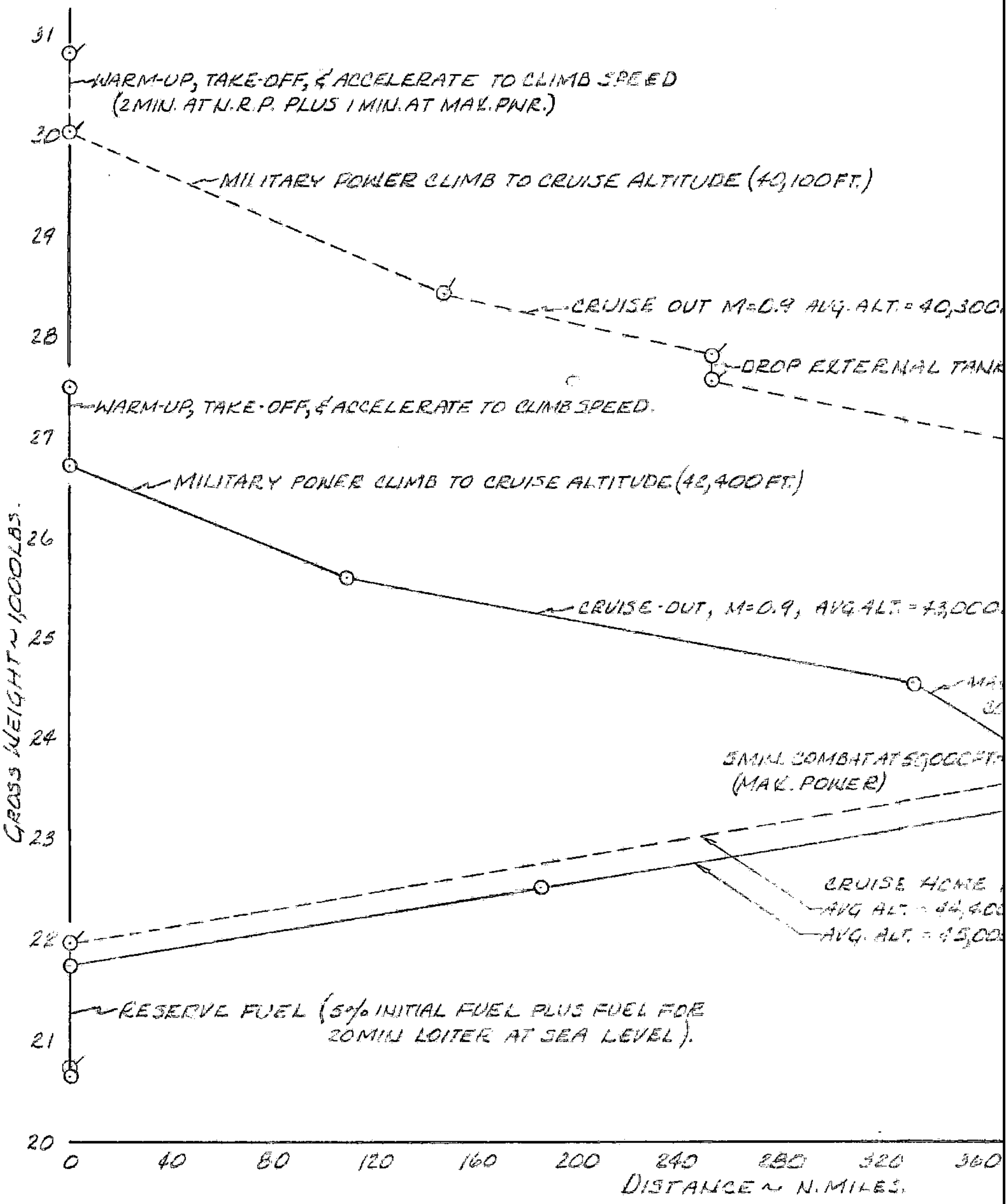
F102A & F102B

RATE OF CLIMB AND TIME TO CLIMB VERSUS ALTITUDE

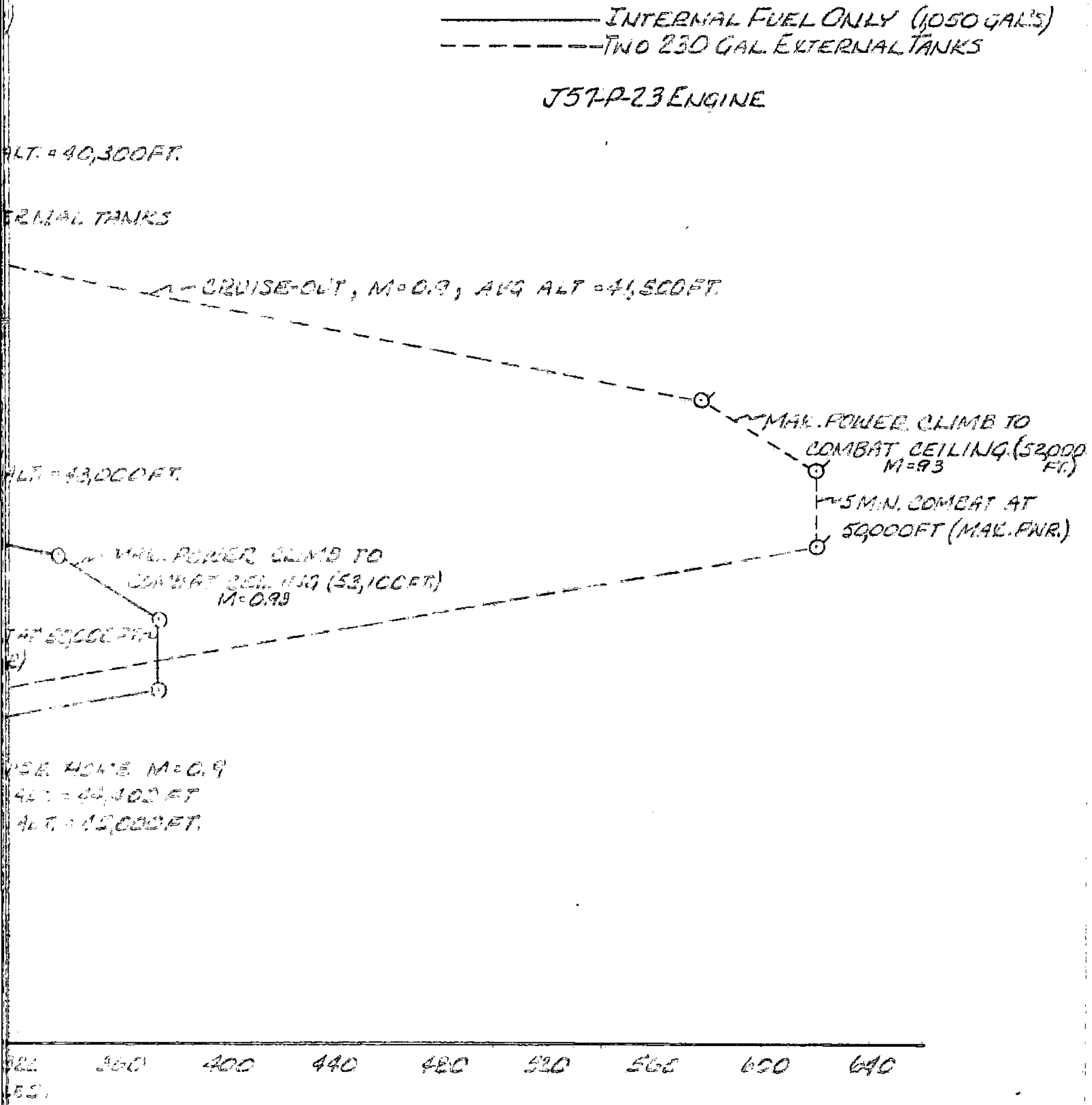
J75(JT4A-24) ENGINE, MILITARY + A.B. POWER

———— F102A, T.O. WT. = 27,500 LBS.
— — — F102B, T.O. WT. = 33,200 LBS.
WEIGHT REDUCTION CONSIDERED AS FUEL IS USED IN CLIMB.




31
30
29
28
27
26
25
24
23
22
21
20
WARM-UP, TAKE-OFF, & ACCELERATE TO CLIMB SPEED
(2 MIN. AT N.R.P. PLUS 1 MIN. AT MAX. PWR.)
MILITARY POWER CLIMB TO CRUISE ALTITUDE (40,100 FT.)
CRUISE OUT M=0.9 AVG. ALT.=40,300
DROP EXTERNAL TANK
WARM-UP, TAKE-OFF, & ACCELERATE TO CLIMB SPEED.
MILITARY POWER CLIMB TO CRUISE ALTITUDE (42,400 FT.)
CRUISE-OUT, M=0.9, AVG. ALT.=43,000
MA
5 MIN. COMBAT AT 50,000 FT.
(MAX. POWER)
CRUISE HOME
AVG. ALT. = 44,400
AVG. ALT. = 45,000
RESERVE FUEL (5% INITIAL FUEL PLUS FUEL FOR 20 MIN. LOITER AT SEA LEVEL).
GROSS WEIGHT ~ 1,000 LBS.
0
40
80
120
160
200
240
280
320
360
DISTANCE ~ N. MILES.

F102A AIRPLANE
MIL-C-5011A AREA INTERCEPT
MISSION SUMMARY



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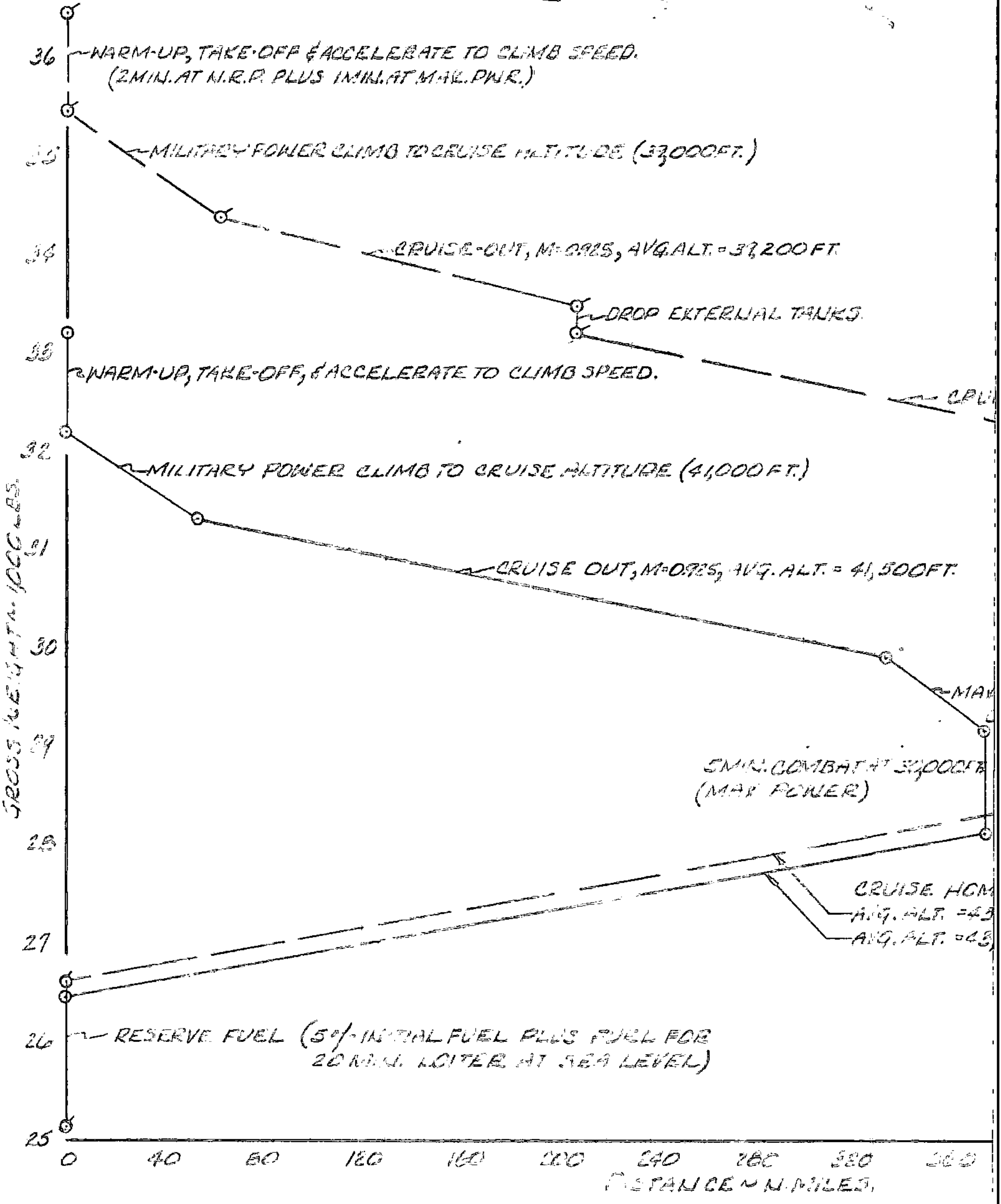
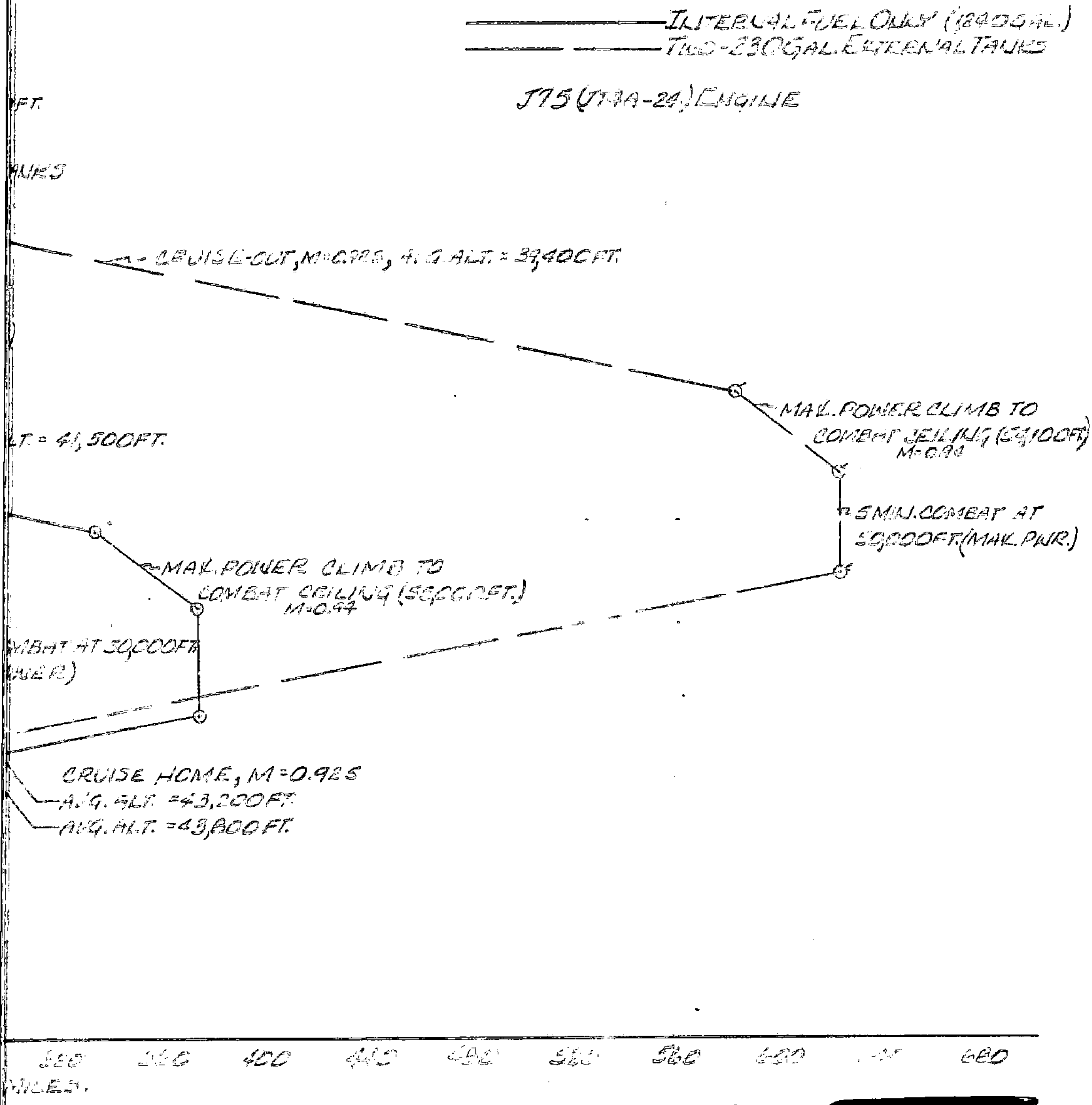


F-102B AIRPLANE

MIL-C-5011A AREA INTERCEPT

MISSION SUMMARY



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SECTION II.

A CARRIER-BASED VERSION
OF THE
AIR FORCE F-102B AIRPLANE

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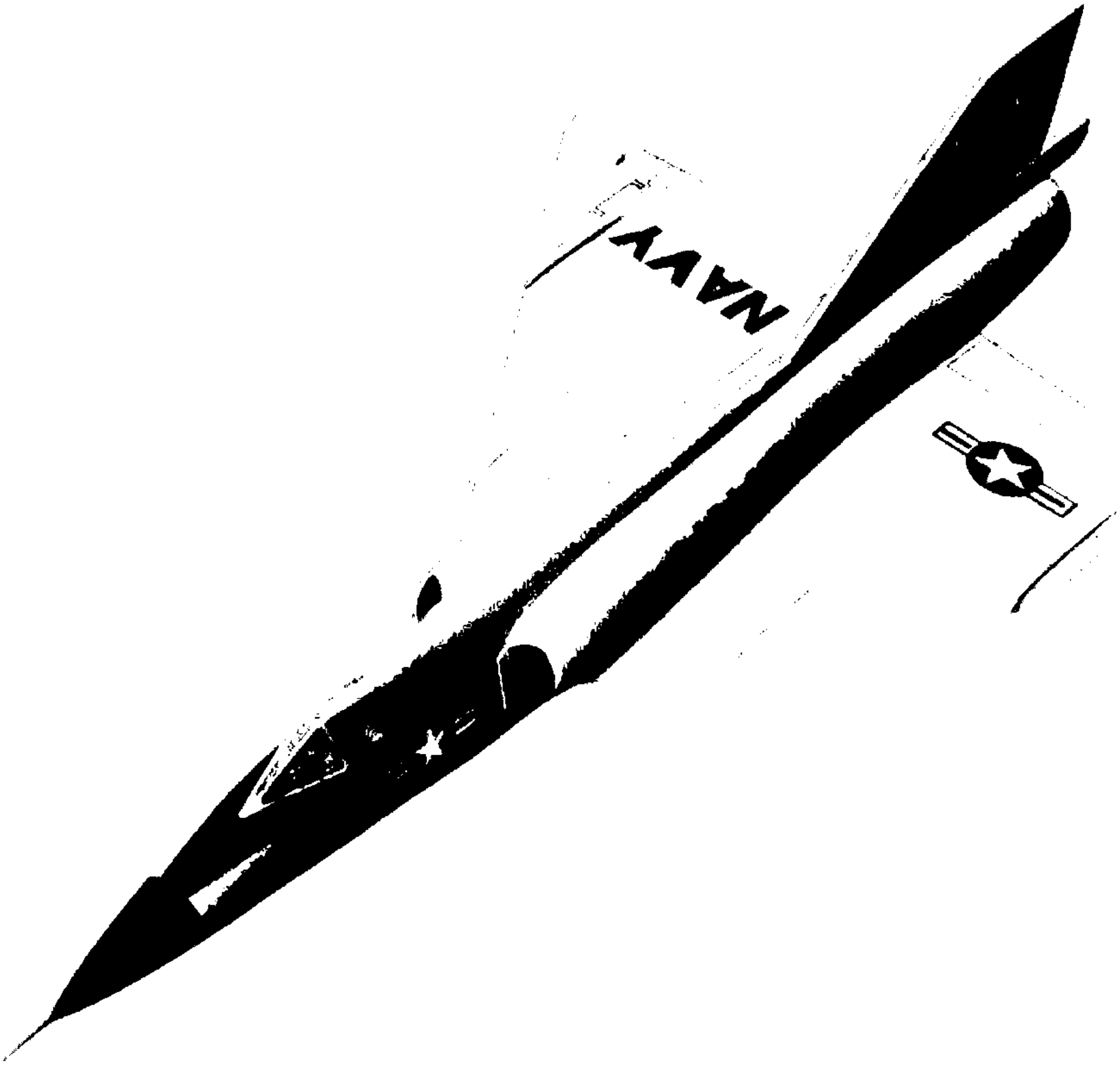

CARRIER BASED VERSION OF F-102B

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SECTION II.

DISCUSSION

This section presents a study of the modifications required to carrier-base the Air Force F-102B airplane.

For the purpose of this study, it was assumed that the F-102B electronics and armament would be retained.

The following changes necessary to meet the carrier requirements were made:

1. Increased stroke on nose and main landing gear.
2. Provisions for a full 360° swivel on the nose gear.
3. Addition of a tail bumper wheel and an arresting hook.
4. Addition of catapult bridal fittings and a hold back fitting.
5. Beef up to all structure to meet the increased demands of the carrier landing and catapult operations.
6. Addition of a 110-gallon fuselage tank adjacent to the existing fuselage transfer tank to increase the internal fuel capacity.
7. Addition of two 325-gallon external fuel tanks on the wings in lieu of the standard 230-gallon tanks on the F-102B.
8. Modification of the nose to provide 20° down vision for landing.
9. Incorporation of wing fold and fin fold to meet hangar deck clearance requirements.

In addition provisions were made for a retractable refueling probe on top of the fuselage immediately behind the canopy. The probe would be extended as shown on drawing SD-56-08028 and would be in an ideal position for the

[redacted]

pilot to fly the probe into the drogue of the refueling plane. The refueling probe is adjacent to the added fuselage tank facilitating the tie-in to the existing airplane fuel system.

The gross weight of this airplane without external tanks has increased from 33,200 pounds to 36,105 pounds. In this condition, it would be possible to catapult from the carrier without afterburner. The addition of external tanks will increase the gross weight to 40,700 pounds. In this condition, afterburner must be used for take-off. Although the airplane can take off without afterburner, the loss in mission capability without external tanks would scarcely warrant the elimination of afterburner noise to accommodate ground personnel.

In the Mil-C-5011A Area Intercept Mission the Air Force F-102B interceptor has a combat radius of 375 nautical miles. This mission requires 5 minutes of combat (military + afterburner power) at 50,000 feet. However, only 2 or 3 minutes of combat power are actually required to complete an attack.

The Navy requirement of 20 minutes of combat (15 military + 5 afterburner) at 40,000 feet altitude obviously would reduce the mission radius by a prohibitive amount. To maintain a respectable mission radius (or loiter time), it is necessary to include external fuel tanks. Two 325-gallon tanks have been added to the carrier-based version. This is the largest size tanks possible consistent with a maximum take-off weight of 40,700 pounds. (Catapult limitation with the J-75 engine + afterburner, 85° day, maximum angle of attack of 17-1/2°, a minimum acceleration of .075 g, and a 25-knot wind over the deck.)

A curve is included in this report showing "combat time" and "total mission time" versus "external fuel" after cruising out 150 nautical miles as required by the "three-hour cycle time" of the Navy mission. This curve graphically illustrates that the total mission time can be increased as the combat time is reduced. A "three-hour cycle time" can be achieved with 3 minutes of combat which is equivalent to a typical interceptor mission. However, it is possible to maintain the 20 minutes of combat desired with a cycle time of two hours.

The following pages include weight and balance, performance, general arrangement and inboard arrangement of the carrier-based version of the F-102B.

[redacted]

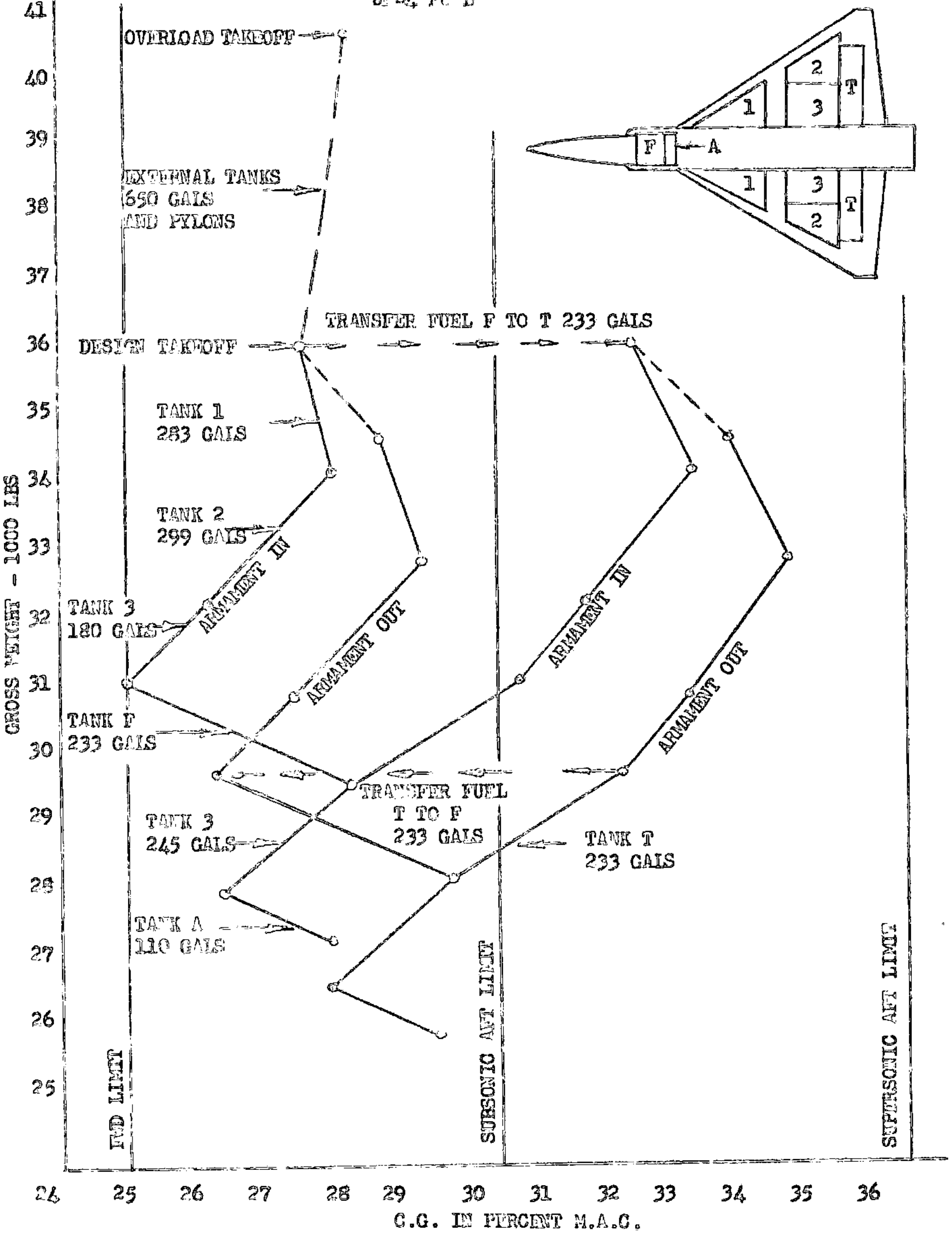
COMPARATIVE GROUP WEIGHT STATEMENTS

F-102B AND F-102B MODIFIED FOR CARRIER OPERATION

	*F-102B Interceptor	F-102B Carrier Version
Wing Group	3232	3884
Tail Group	634	659
Body Group	4504	4759
Alighting Gear Group	1220	2205
Surface Controls Group	447	447
Engine Section	37	37
Propulsion Group	(8224)	(8224)
Engine Installation	6125	6125
Air Induction System	916	916
Exhaust System	262	262
Cooling System	45	45
Lubricating System	52	52
Fuel System	725	725
Engine Controls	28	28
Starting System	71	71
Fixed Equipment Group	(4922)	(5195)
Instruments and Navigational Equipt.Grp.	121	121
Hydraulic and Pneumatic Group	428	448
Electrical Group	552	552
Electronics Group	2493	2493
Armament Group	681	681
Furnishings and Equipment Group	269	269
Air Conditioning & Anti-Icing Group	320	320
Auxiliary Gear Group	58	311
WEIGHT EMPTY	23,220	25,410
Useful Load	(9980)	(10,695)
Pilot, Survival Kit & Parachute	270	270
Fuel-Unusable (JP-4) 36 Gals.	234	234
Fuel-Usuable (JP-4) at 6.5#/Gal.	8060	8775
Oil-Trapped and Engine	60	60
Rocket-Ding Dong (1) MB-1 Weapon	815	815
Ding Dong Ejector Cartridges (5)	1	1
Missiles - Falcon GAR-IA and/or -IC (4)	540	540
NORMAL TAKEOFF GROSS WEIGHT	33,200	36,105

* Based on the current status of Monthly Weight and Balance Status Report No. 3, dated 25 January 1956.

F-102B CARRIER BASED AIRPLANE
CENTER OF GRAVITY ENVELOPE
(1+4 CONFIGURATION)
JP-4 FUEL



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CARRIER-BASED F-102 B

PERFORMANCE SUMMARY TABULATION

Engine		Pratt & Whitney J 75 (JT4A-24)
Rated static sea level thrust (max.)	lbs.	23,500
Take-off gross weight		
Without external fuel	lbs.	36,105
With two 325-gal. external tanks	lbs.	40,700
Combat Weight (a)	lbs.	33,120
Landing Weight (b)	lbs.	28,985
Internal Fuel	lbs.	8,775
External Fuel	lbs.	4,225
High speed with military + afterburner power (c)		
35,000 Feet	Mach no.	2.00
50,000 Feet	Mach no.	1.67
Combat ceiling with military + afterburner power (R/C = 500 Ft./min. at combat wt.)	ft.	52,000
Time to climb to altitude with military + afterburner power (from start of take-off run, without external tanks)		
35,000 Feet	min.	2.8
50,000 Feet	min.	5.6
Combat radius - Mil-C-5011A area intercept mission (combat 5 min. at max. power at 50,000 feet)		
With internal fuel only	N. Mi.	403
With 650 gal. external fuel	N. Mi.	705
Landing Speed - (min. airspeed with approach power at landing weight, see Page 34)	Knots	142

a) Weight at start of combat for Mil-C-5011A Point Intercept Mission take-off without external tanks. Corresponds to weight upon reaching combat ceil -

[redacted]

Carrier-Based F-102B
Performance Summary Tabulation
(Continued)

ing at M = 0. 94 immediately following take-off.

b) Landing weight corresponding to Mil-C-5011A Area Intercept Mission. Includes full armament, 5% initial fuel (external fuel included) and 20 minutes reserve fuel at sea level at speed for maximum endurance.

c) Gross Weight = Weight upon reaching combat ceiling immediately following take-off without external tanks (see plot on Page 27).

d) Structural high speed of airplane.

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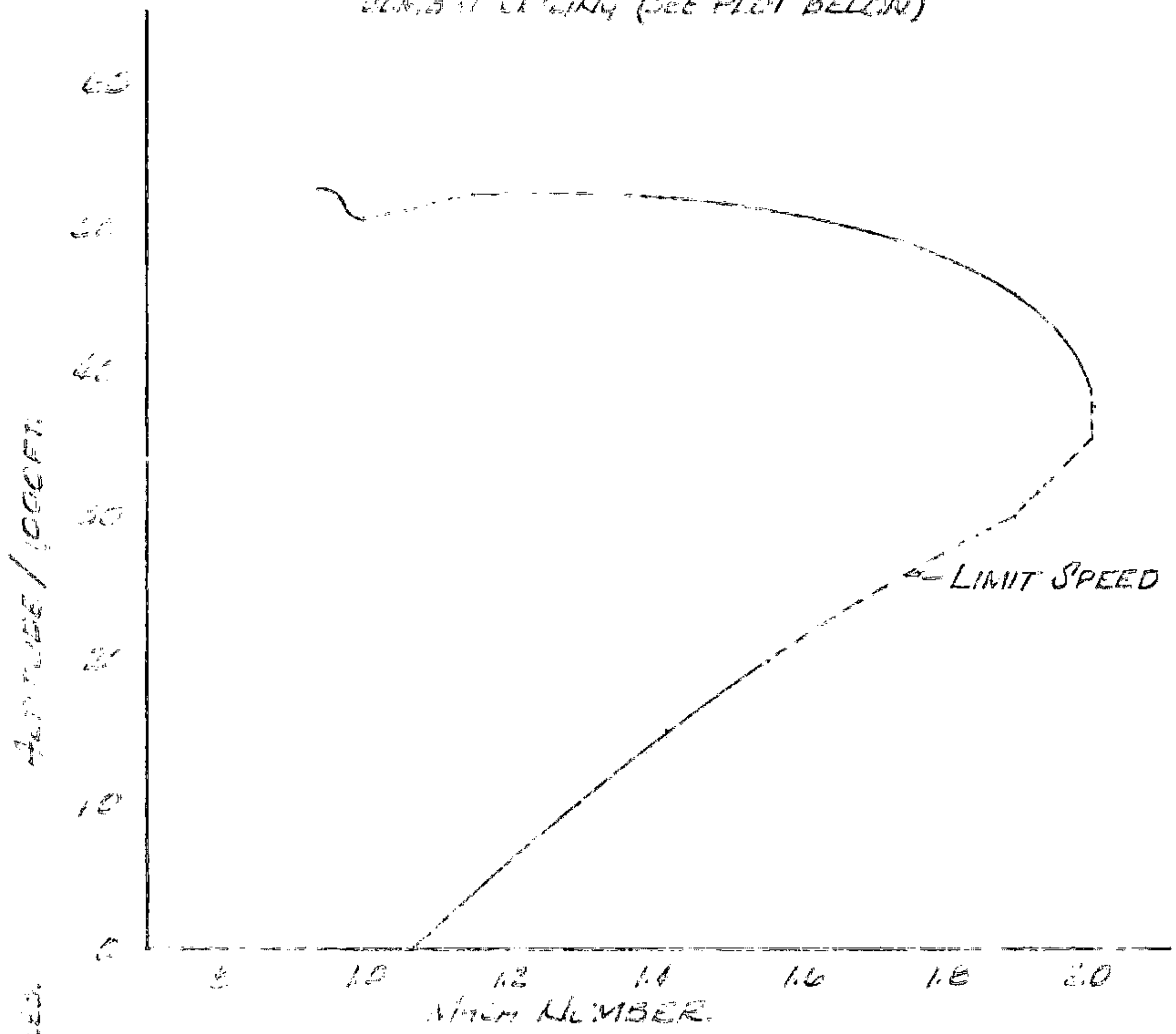
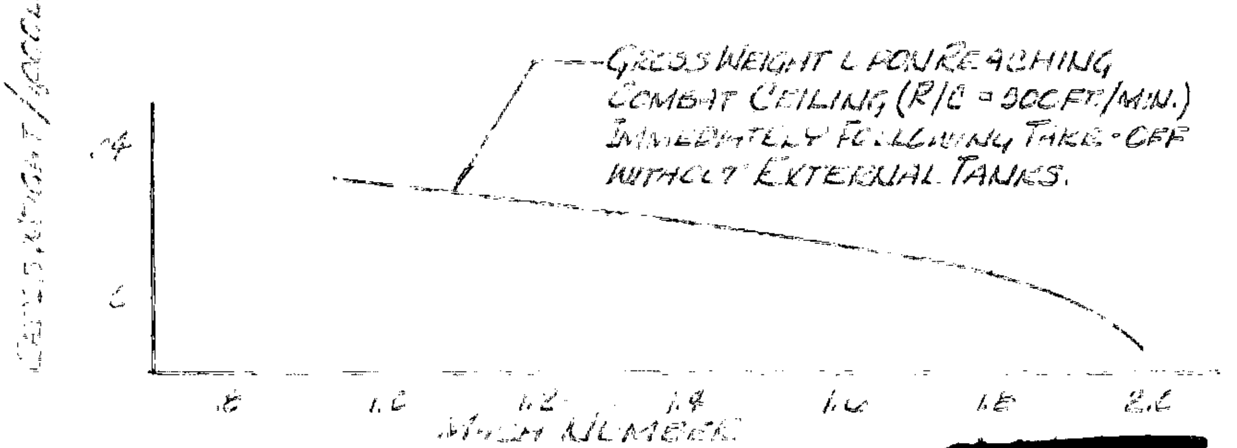
CARRIER BASED FIELD

HIGH SPEED ENVELOPE

J75 ([illegible]) ENGINE
MILITARY + A.B. POWER.

GROSS WEIGHT = WEIGHT UPON REACHING
COMBAT CEILING (SEE PLOT BELOW)



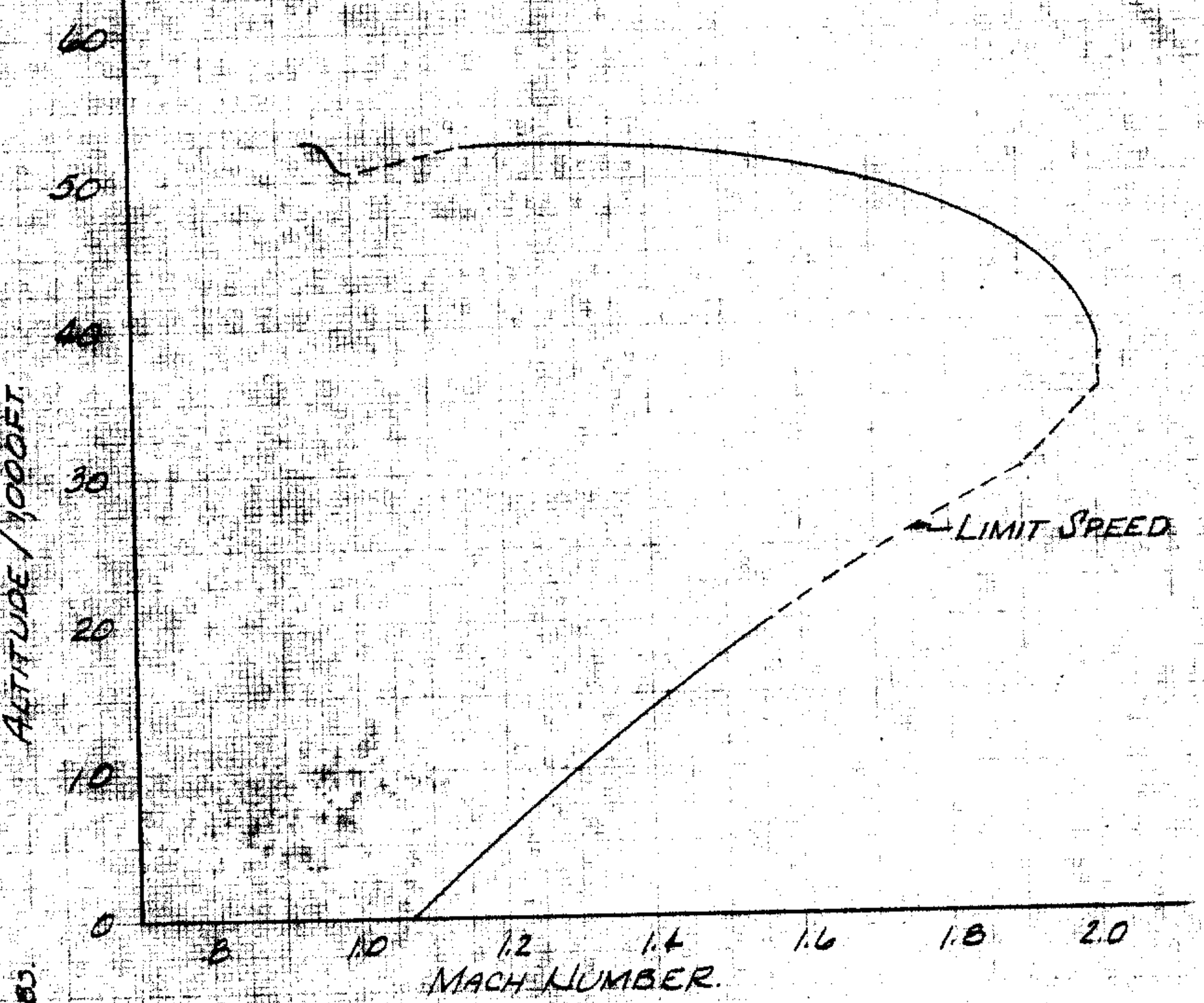


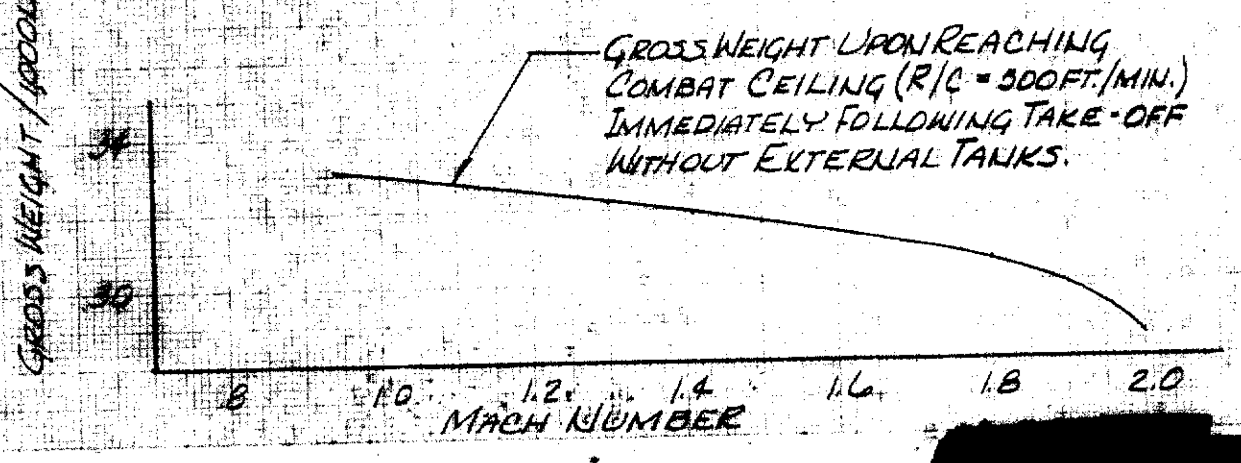
CARRIER BASED F102 B

HIGH SPEED ENVELOPE

TWO [illegible] ENGINE
MILITARY [illegible] POWER.

GROSS WEIGHT = WEIGHT UPON REACHING COMBAT CEILING (SEE PLOT BELOW)





GROSS WEIGHT UPON REACHING COMBAT CEILING (R/C = 500 FT./MIN.) IMMEDIATELY FOLLOWING TAKE-OFF WITHOUT EXTERNAL TANKS.

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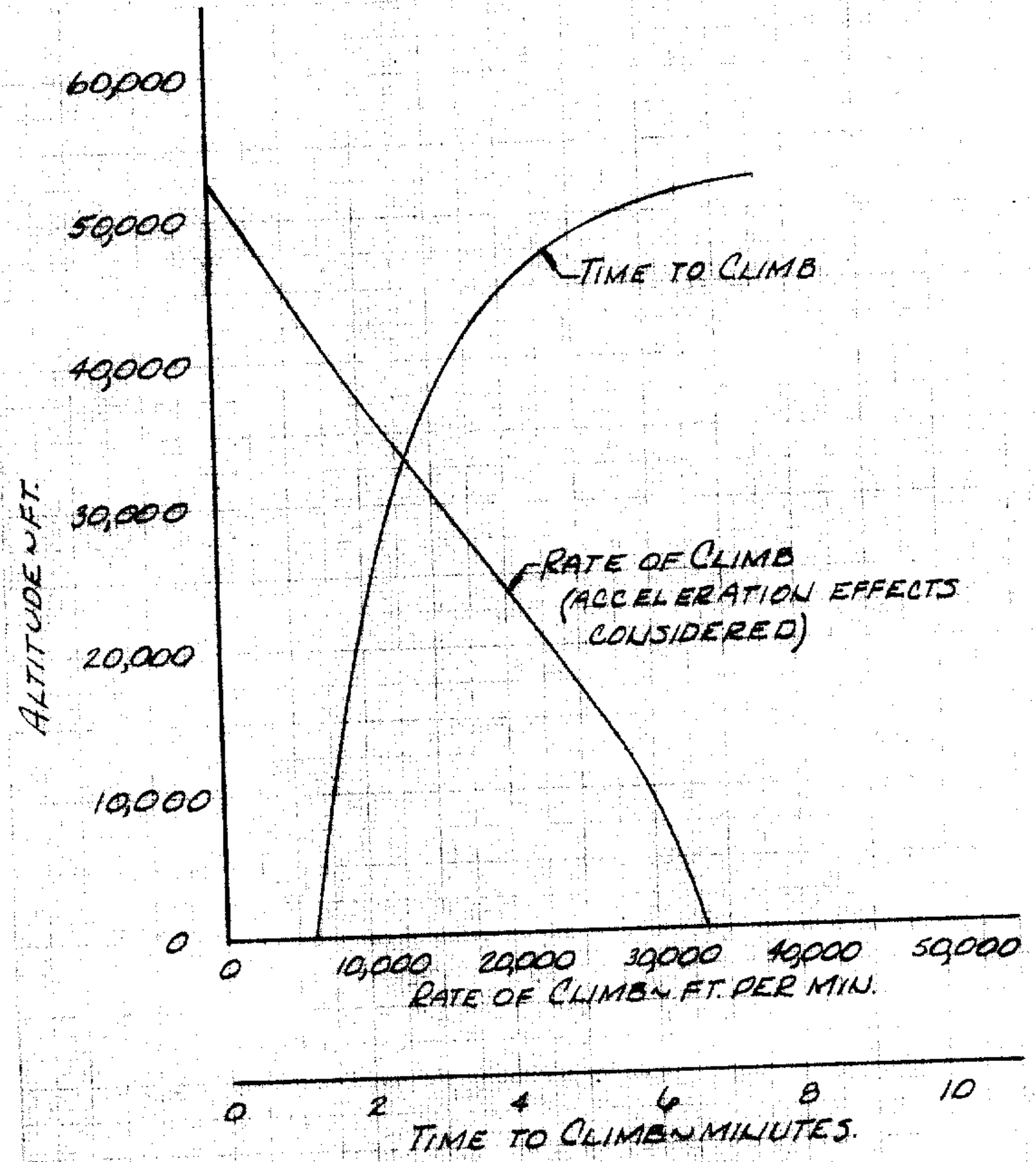
CARRIER BASED F-102B

RATE OF CLIMB AND TIME TO CLIMB
VERSUS
ALTITUDE

J75 (JT4A-24) ENGINE, MILITARY + A.B. POWER
TAKE-OFF WEIGHT = 36,105 LBS. (WITHOUT EXTERNAL TANKS)
WEIGHT REDUCTION CONSIDERED AS FUEL IS USED IN CLIMB.



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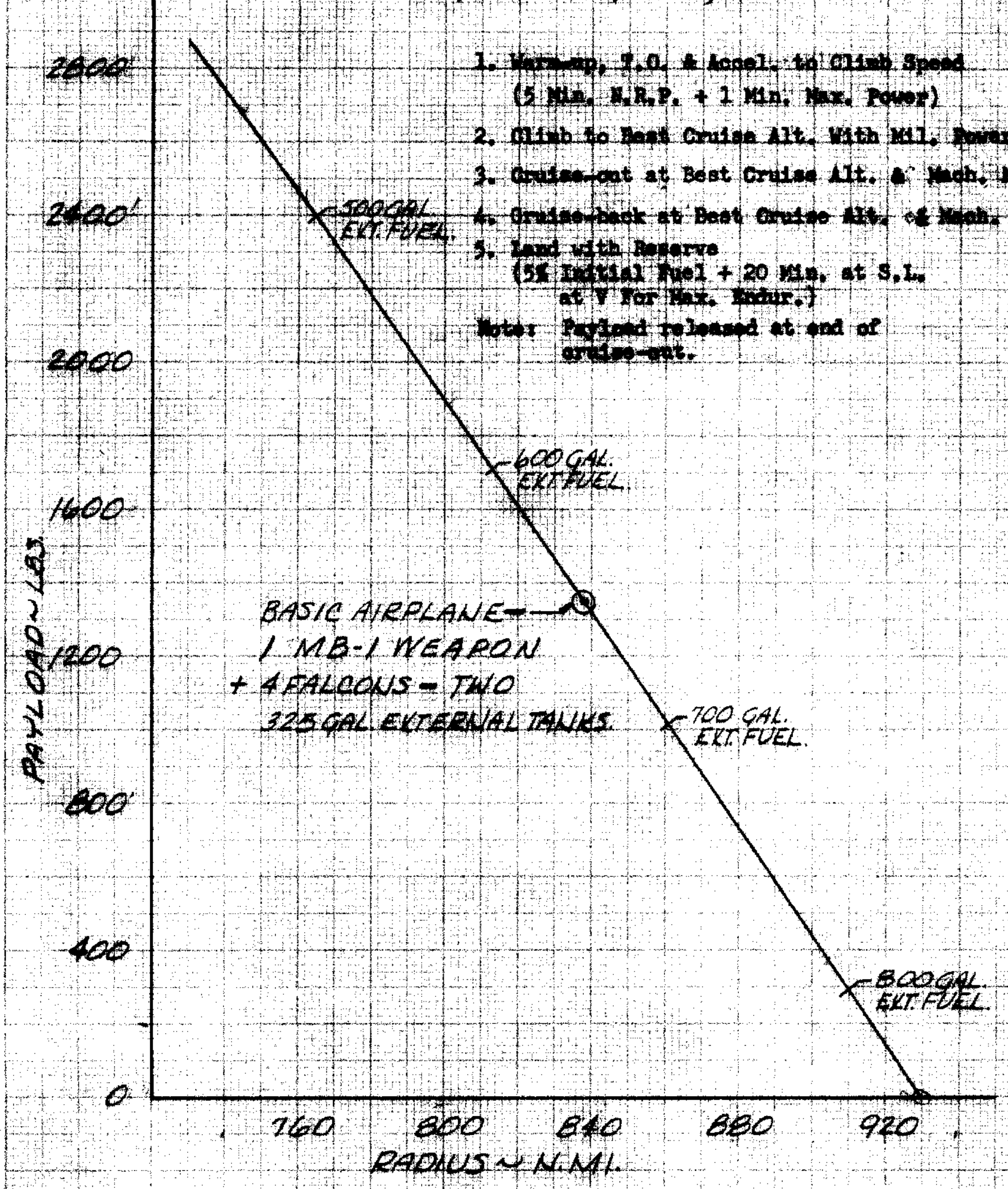
CARRIER-BASED F102B

RADIUS VERSUS PAYLOAD

TAKE-OFF GROSS WEIGHT = 49700 LBS.



1. Warm-up, T.O. & Accel. to Climb Speed
   (5 Min. N.R.P. + 1 Min. Max. Power)
2. Climb to Best Cruise Alt. With Mil. Power
3. Cruise-out at Best Cruise Alt. & Mach. No.
4. Cruise-back at Best Cruise Alt. & Mach. No.
5. Land with Reserve
   (5% Initial Fuel + 20 Min. at S.L.
   at V For Max. Endur.)

Note: Payload released at end of cruise-out.

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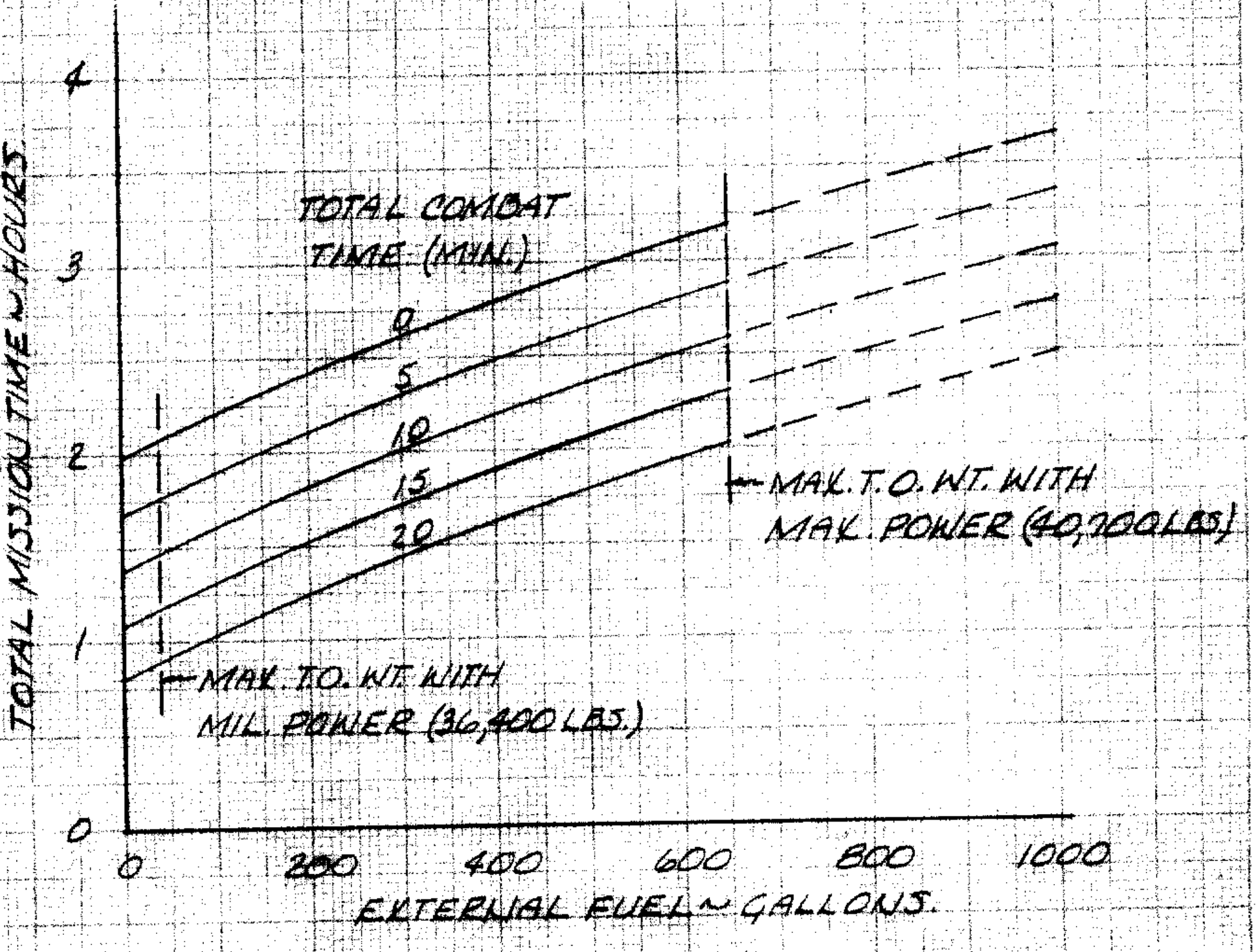
CARRIER BASED F-102B

TOTAL MISSION TIME AS A FUNCTION OF EXTERNAL FUEL LOAD AND COMBAT TIME

INTERNAL FUEL = 8775 LBS (1350 GAL)

150 N.MI. COMBAT RADIUS MISSION

1. Warm-up T.O. & Accel. to Climb Speed (5 Min. N.R.P. + 1 Min. Max. Power
2. Climb to Cruise Altitude (Approx. 40,000 Ft.)
3. Cruise-out To 150 N. Mi. From Take-off Point
4. Loiter at 40,000 Ft.
5. Combat Avg. M = 1.5 at 40,000 Ft. (3 Min. at Mil. Power for each Min. at Max. Power)
6. Cruise-back 150 N. Mi.
7. Land with Reserve
   (5% Initial Fuel + Fuel for 20 Min. at S.L. at V for
   Max. Endur.)



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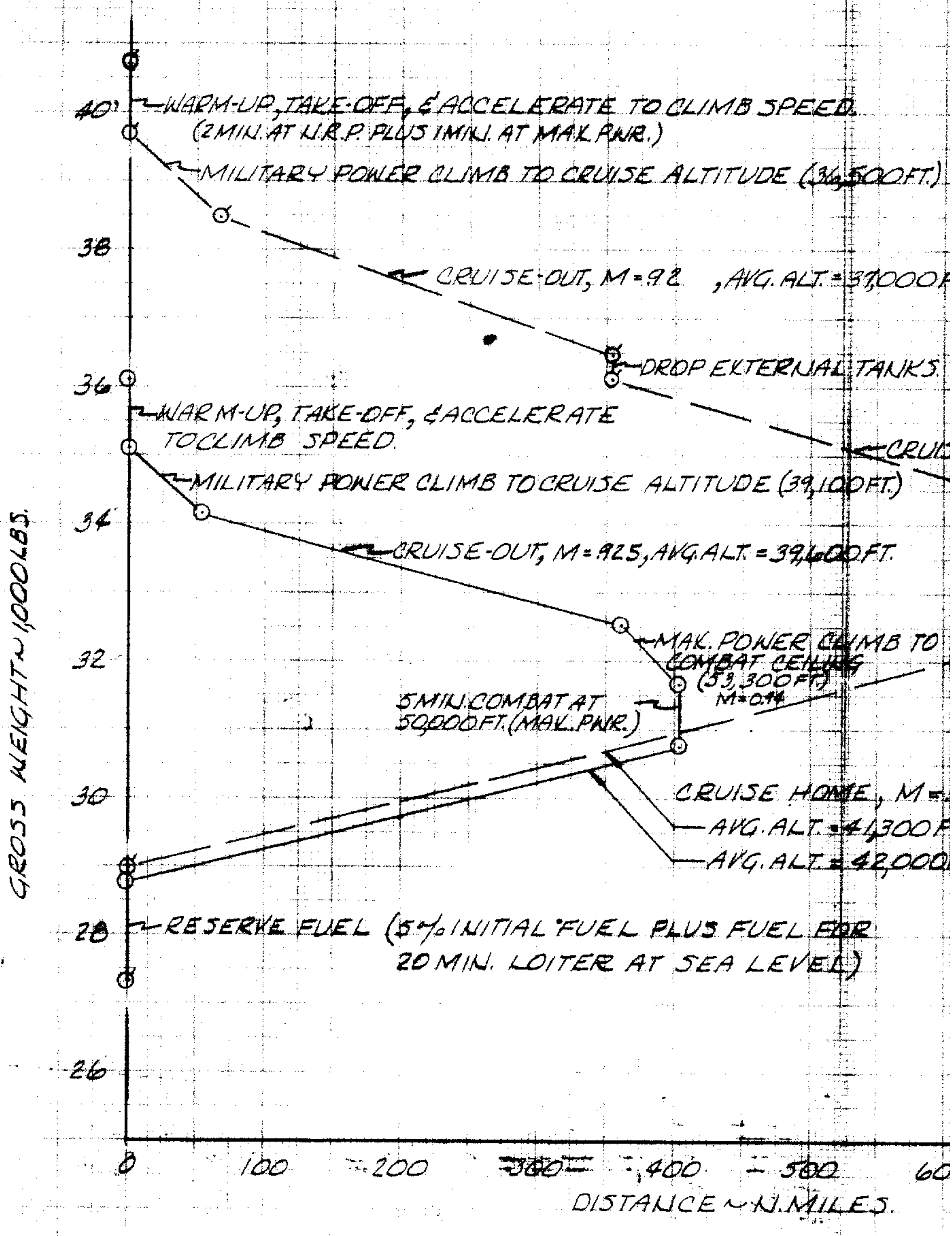
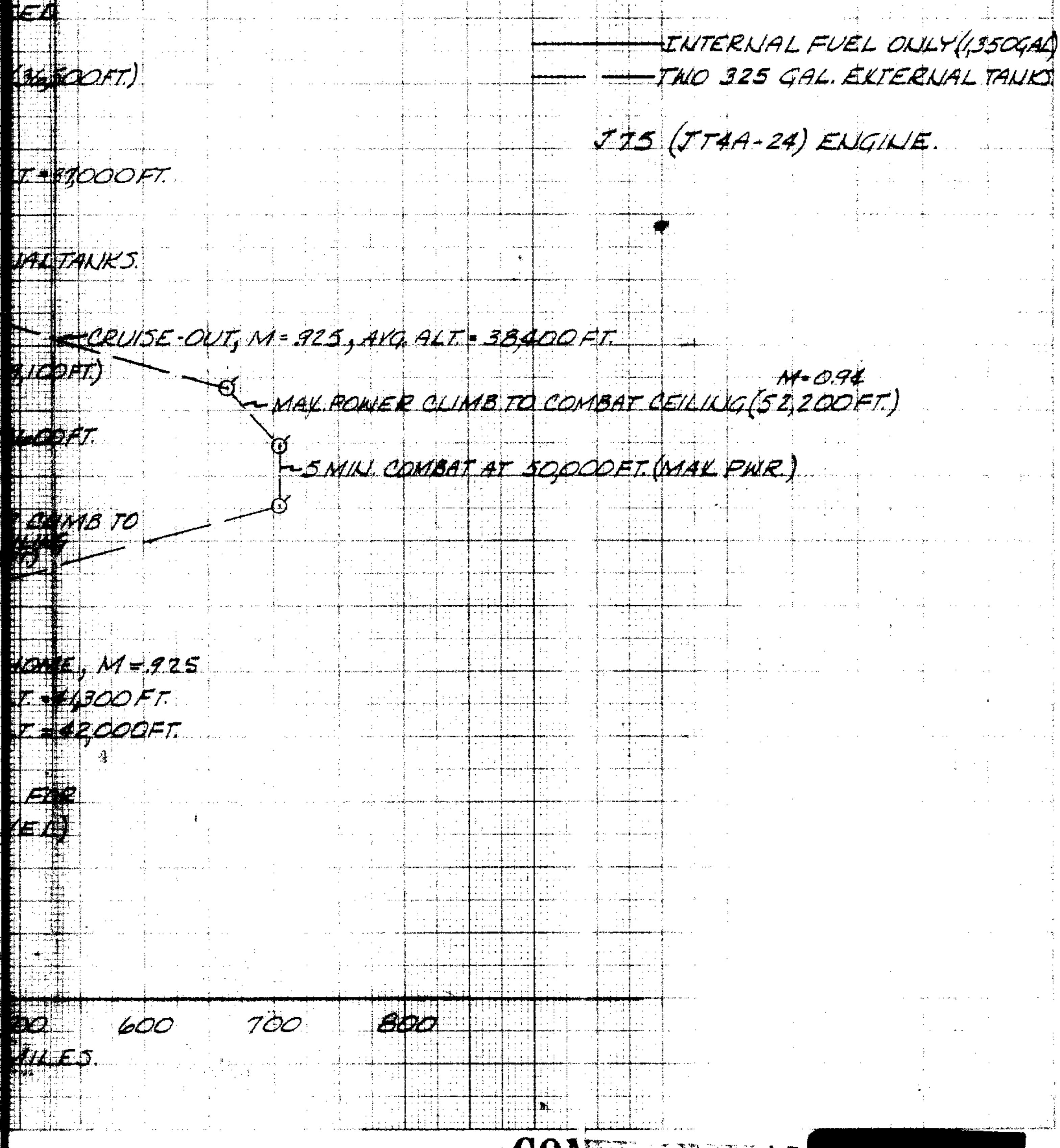


CARRIER BASED F102B

MIL-C-5011A AREA INTERCEPT MISSION SUMMARY



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K+E 10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO.
ALBANENE ®

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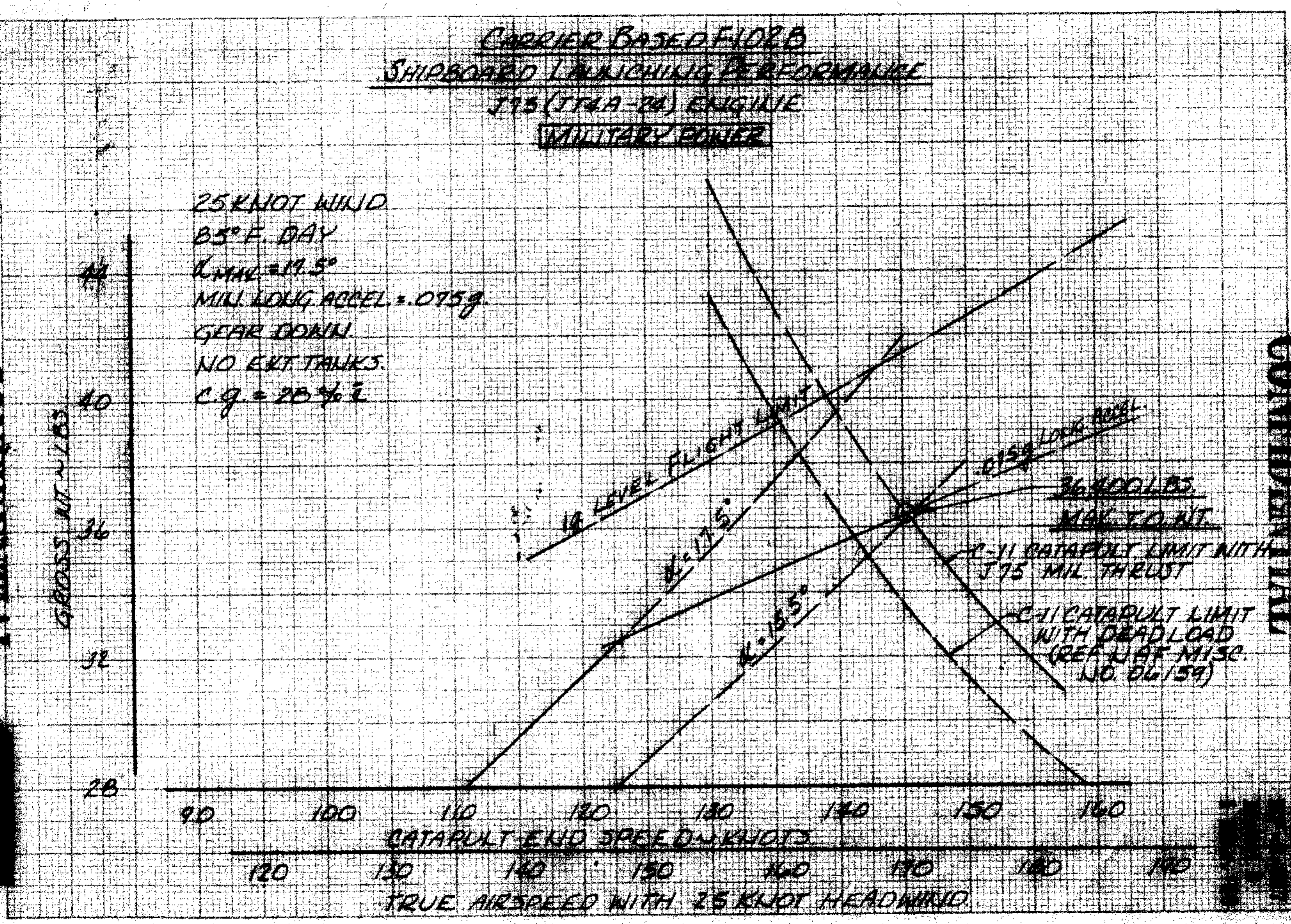

CARRIER BASED F102B
SHIPBOARD LAUNCHING PERFORMANCE
J75 (JT4A-24) ENGINE
MILITARY POWER

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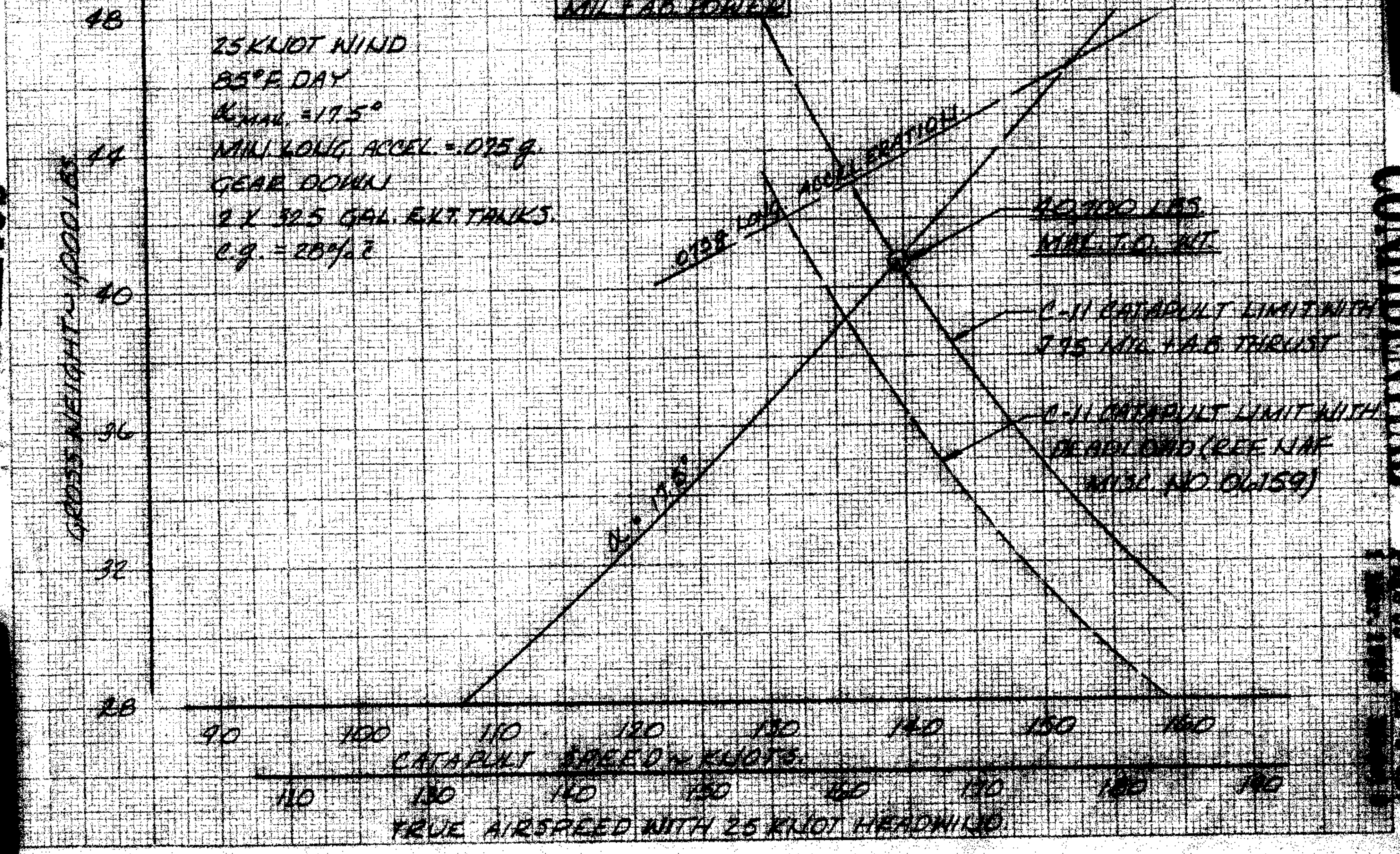
CARRIER BASED F102B
SHIPBOARD LAUNCHING PERFORMANCE
J75 (JT4A-21) ENGINE
MIL + AB POWER



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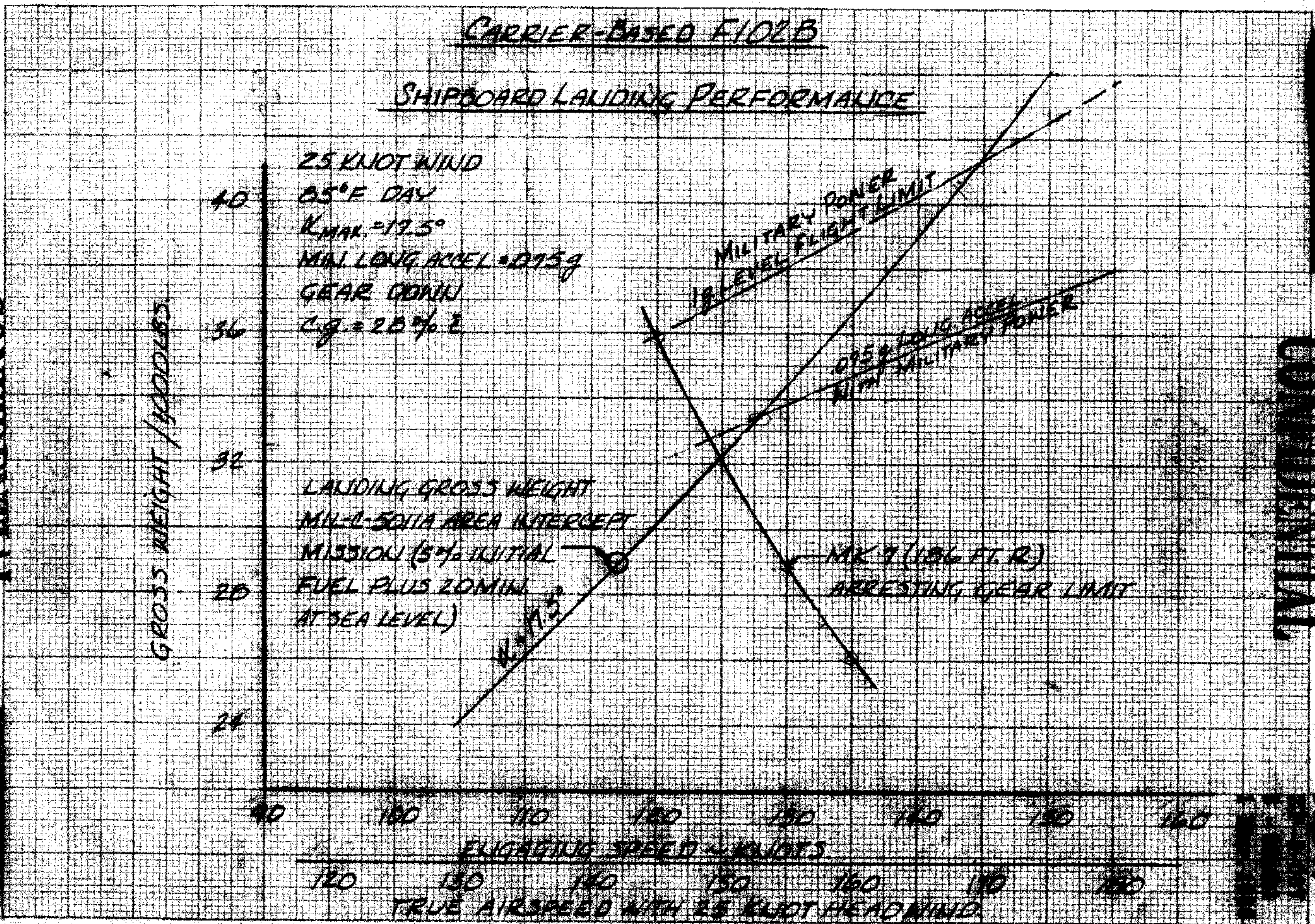

CARRIER-BASED F102B

SHIPBOARD LANDING PERFORMANCE

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SECTION III.

NAVY CARRIER-BASED VERSION
OF THE F-102B AIRPLANE

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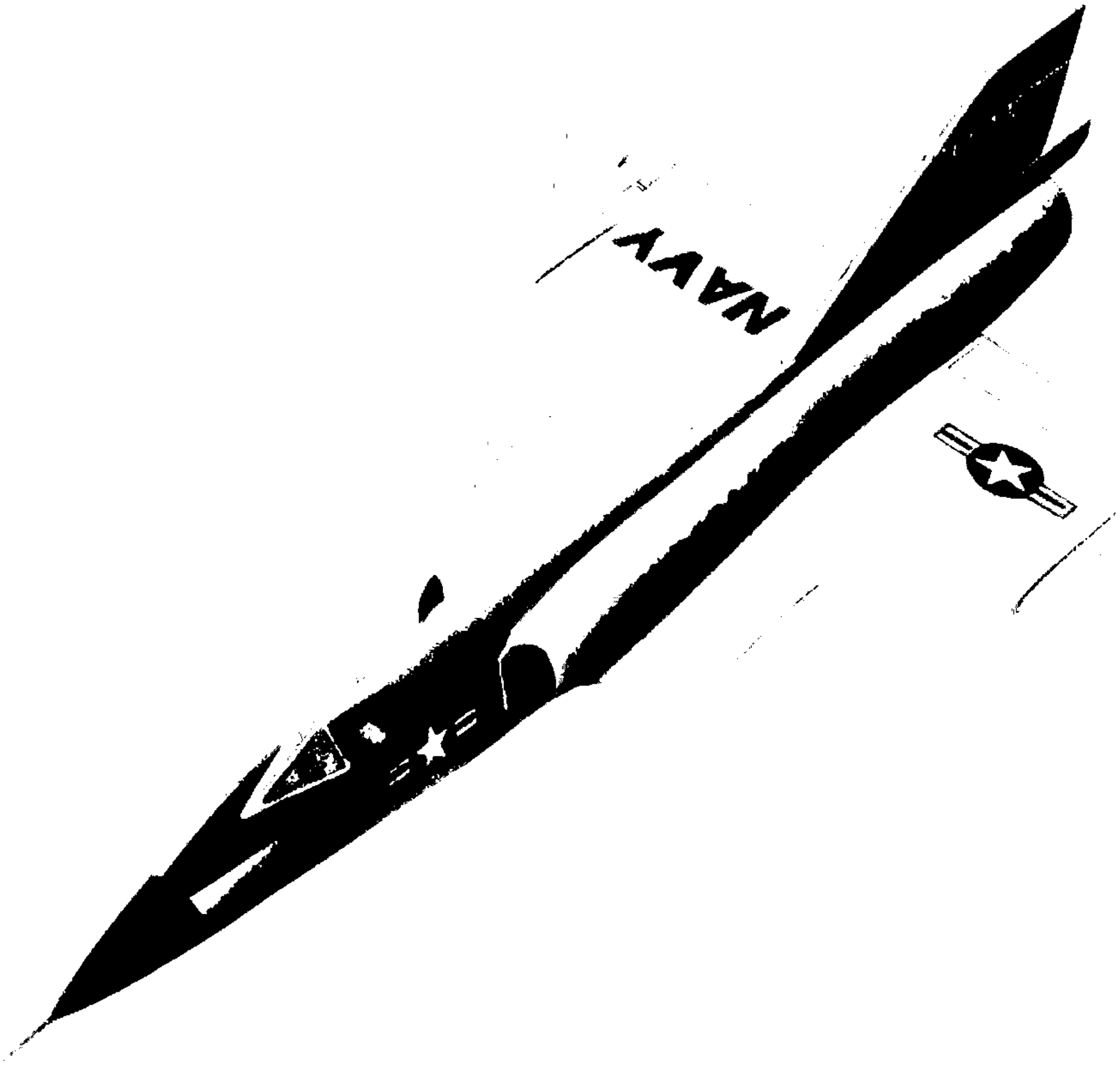

CARRIER BASED VERSION OF F-102B

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SECTION III.

DISCUSSION

A previous Convair Report No. ZP-55-13006, dated 20 June, 1955, presented a Navy carrier-based version of the F-102B airplane.

This study established changes required to build a complete Navy version of the F-102B including a fire control system and four Sparrow 1-A missiles carried internally.

Portions of that study are included in this report for reference only.

The following pages include weight and balance information and drawings of this version.

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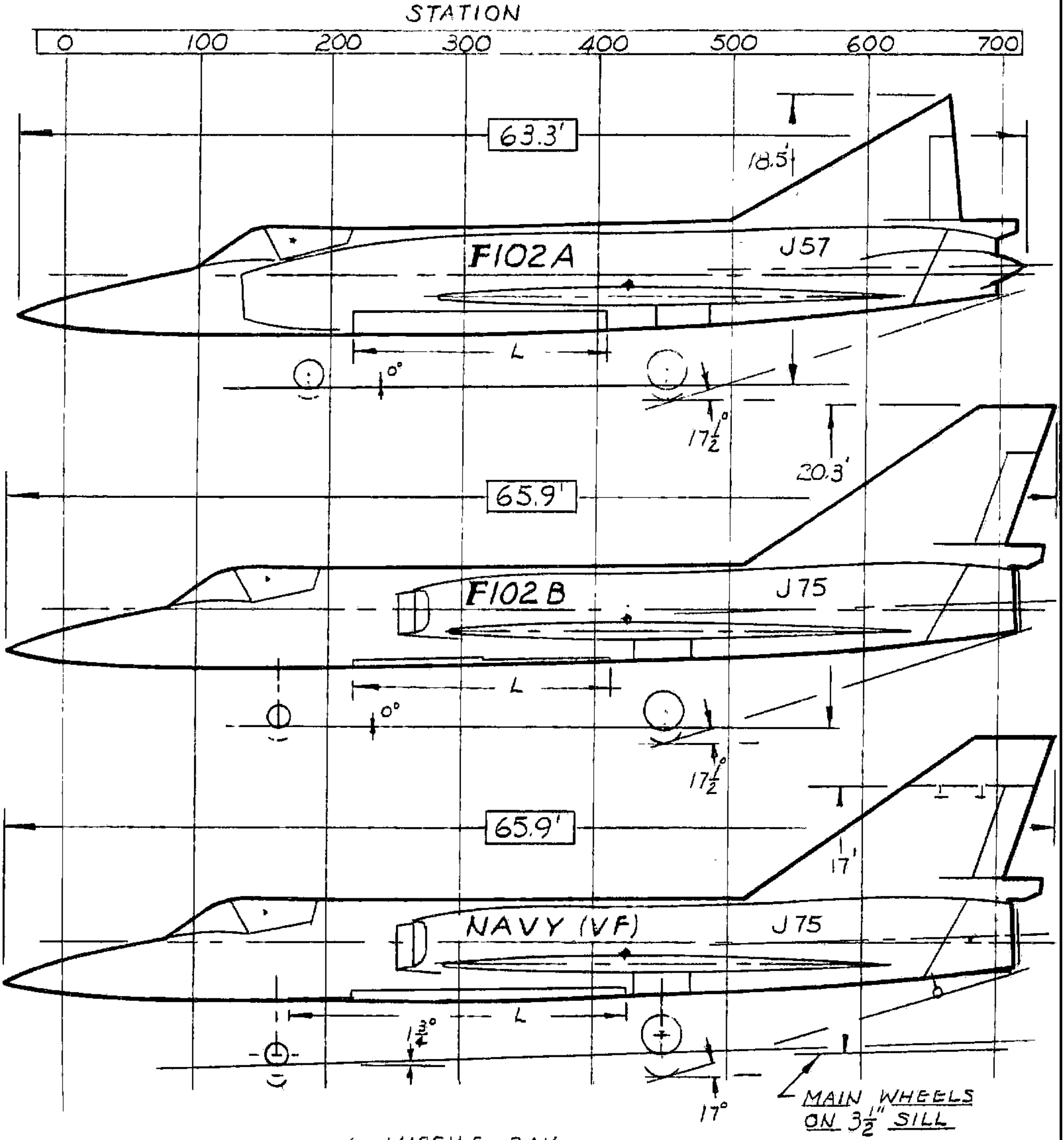

L= MISSILE BAY

CONFIGURATION COMPARISON

ANALYSIS
PREPARED BY
CHECKED BY D. I. HIXSON
REVISED BY

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Page 39
ZP-8-037
Model 8
1 Mar. '56

GROUP WEIGHT STATEMENT

CARRIER BASED FIGHTER VERSION OF F-102B

	NAVY FIGHTER			F-102B*		
Wing		3838.0			3232.0	
Tail		631.4			606.4	
Fuselage		4857.2			4423.2	
Alighting Gear		2211.8			1226.8	
Surface Controls		461.5			461.5	
Nacelle		33.5			33.5	
Propulsion		8242.3			8351.3	
Engine (1) J75	5825.0			6125.0		
Air Induction	884.6			884.6		
Exhaust	232.7			232.7		
Cooling System	71.8			71.8		
Lubricating	35.2			35.2		
Engine Controls	22.6			22.6		
Starting System	72.8			72.8		
Fuel System	1097.6			906.6		
Instruments		137.4			137.4	
Hydraulic & Pneumatic Sys.		345.6			325.6	
Electrical System		454.0			332.0	
Electronics		1150.0			2138.1	
Armament		459.1			391.1	
Furnishings		231.3			247.3	
Air Conditioning & Anti-Ice.		377.2			347.2	
Auxiliary Gear		305.6			68.6	
Ballast		300.0				
WEIGHT EMPTY			24036			22322
Pilot, Chute & Survival Kit		270.0			270.0	
Unusable Fuel		234.0			234.0	
Unusable Oil		60.0			60.0	
Fuel		8060.0			8060.0	
Armament (4) Sparrow 1-A missiles		1340.0			2054.0	
USEFUL LOAD			9964			10678
TAKE-OFF WEIGHT			34000			33000

*Data from F-102B
Detail Specification
dated 7/1/55

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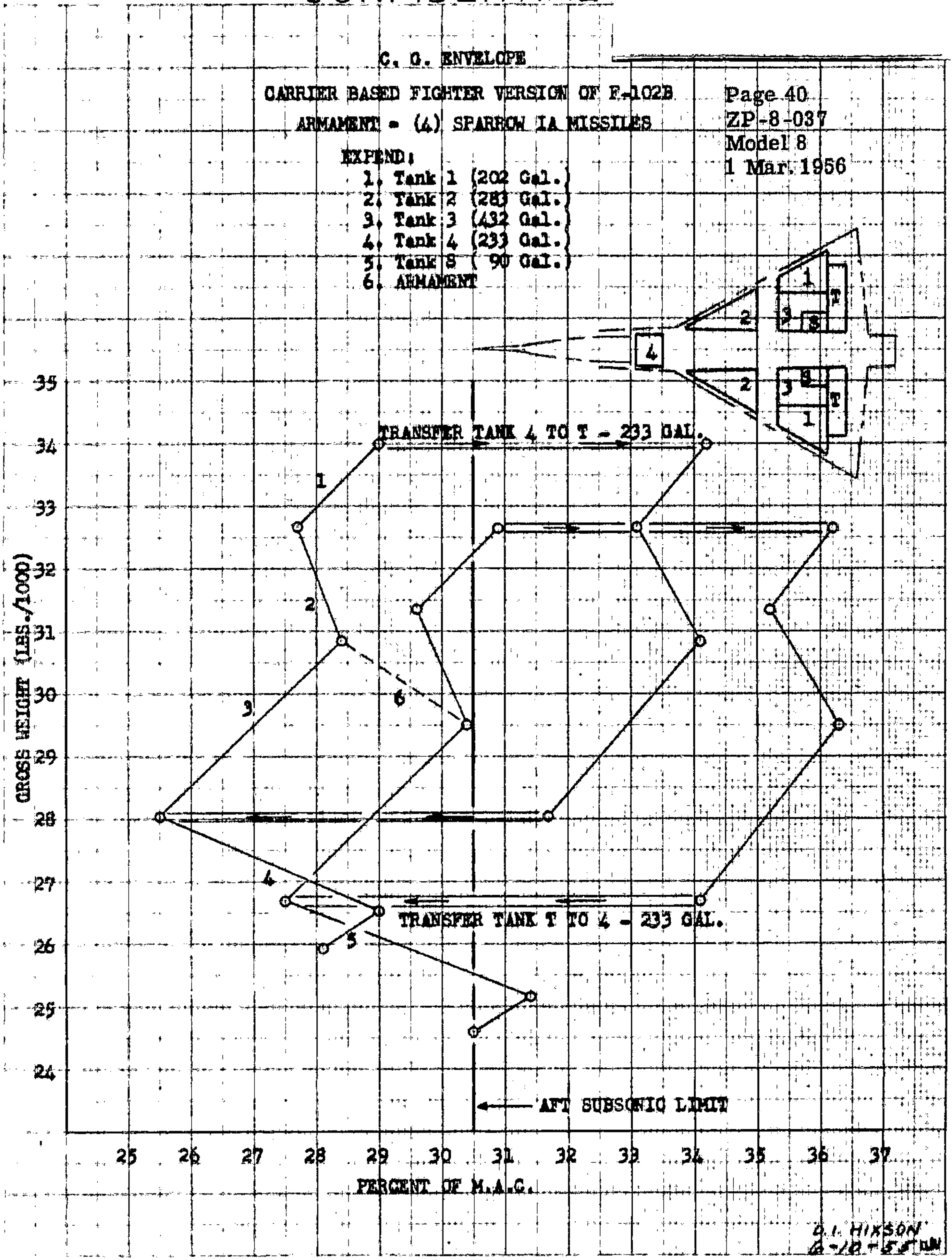
CARRIER BASED FIGHTER VERSION OF F-102B
ARMAMENT - (4) SPARROW IA MISSILES

EXPEND:
1. Tank 1 (202 Gal.)
2. Tank 2 (283 Gal.)
3. Tank 3 (432 Gal.)
4. Tank 4 (233 Gal.)
5. Tank 8 (90 Gal.)
6. ARMAMENT



D. I. HIXSON
6-10-55 DW

III. CARRIER-BASED VERSION OF F-102B (NAVY)

III. CARRIER-BASED VERSION OF F-102B (NAVY)

III. CARRIER-BASED VERSION OF F-102B (NAVY)

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II. CARRIER-BASED VERSION OF F-102B AIRPLANE

II. CARRIER-BASED VERSION OF F-102B AIRPLANE

II. CARRIER-BASED VERSION OF F-102B AIRPLANE

I. EXISTING F-102A & F-102B AIRPLANES

I. EXISTING F-102A & F-102B AIRPLANES

I. EXISTING F-102A & F-102B AIRPLANES

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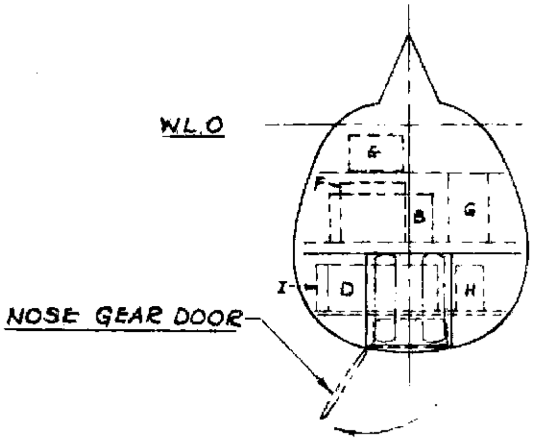
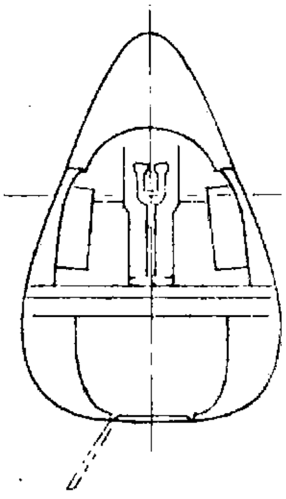
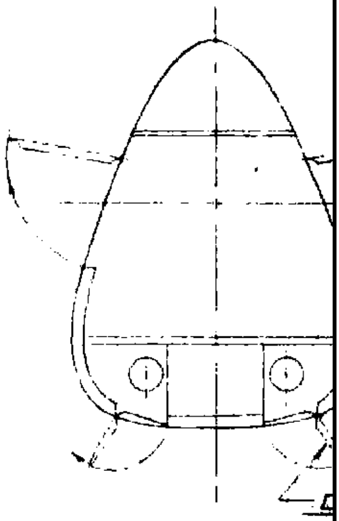


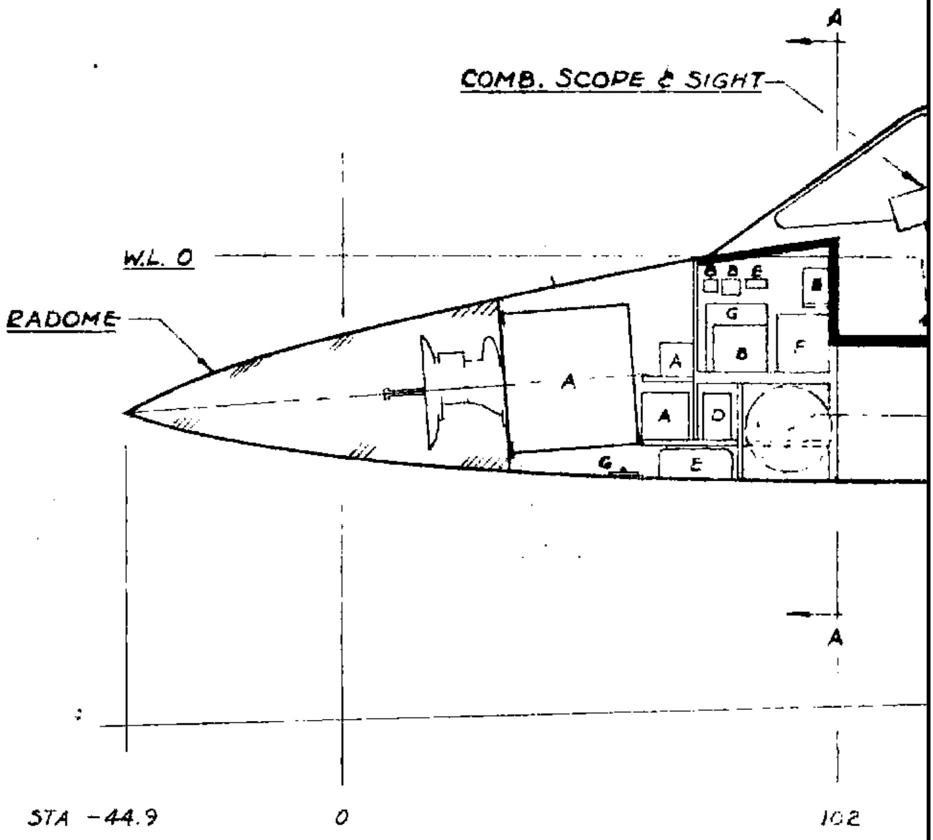
SEC. A-A



SEC. B-B



SEC. C-C



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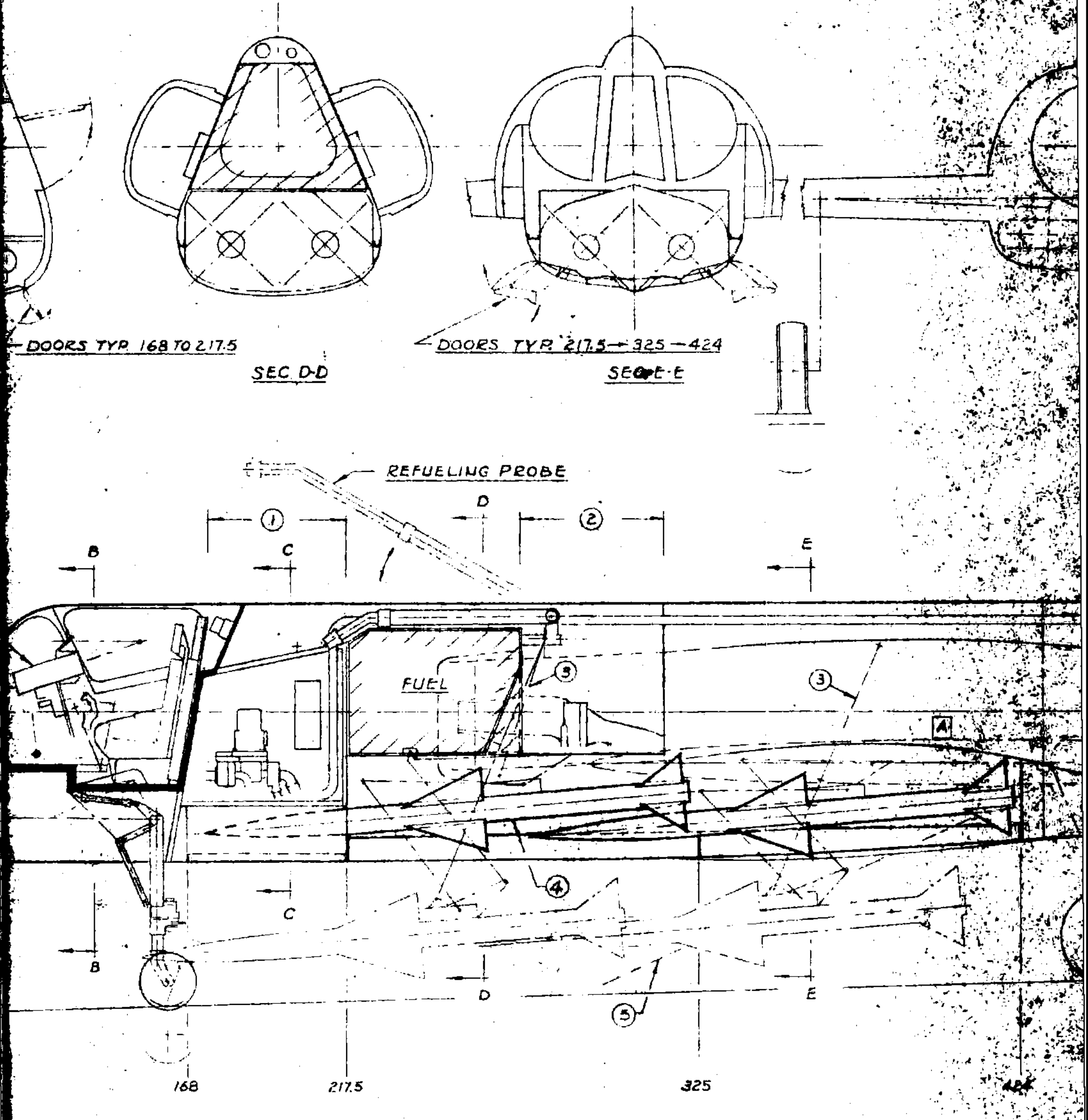
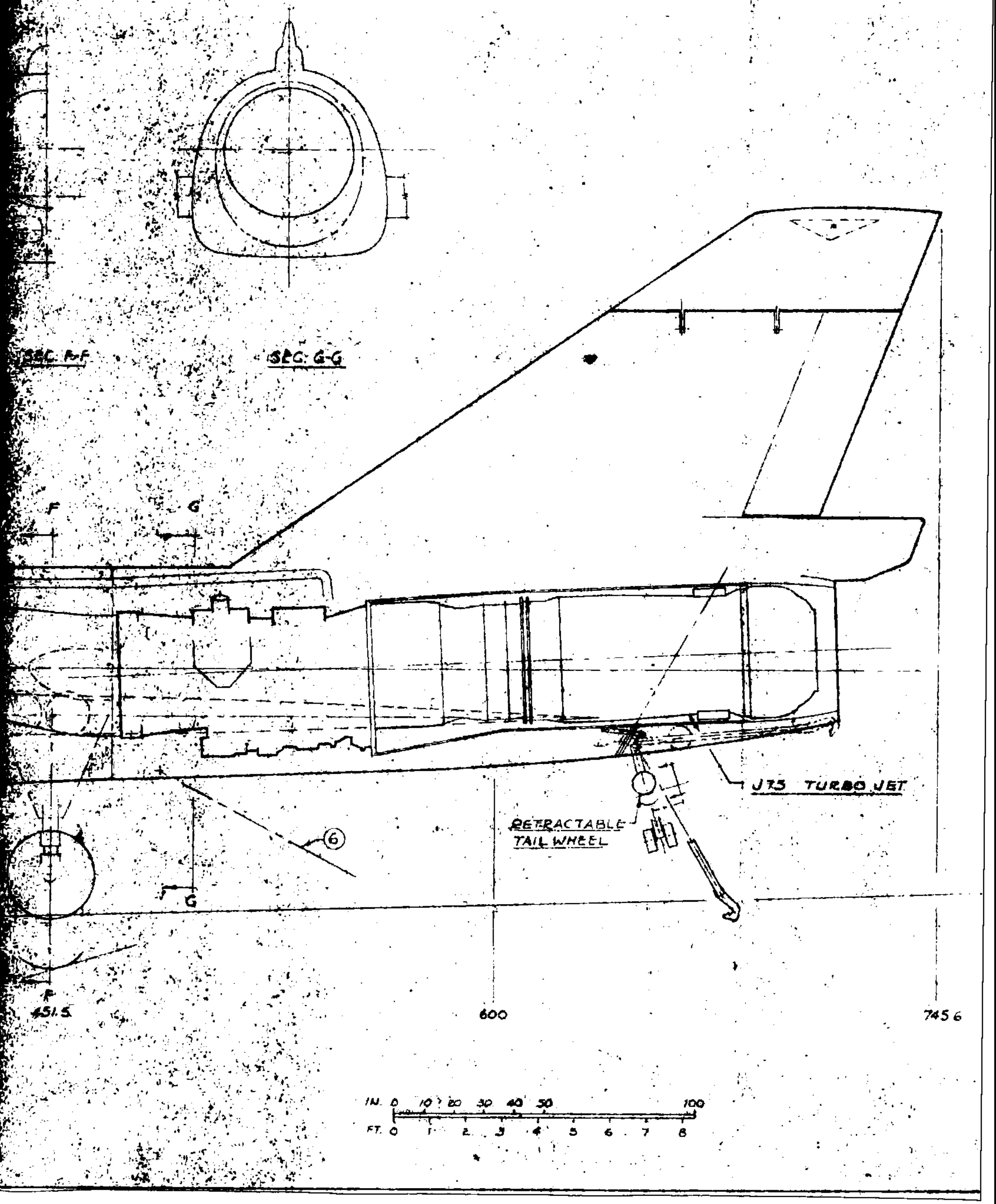


① REFRIG. UNIT, AIR TURBINE ALTERNATOR DRIVE, MISCL ELECTRICAL & HYD. EQUIP
② HEAT EXCHANGER, MISCL ELECT & HYD. EQUIP
③ ACTUATORS: MISSILE DISPLACING GEAR
④ SPARROW IA MISSILE (4)
⑤ CATAPULT BRIDLE
⑥ HOLD BACK CABLE


SEC. F-F
SEC. G-G
F
G
J75 TURBO JET
RETRACTABLE TAIL WHEEL
6
G
F
451.5
600
745.6
IN. 0 10 20 30 40 50 100
FT. 0 1 2 3 4 5 6 7 8

ZPB-037

PAGE

POWER PLANT

J75 (JT4A-24) TURBO JET (1)
RATINGS : MAX. (A/B) 23500 LB.
MIL. 15500 LB.

ARMAMENT

SPARROW I A AIR-TO-AIR MISSILE (4)

ELECTRONICS

FIRE CONTROL
- A. ACS AERO 11 A
  INCLUDES
  - APQ-50 RADAR
  - ACD 11B COMPUTER

NAVIGATION
- B. ARN-21 TACAN
- C. DEAD RECKONING TRACKER
- D. APN-22 ALTIMETER
- E. ARA-25 DF

COMMUNICATIONS
- F. ARC-52

IFF
- G. APX-20
- H APA-89 CODER

INTERPHONE (GROUND)
- I. AIC-10

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CONVAIR

PRELIMINARY DESIGN DRAWING

INTERIOR ARRANGEMENT
CARRIER BASED FIGHTER VERSION
OF F-102 B

BY | CHECKED | APPROVED | SCALE 1/20 | DATE 10 JUN 55

CONVAIR—SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION
SD-55-13005

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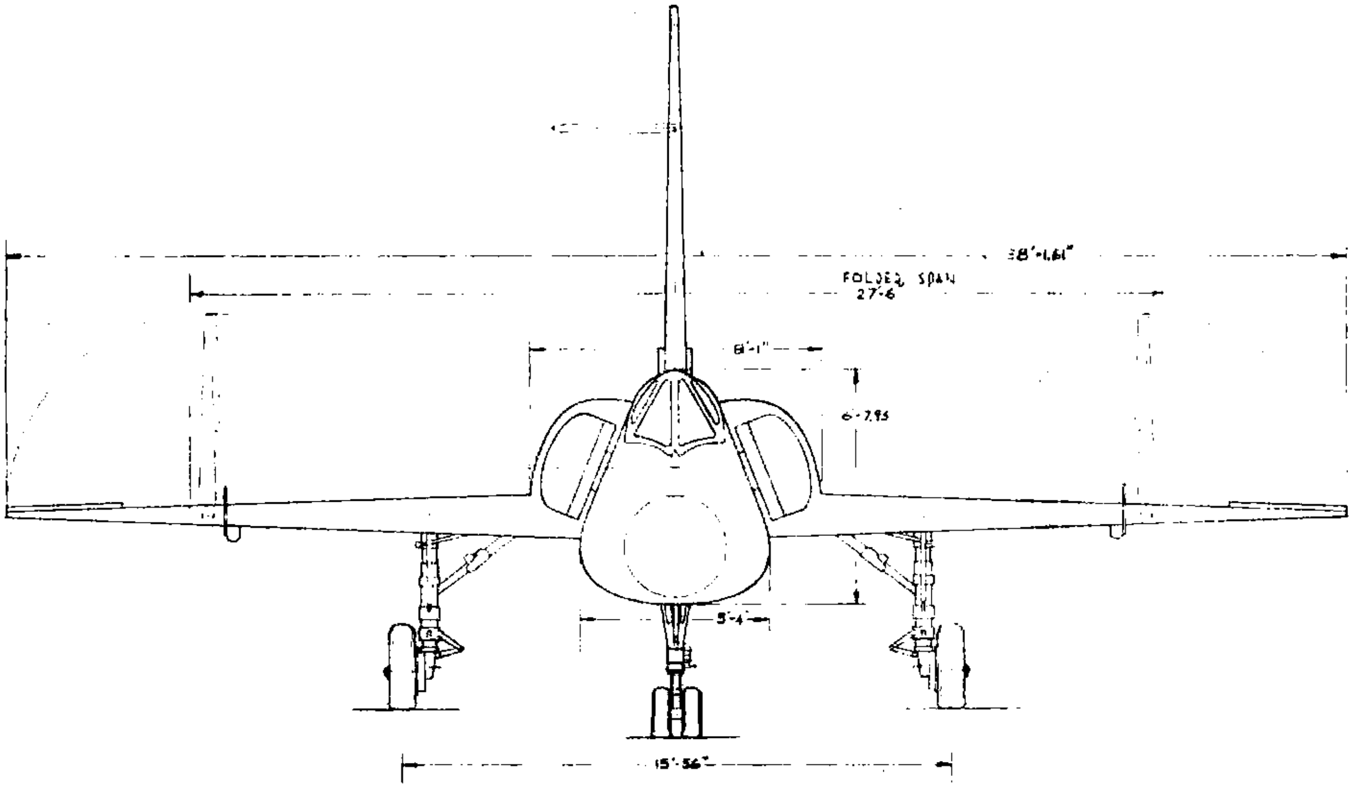
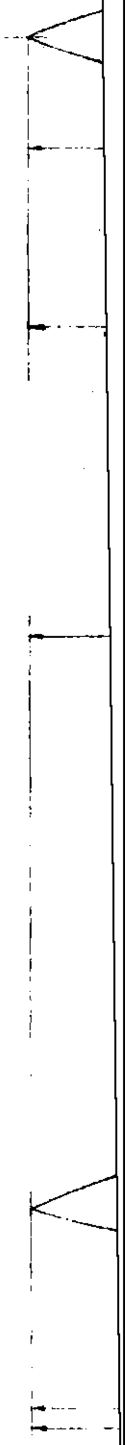
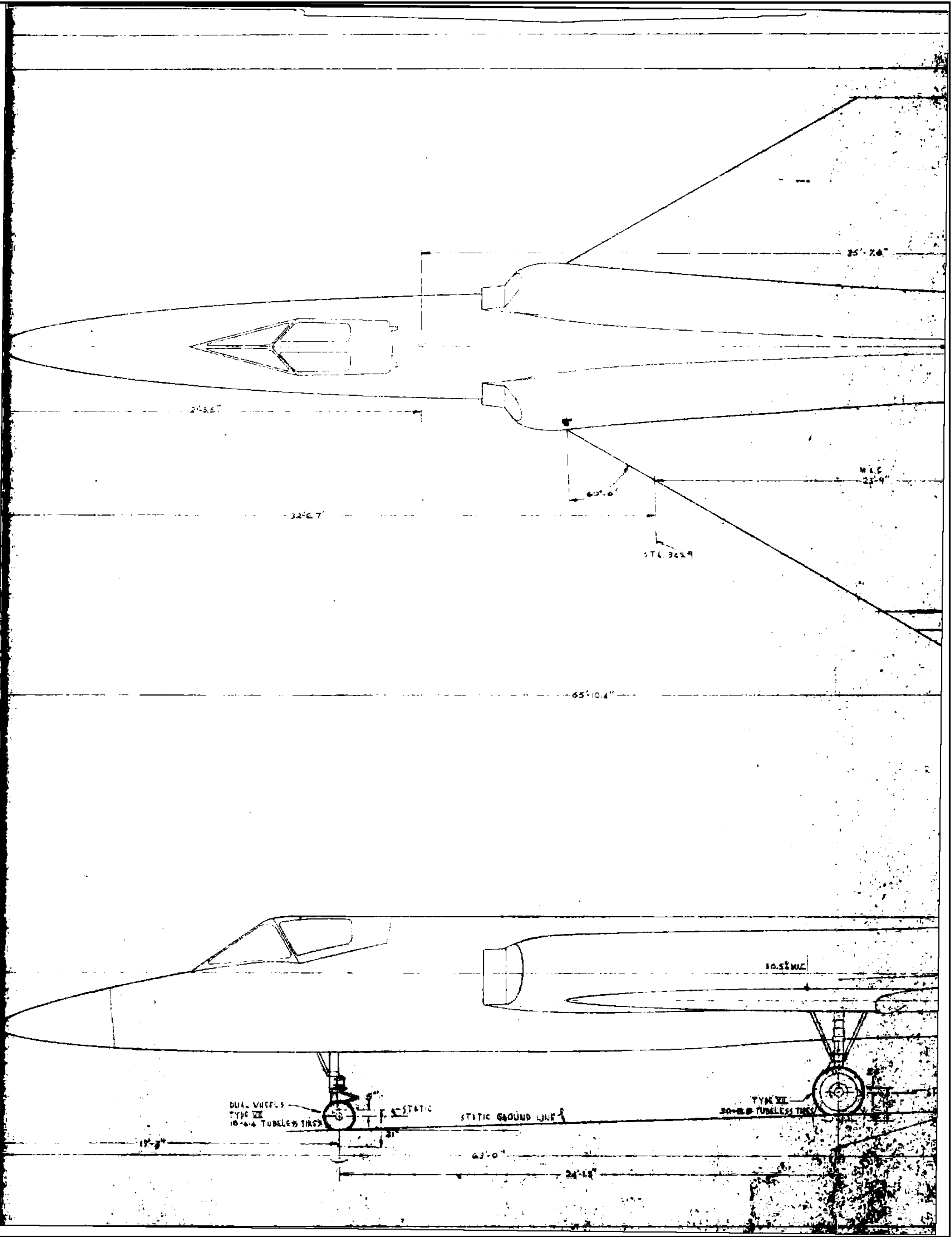
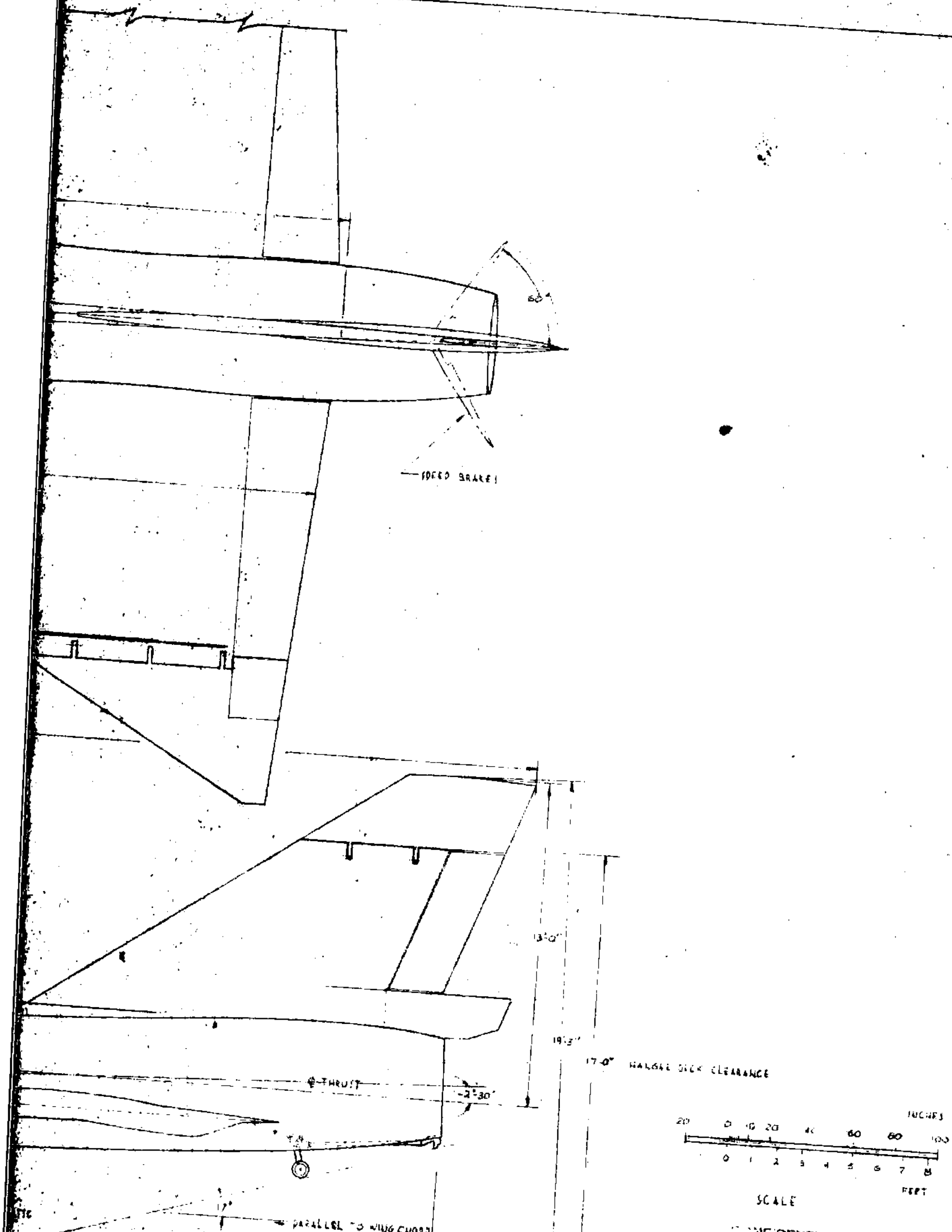
CHARACTERISTICS

WING	
AIRFOIL SECTION	NACA 0004-65 MOD
AREA	697.83 SQ.FT
ELEVON AREA	66.66 SQ.FT.
VERTICAL TAIL	
AIRFOIL SECTION	NACA 0004-65 MOD.
AREA	105 SQ.FT.
RUDDER AREA	12.26 SQ.FT
SPEED BRAKE	
PROJECTED FRONTAL AREA	11.1 SQ.FT.
POWERPLANT	
MANUFACTURER	PRATT-WHITNEY
DESIGNATION	J-75(JT4A-24)
MAX THRUST WITH AFTERBURNER	23,500 LBS.
MILITARY THRUST	15,500 LBS






35'-7.6"
24'-3.5"
M.A.C.
23'-9"
60°-6'
32'-6.7"
STA. 345.9
65'-10.4"
30.5% MAC
DUAL WHEELS
TYPE VII
18-4.4 TUBELESS TIRES
8"
STATIC
STATIC GROUND LINE
TYPE VII
30-8.8 TUBELESS TIRES
17'-3"
63'-0"
24'-1.5"



60°
SPEED BRAKES
13'-0"
19'-3"
17'-0" HANGAR DECK CLEARANCE
℄ THRUST
-2°30'
17°
PARALLEL TO WING CHORD
GROUND LINE WITH MAIN WHEELS OVER 3½" SILL
20 0 10 20 40 60 80 100 INCHES
0 1 2 3 4 5 6 7 8 FEET
SCALE

CONFIDENTIAL

CONVAIR

PRELIMINARY DESIGN DRAWING

GENERAL ARRANGEMENT
CARRIER BASED
FIGHTER VERSION OF F-102B

WGL: SIEGEL
SCALE 1/20
DATE 6-10-55
CONVAIR—SAN DIEGO CALIFORNIA

ZP-8-051 A c/s

UNCLASSIFIE

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Filed in Misc Tube

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CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-051 A
DATE 17 April, 1957
MODEL F-102

UNCLASSIFIED

TITLE

THE F-102 INTERCEPTOR AND SELECTED ALTERNATE VERSIONS

PREPARED BY T. M. Hemphill
T. M. Hemphill

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert
Chief, Pre Design

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INTRODUCTION

The F-102, a revolutionary All Weather Supersonic Interceptor in service in units of Air Defense Command, guarding the United States, is in quantity production at the Convair San Diego, Division of General Dynamics Corporation.

The F-102, because it is a delta wing airplane, is fundamentally superior to the more conventional types for fighter missions. It is also simpler, safer, and more rugged.

The size and arrangement of the F-102 airplane makes it really adaptable to various types of equipment and armament. Thus it has great potential, not only as an interceptor but for other uses as well.

This report briefly describes the F-102 Interceptor and three other versions of the airplane; a Fighter Bomber, a Photo Reconnaissance Airplane, and a Ferret Airplane. Although, other versions are feasible, these are the ones which it is felt will be most useful and attractive to allies of the United States.

This report is intended to provide a basis for discussion to develop configurations most suitable for users requirements.

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ADVANTAGES OF THE F-102

The F-102, because it is a delta wing airplane, is fundamentally superior to the more conventional types for fighter missions. Several of the reasons for its superiority and the way it manifests itself are outlined below:

1. In the design of the delta wing type airplane the conventional horizontal tail is omitted. This eliminates a number of working parts (surfaces) along with the direct and interference drag associated with them. In order to achieve longitudinal control the elevators are built into the wing trailing edge. Actually they are combined with the ailerons to become "elevons". A loss in maximum lift would result from the above but to prevent it, the wing is increased in area by 60% to 90% over what it would be with the conventional wing-tail combination. The resulting long chord places the elevons far enough aft to be very effective in longitudinal control. But more important, the large increase in wing area provides a still greater increase in wing volume (two and a half to four times that of a straight or swept wing airplane). The drag increase which would be associated with the increased wing area is offset by the elimination of the horizontal tail.

The large wing provides space for fuel, not possible with conventional designs, thus freeing space in the fuselage for armament, fire control, guidance, and communications equipment. Most other designs are required to carry armament externally thus degrading performance. The large wing also permits low landing speed and short takeoff and landing distances.

2. The increased space made available in the fuselage by carrying fuel in the wing provides better access for maintenance of equipment, accessories, and power units carried therein.

3. Because of the great amount of taper in both thickness and chord the delta wing is inherently very stiff and light. The great stiffness makes it resistant to aeroelastic problems - flutter, and reduced control. Its light weight contributes to its superior performance.

[redacted]

4. The delta wing shape contributes heavily to smoothness and stability experienced by the F-102 airplane in passing through the transonic speed region. It is also largely responsible for the superior high altitude maneuverability and low speed control and stability of the F-102. Good slow speed stability is particularly important during the transition training of pilots.

5. The delta wing arrangement provides design simplicity. There are only three movable control surfaces as compared with a possible seven on the conventional arrangement with flaps and leading edge slots or flaps. The absence of these additional surfaces means greater reliability and less maintenance; less pilot fatigue.

6. Supersonic airplanes with conventional tails exhibit erratic behavior under certain flight conditions when the horizontal tail enters or leaves the wake from the wing. Delta wing airplanes, having no horizontal tails, do not have this difficulty.

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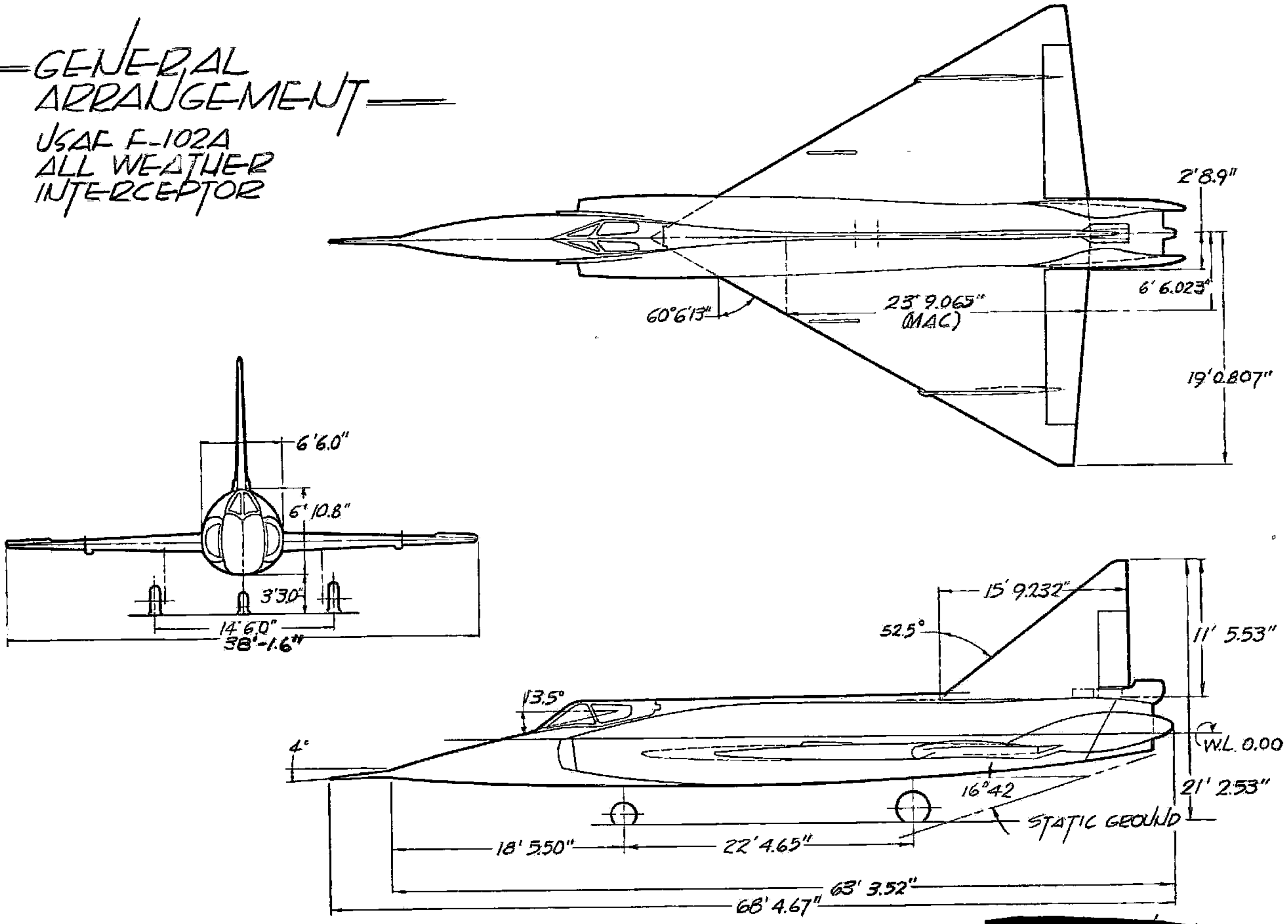
GENERAL ARRANGEMENT

USAF F-102A
ALL WEATHER
INTERCEPTOR



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NEW DEVELOPMENTS

The F-102 program includes a plan for the incorporation of New Developments into its design. Two major improvements are a change in wing leading edge camber known as Case XX and an engine improvement which is identified by a designation change from J57-P23 to J57-P47. The effect of these improvements is shown in the table below and the performance curves that follow.

PERFORMANCE SUMMARY

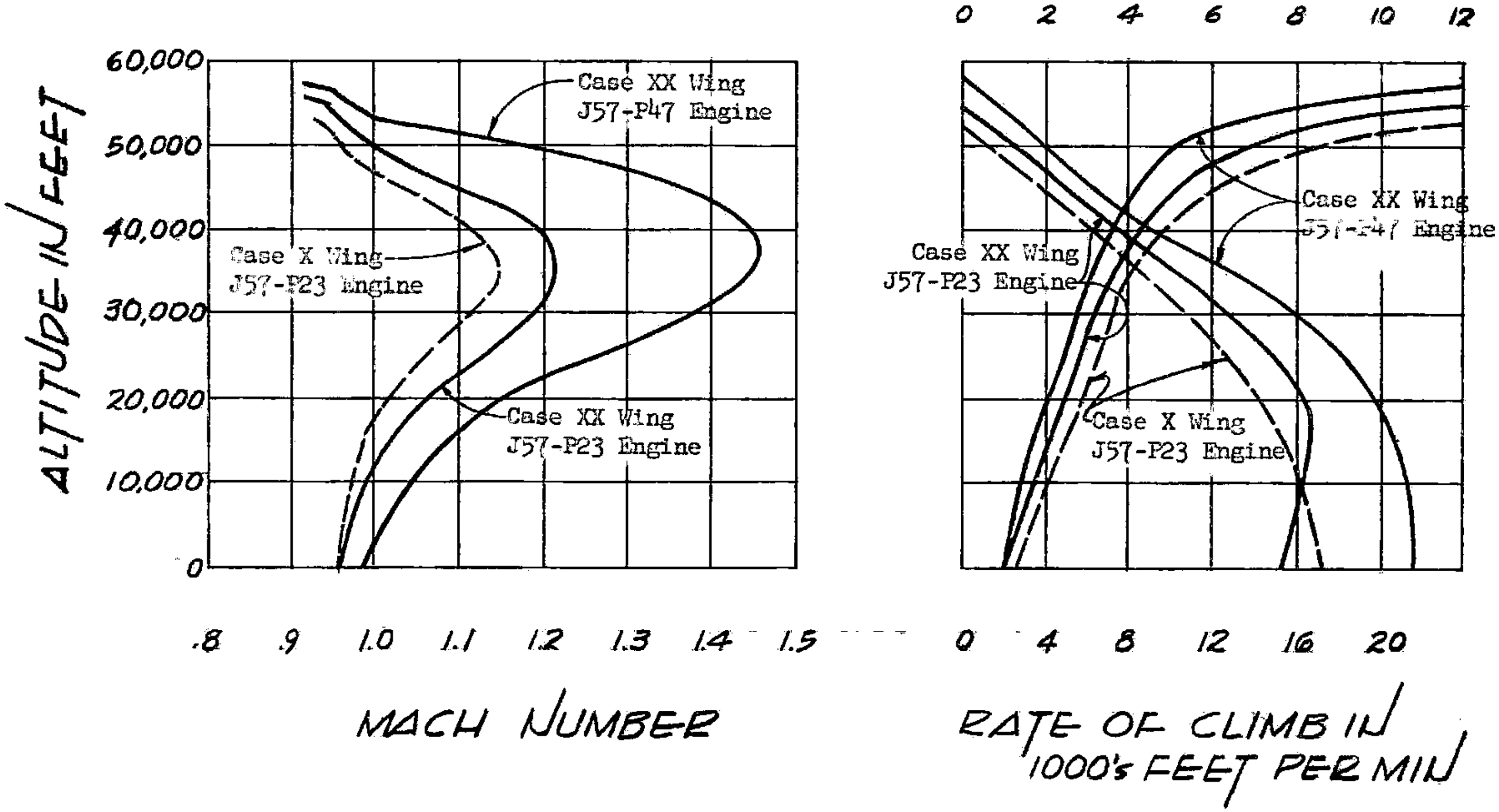
INTERCEPTOR

		F-102 CASE X J57-P23	F-102 CASE XX J57-P23	F-102 CASE XX J57-P47
RATED THRUST @ SEA LEVEL	LB.	16000	16000	16900
TAKE OFF WEIGHT	LB.	27884	27966	27809
FUEL	LB.	6825	6825	6825
V-MAX A/B POWER @ 35000	MACH	1.15	1.23	1.45
V-MAX A/B POWER @ 50000	MACH	.97	1.0	1.19
COMBAT CEILING - A/B POWER	FT.	51700	54200	56000
TIME TO CLIMB A/B POWER TO 40000 FT.	MIN.	4.7	4.1	3.5
TIME TO CLIMB A/B POWER TO 50000 FT.	MIN.	7.9	6.4	5.1
TAKE OFF OVER 50' OBSTACLE	FT.	3700	3700	3500
LAND OVER 50' OBSTACLE	FT.	4100	4100	4100
LAND WITH DRAG CHUTE	FT.	3500	3500	3500
COMBAT RADIUS (AREA INTERCEPT MISS.)	N. MI.	340	430	410
LOITER TIME (POINT INTERCEPT MISS.)	HR.	1.2	1.6	1.5

[redacted]

PERFORMANCE

(AT COMBAT WEIGHT)



SPEED VS ALTITUDE

TIME TO CLIMB IN MIN

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THE F-102 INTERCEPTOR

The F-102 Interceptor was designed to intercept enemy bombers under all conditions of weather and light. The characteristics which were necessarily built into the airplane are short take-off and landing runs, an extremely high rate of climb, a maximum combat ceiling, good stability and high maneuverability at all altitudes, supersonic speed, and an adequate radius of action.

In order to achieve the highest possible probability of kill maximum guidance is provided for both the airplane and the missiles it carries. Rockets are carried simultaneously with the missiles and they utilize the same fire control system.

The interior arrangement of the F-102 Interceptor is shown on Convair Drawing SD-56-08051. In the nose are the radar, fire control system and part of the communication, navigation and landing systems.

Immediately aft of the nose section are the cockpit and nose wheel compartment. The battery and a number of electrical units occupy the same compartment as the nose wheel and are afforded easy access through the nose wheel door opening.

The space behind the cockpit contains the refrigeration and air conditioning equipment. It also houses electrical and electronics components.

The lower portion of the mid section of the fuselage is the armament bay. It provides space for six Falcon missiles mounted on parallel type mechanisms which position the missiles several inches clear of and below the fuselage when in the extended position. In the "up" position the missiles are housed completely within the fuselage. Rockets are carried within the structure of the missile bay doors.

The upper mid section of the fuselage carries many of the electrical system components.

The main wheel well and power plant sections are aft of the armament bay.

[redacted]

In operation, the interceptor pilot receives instructions from the ground radar operator which enable him to put the airplane in position for a firing run on the enemy bomber. The ground radar operator can also tell the interceptor pilot where the image of the bomber should appear on the pilots scope.

Once the target becomes visible on his "scope", the pilot is able to challenge it and get an identifying response to determine whether or not it is friendly. If the target is unfriendly, "lock on" can be accomplished by the pilot manually manipulating a pair of cursors (cross hairs) to the position of the target image on the scope.

From "lock on" to armament release all operations are automatic including guidance, opening of the armament bay doors, displacing armament into firing position, firing armament and closing doors. The pilot monitors the operation and steers away from the debris after armament release.

There are two means of returning to base. The ground control radar operator can give the pilot steering courses. Or, the pilot can use the ground mapping feature of the radar equipment.

The development programs include improvements in the system which will permit the ground controller to direct the outbound and inbound portions of an attack. These improvements utilize steering signals transmitted through data link components to the aircraft control system.

Other types of fire control systems are available. These include other radars, infra red, visual and combinations of these.

[redacted]

F-102 INTERCEPTOR

WEIGHT SUMMARY

(J57-P23 Engine)

Item	
Structure	8,026 Lb.
Propulsion Group	6,439
Electrical, Hydraulics & Pneumatics	890
Electronics	2,248
Armament Provisions	592
Furnishings and Fixed Equipment	647
Surface Controls	435
WEIGHT EMPTY	19,277 Lb.
Useful Load	
Pilot, Survival Kit, Parachute	248
Unusable Fuel	154
Fuel usable (JP-4) 1050 Gal.	6,825
Oil	46
Missiles - Falcons (6)	762
Rockets - 2.75" (24)	432
Missile Launchers	222
NORMAL TAKE-OFF GROSS WEIGHT	27,966

NOTE:

The F-102 has provisions of carrying 2 - 230 gal. external fuel tanks, one on each wing to approximately double the radius of action. The weight of tanks, all external provisions and fuel is 3,272 Lbs.

[redacted]

SELECTED ALTERNATE VERSIONS

Brief studies of selected alternate versions of the F-102 Interceptor are shown on the following pages. The spaces available in the nose and belly of the F-102 after the removal of the Interceptor electronics and armament are so large that a great variety of equipment can be installed there to produce many different fighter versions.

The studies shown here are simply to indicate what can be done and to serve as a basis for discussions to develop the configuration desired by the customer.

In general those versions other than the Interceptor have missions with greater radii of action. Accordingly space not used in the armament bays for equipment is used for additional fuel. Also external fuel tanks are attached to the wings outboard of the wheels.

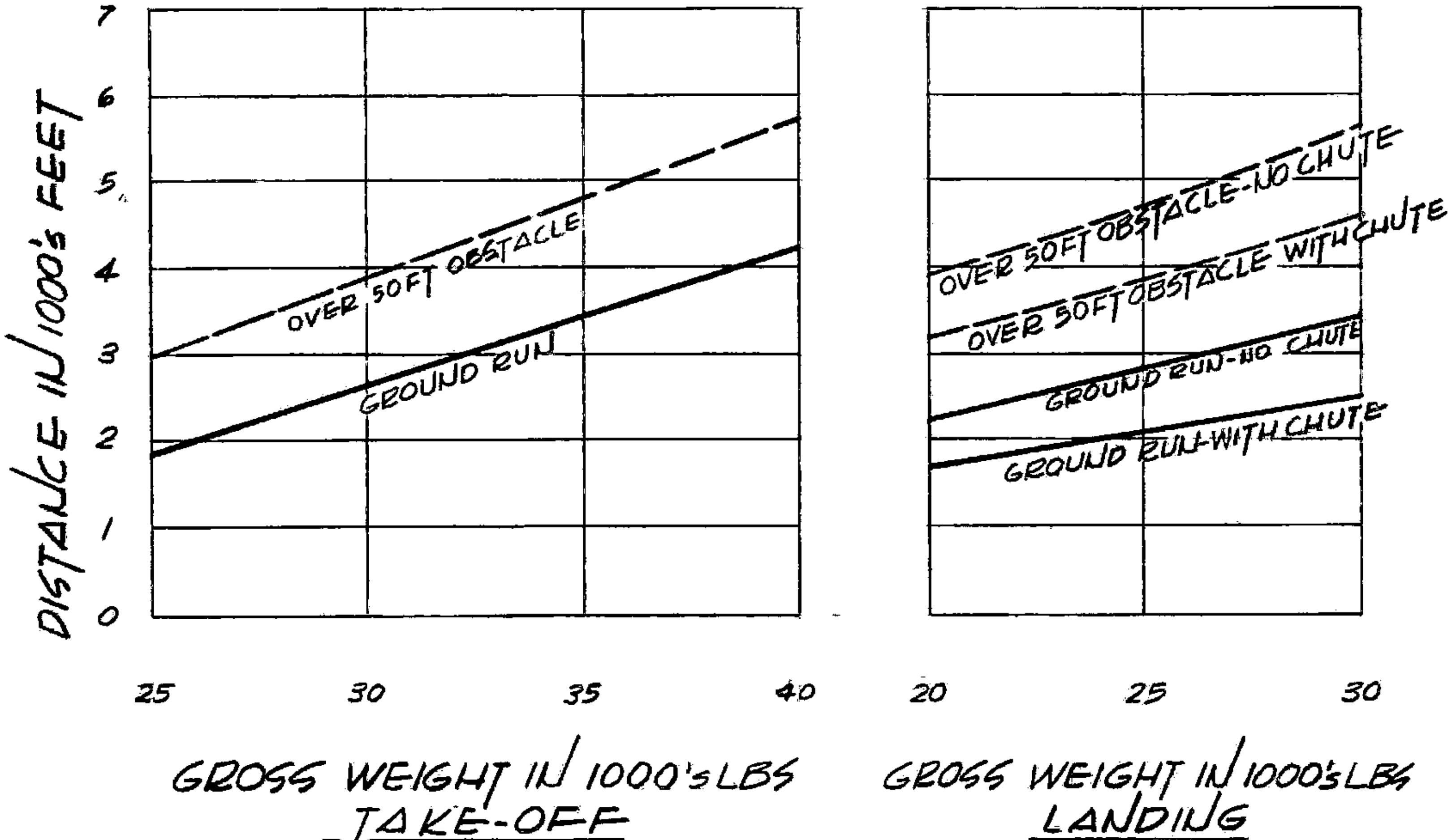
The additional fuel thus made available causes the take-off weight to increase. A typical value for the take-off weight would be 37,000 lbs. Even so, since the increase in gross weight is in fuel, the combat weight would increase little, if any.

The graphs on the following two pages show the approximate effect on the take-off performance and load factors of the increased weight.

[redacted]

TAKE-OFF AND LANDING CHARACTERISTICS

(J57-P23 ENGINE)



NOTE: J57-P55 ENGINE WILL IMPROVE T.O. CHARACTERISTICS APPROX 5%

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14

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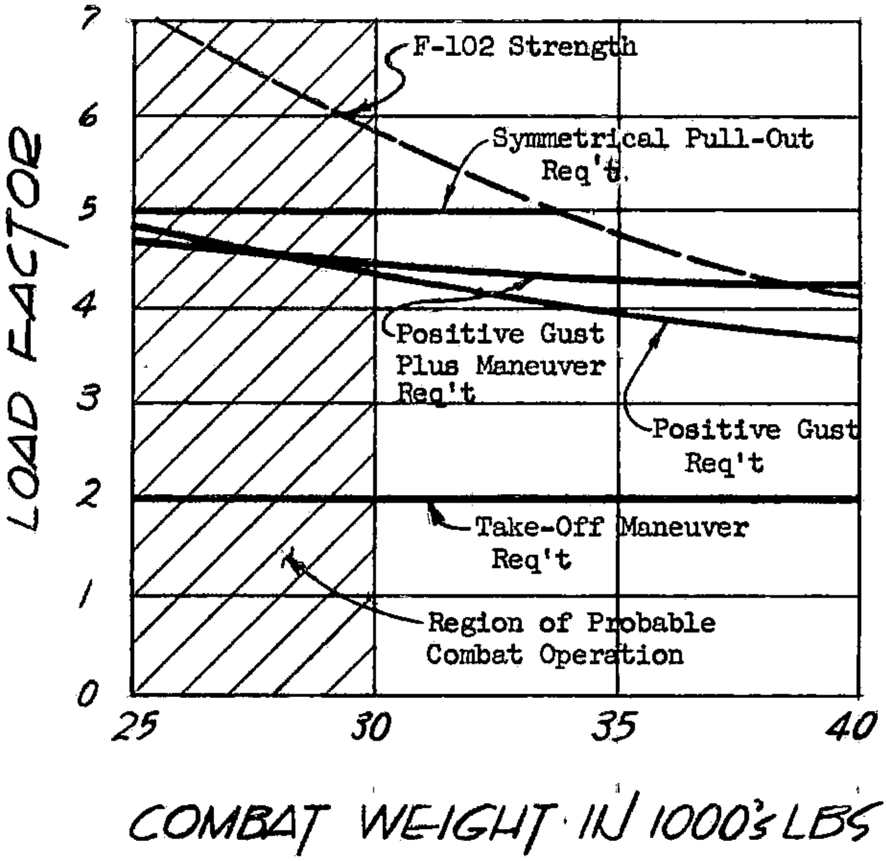
AVAILABLE STRENGTH VS TYPICAL REQUIREMENTS



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THE F-102 FIGHTER BOMBER

The F-102 makes an excellent Fighter Bomber. In addition to its fine performance and handling characteristics, the physical arrangement of the airplane makes it ideal for conversion to a Fighter Bomber.

In its nose the Interceptor carries a large amount (see Weight Summary, page 12) of radar and electronics fire control equipment. A modern fighter bomber carries similar equipment in this location. The interceptor missiles are carried in the bottom of the fuselage in armament bays near the airplanes' center of gravity. This space is readily adapted to carrying bombs and additional fuel. The design of the armament bay doors with integral rocket tubes would be retained.

The Fighter Bomber Study, Drawing SD-57-08054, shows the interior space fitted with a most modern bombing system wherein all possible functions are mechanized. Such a system would provide versatile all weather operation. It would include loft, toss, over-the-shoulder, level, and dive bombing modes as well as a limited air-to-air capability. A mode for air-to-ground rocketry would also be provided.

The communication, navigation and other systems used in the study have commensurate capabilities.

Carrying bombs internally stored on the airplane centerline is particularly important. The bombs produce no drag and therefore do not degrade the airplane's performance as do externally carried stores. Carriage of the weapons on the centerline eliminates the trim changes that arise from the unbalanced forces and moments produced during toss maneuvers on airplanes with unsymmetrically wing mounted bombs.

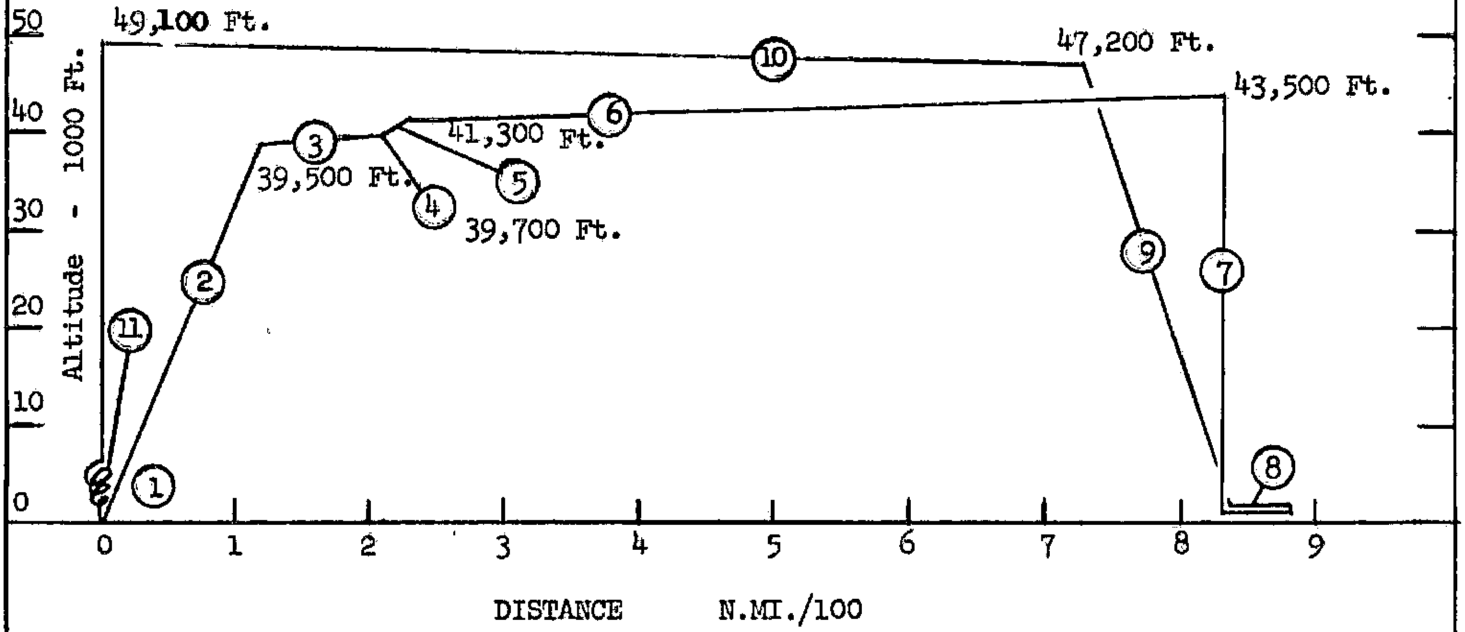
[redacted]

There are numerous combinations of equipment that could be used to provide a fire control system for the F-102 Fighter Bomber. The system envisioned above was chosen because it is the most sophisticated of those studied. It provides all the features that it is anticipated a customer could desire. As a consequence it sets the maximum space and weight requirement. Should a customer choose to have other fire control equipment installed, it is expected that there would be ample weight and space allowance available.

[redacted]

F-102A FIGHTER-BOMBER

MISSION A

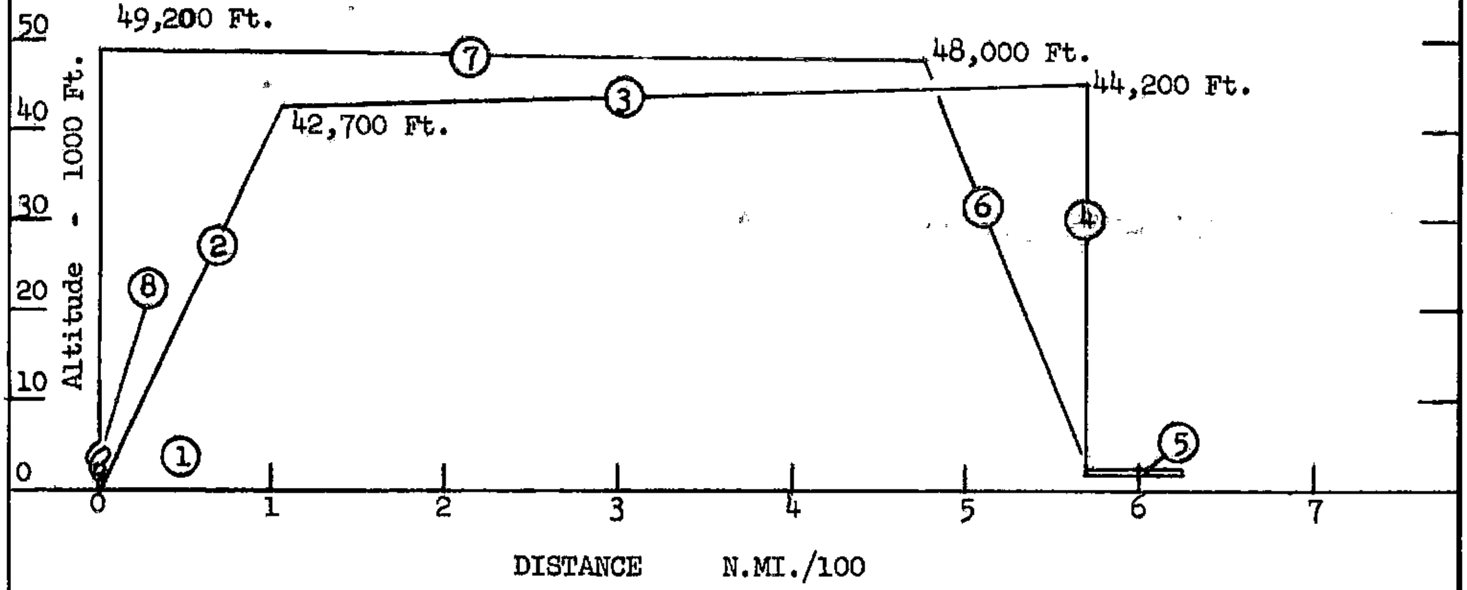


1. Take-off @ sea level. Take-off fuel allowance is 1 minute @ A/B power @ sea level plus 2 minutes @ normal power @ sea level.
2. Climb to cruise ceiling.
3. Cruise @ cruise ceiling until ext. tanks empty.
4. Drop external tanks.
5. Climb to clean airplane cruise ceiling.
6. Cruise @ clean airplane cruise ceiling.
7. Descend to sea level. (No distance credit or fuel penalty).
8. Penetrate 50 nautical miles @ military power and return.
9. Climb to cruise ceiling.
10. Cruise back @ cruise ceiling.
11. Reserve fuel allowance is 5% of initial fuel plus 20 minutes @ maximum endurance @ sea level.

[redacted]

F-102A FIGHTER BOMBER

MISSION B



1. Take-off @ Sea Level. Take-off fuel allowance is 1 minute @ A/B power @ sea level plus 2 minutes @ normal power @ sea level.
2. Climb to clean airplane cruise ceiling.
3. Cruise at clean airplane cruise ceiling.
4. Descend to sea level (no distance credit or fuel penalty).
5. Penetrate 50 nautical miles @ military power and return.
6. Climb to cruise ceiling.
7. Cruise back @ cruise ceiling.
8. Reserve fuel allowance is 5% of initial fuel plus 20 minutes @ maximum endurance @ sea level.

NOTE: In Mission "B" External (Droppable) Fuel Tanks are not carried as they were in Mission "A".

[redacted]

K+E KEUFFEL & ESSER CO. 10 X 10 TO THE CM. 359T-14LG
ALBANENE ® MADE IN U.S.A.

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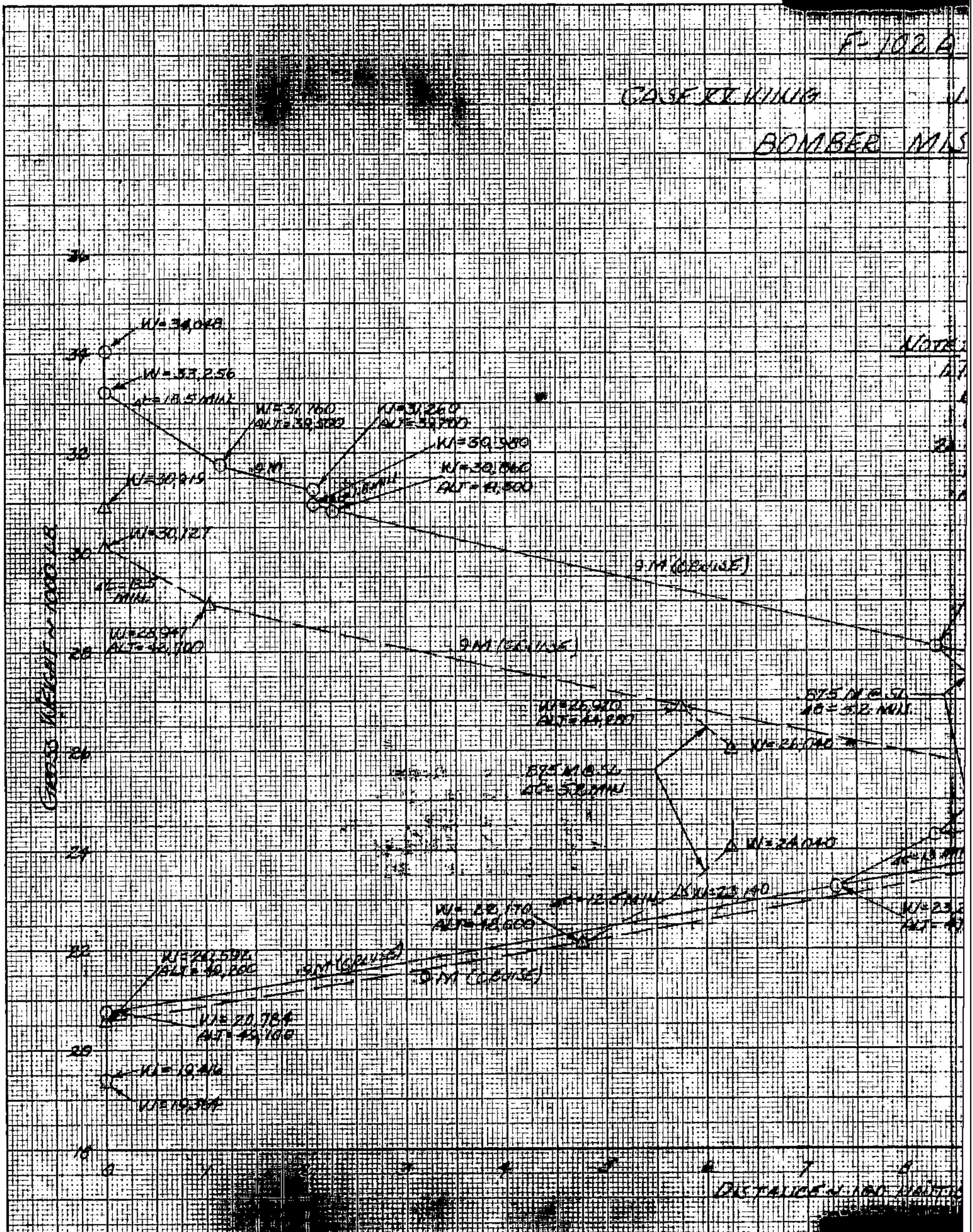
F-102A

CASE II WING

BOMBER MIS



[redacted]

2 B

J57P-23 ENGINE

MISSION

MISSION FLIGHT PLANS:

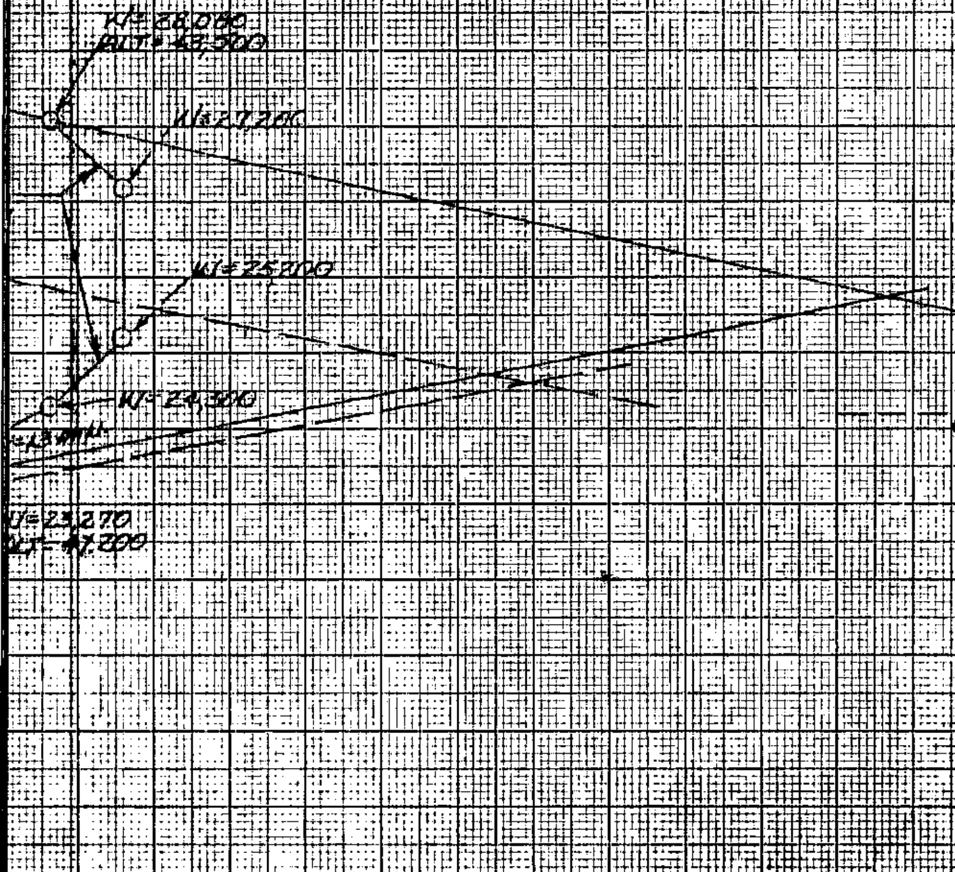
A. 2-230 GAL. EXTERNAL WING FUEL TANKS PLUS ADDITIONAL INTERNAL MISSILE BAY FUEL.

1. TAKE-OFF @ SEA LEVEL
2. CLIMB TO CRUISE CEILING
3. CRUISE @ CRUISE CEILING UNTIL EXT. TANKS EMPTY.
4. DROP EXT. TANKS.
5. CLIMB TO CLEAN AIRPLANE CRUISE CEILING.
6. CRUISE @ CLEAN AIRPLANE CRUISE CEILING.
7. DESCEND TO SEA LEVEL (NO DISTANCE CREDIT OR FUEL PENALTY.)
8. PENETRATE 50 NAUTICAL MILES @ MILITARY POWER AND RETURN.
9. CLIMB TO CRUISE CEILING
10. CRUISE BACK @ CRUISE CEILING.

B. ADDITIONAL INTERNAL MISSILE BAY FUEL.
OMIT ②, ③, & ④ FROM PLAN A.

NOTES:

1. TAKE-OFF FUEL ALLOWANCE IS 1 MINUTE @ A/B POWER @ SEA LEVEL PLUS 2 MINUTES @ NORMAL POWER @ SEA LEVEL.
2. RESERVE FUEL ALLOWANCE IS 5% OF INITIAL FUEL PLUS 20 MINUTES @ MAXIMUM ENDURANCE @ SEA LEVEL.



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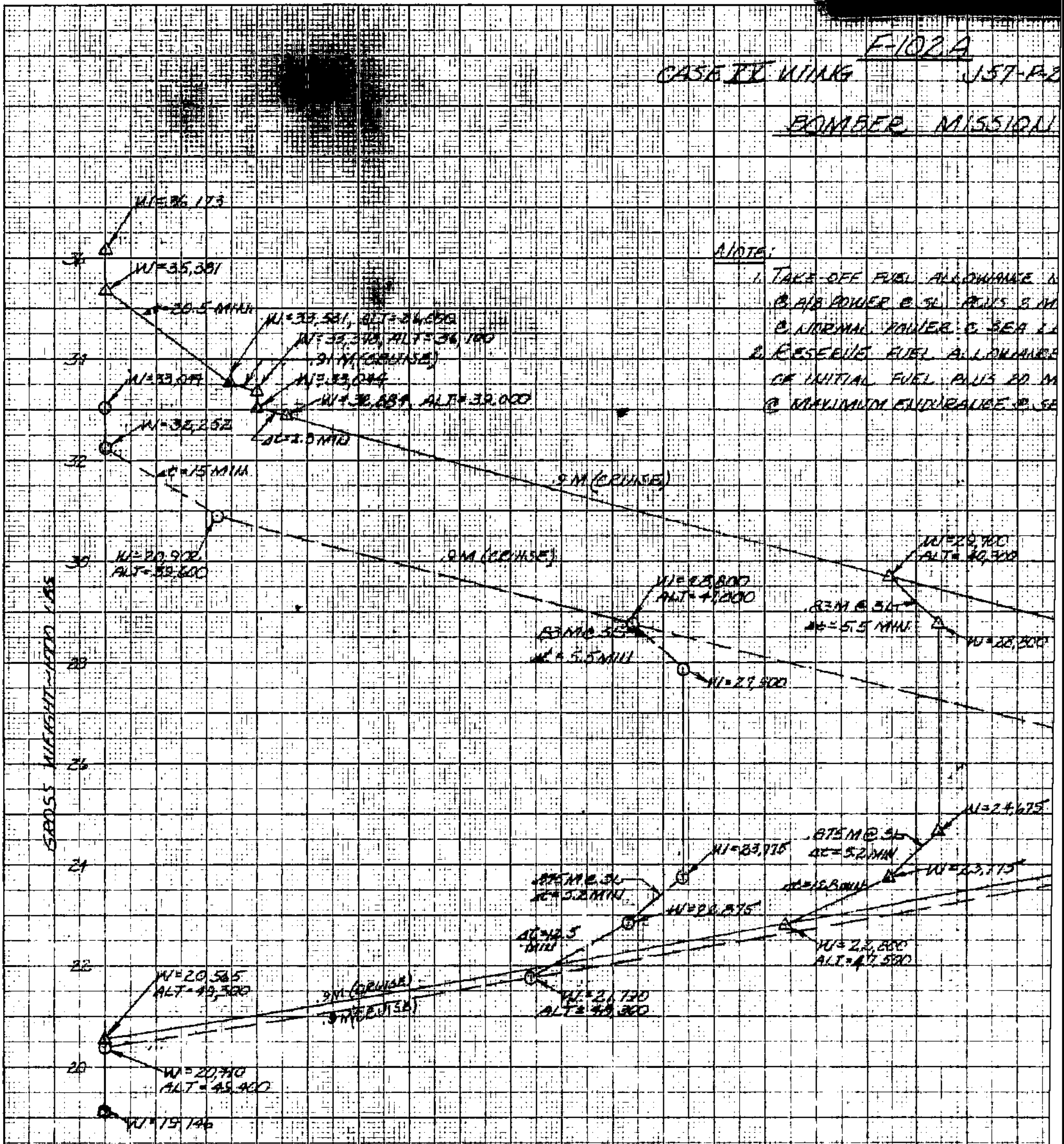


F-102A

CASE IV WING J57-P

BOMBER MISSION

NOTE:

1. TAKE-OFF FUEL ALLOWANCE
@ A/B POWER @ SL PLUS 3 M
@ NORMAL POWER @ SEA L

2. RESERVE FUEL ALLOWANCE
OF INITIAL FUEL PLUS 20 M
@ MAXIMUM ENDURANCE @ SE

[redacted]

-23 ENGINE

N

IS 1 MINUTE
MINUTES
LEVEL.
CE IS 5%
MINUTES
SEA LEVEL.

MISSION FLIGHT PLANS:

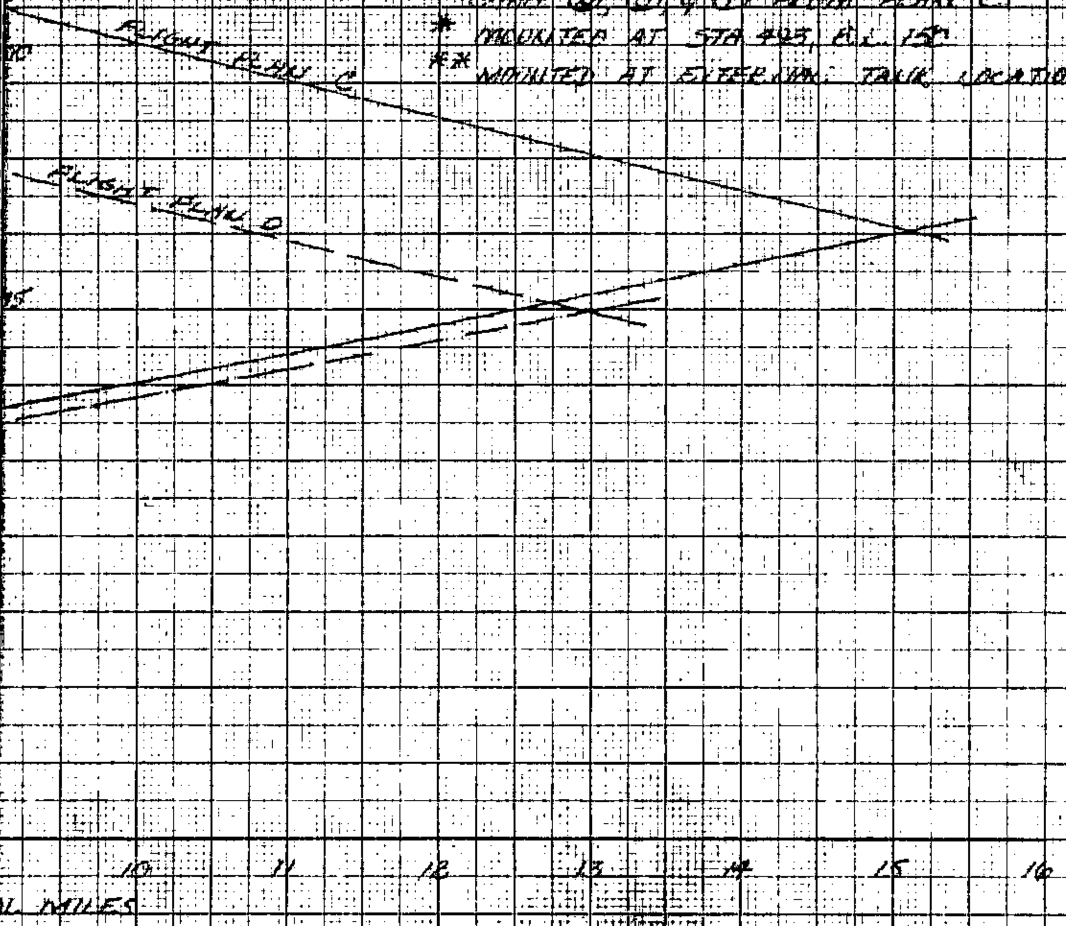
C. 2-230 GAL. EXTERNAL WING FUEL TANKS AND 2-1000 LB GP BOMBS MOUNTED* EXTERNALLY

1. TAKE-OFF [illegible] LEVEL.
2. [illegible] CRUISE CEILING
3. [illegible] @ CRUISE CEILING UNTIL EXTERNAL TANKS EMPTY
4. DROP EXTERNAL TANKS.
5. CLIMB TO CRUISE CEILING WITH 2-1000 LB. BOMBS.
6. CRUISE AT CRUISE CEILING WITH 2-1000 LB. BOMBS.
7. DESCEND TO SEA LEVEL (NO DISTANCE CREDIT OR FUEL PENALTY)
8. PENETRATE 50 NAUTICAL MILES AT MILITARY POWER AND RETURN
9. CLIMB TO CRUISE CEILING.
10. CRUISE BACK @ CRUISE CEILING.

D. 2-1000 LB. S.A BOMBS MOUNTED** EXTERNALLY
OMIT ②, ③ & ④ FROM PLAN C.

* MOUNTED AT STA 425, B.L. 150

** MOUNTED AT EXTERNAL TANK LOCATION



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WEIGHT SUMMARY

F-102 FIGHTER BOMBER

Structure	8,394
Propulsion	6,631
Fixed Equipment	3,591
WEIGHT EMPTY	18,616
Fixed Useful Load	530
Internal Fuel-Usable	9,555
Armament	
Bombs	2,000
Rockets	216
Installation	2
NORMAL GROSS WEIGHT	30,919
External Tanks, Pylons, Trapped Fuel	334
External Fuel- Usable (430 Gal.)	2,795
LONG RANGE GROSS WEIGHT	34,048

[redacted]

F-102 FIGHTER BOMBER WITH EXTERNAL STORES

In designing the F-102, the philosphy has been to produce an airplane with sufficient internal space to accommodate all the necessary equipment and armament. The object, of course, is to have an airplane whose performance is not degraded when the armament, which in competitors aircraft is external, is carried.

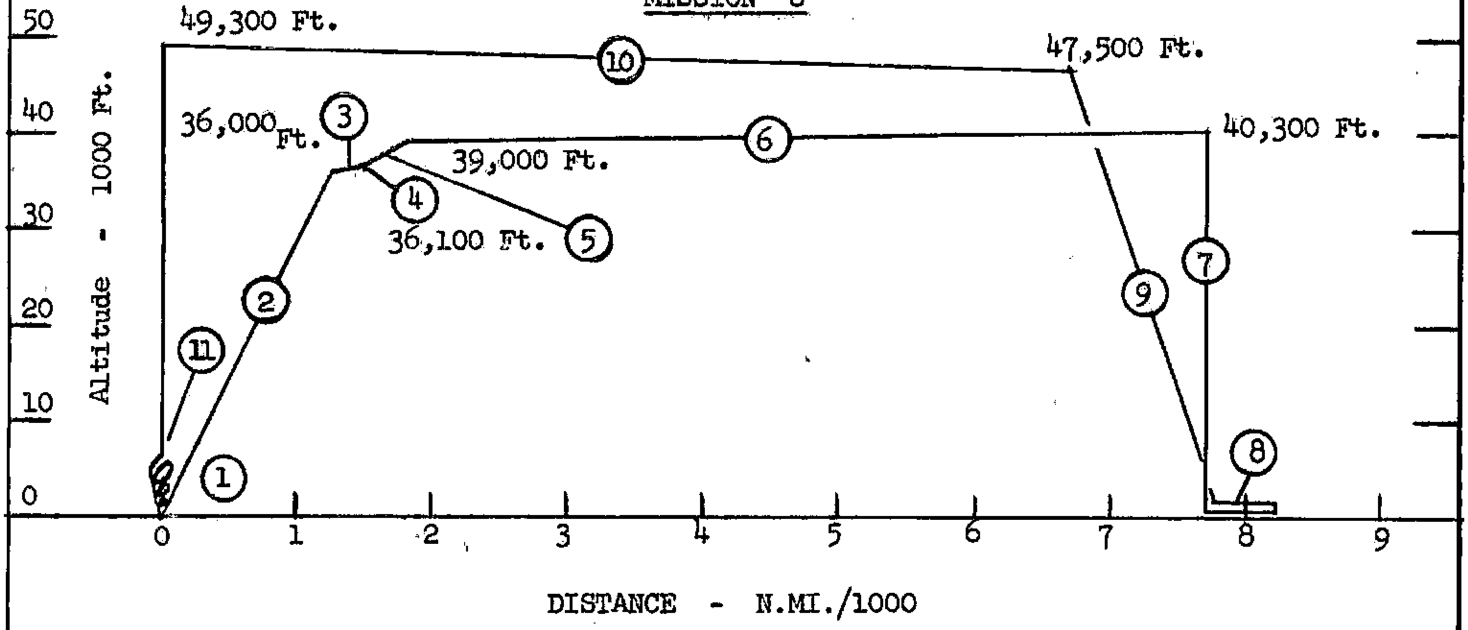
However, even after having followed this design philosphy, it is still possible to make provisions for carrying any number of types of weapons externally slung beneath the wing, and comparatively speaking, the performance is still excellent.

On the following pages the effect of carrying bombs simultaneously externally and internally, is illustrated. Missiles, rockets or guns could be carried in a similar manner.

[redacted]

F-102A FIGHTER-BOMBER

MISSION C

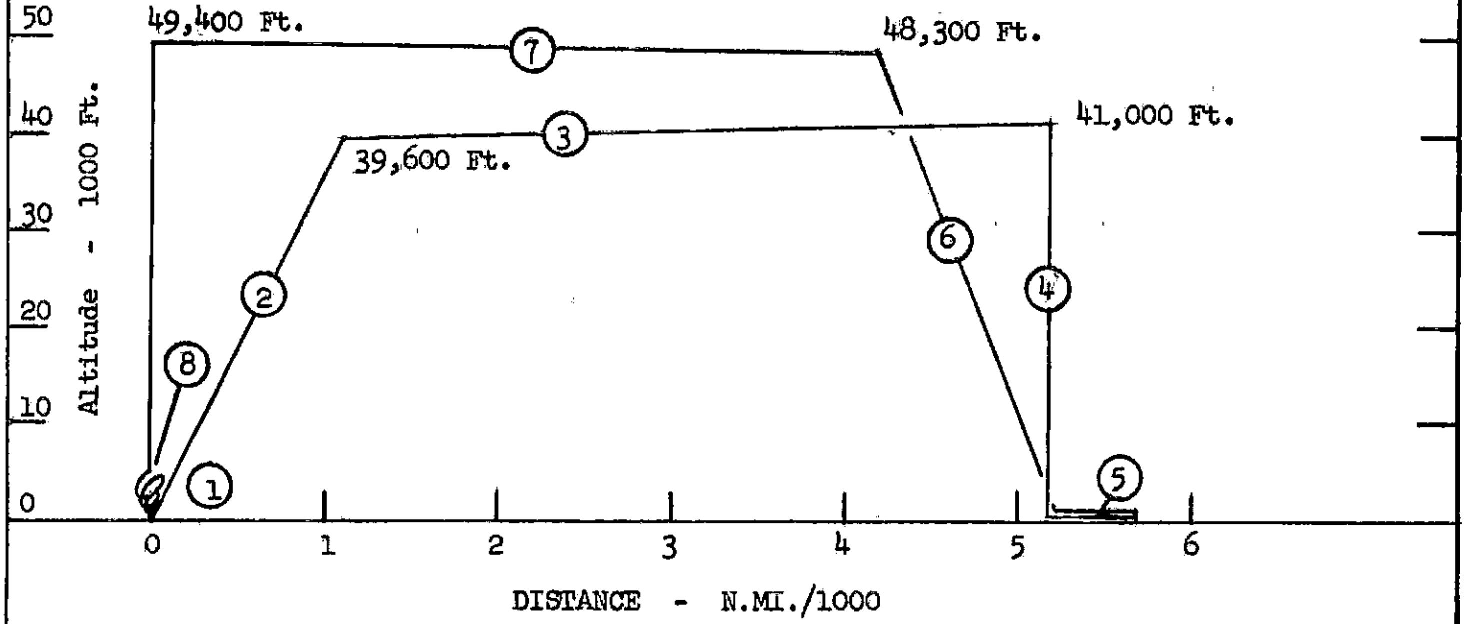


1. Take-off @ sea level. Take-off fuel allowance is 1 minute @ A/B power @ sea level plus 2 minutes, @ normal power @ sea level.
2. Climb to cruise ceiling.
3. Cruise @ cruise ceiling until external tanks empty.
4. Drop external tanks.
5. Climb to cruise ceiling with 2 - 1000 lb. bombs.
6. Cruise at cruise ceiling with 2 - 1000 lb. bombs.
7. Descend to sea level (no distance credit or fuel penalty).
8. Penetrate 50 nautical miles at military power and return.
9. Climb to cruise ceiling.
10. Cruise back @ cruise ceiling.
11. Reserve fuel allowance is 5% of initial fuel plus 20 minutes @ maximum endurance @ sea level.

[redacted]

F-102A FIGHTER-BOMBER

MISSION D



1. Take-off at sea level. Take-off fuel allowance is 1 minute @ A/B power @ sea level plus 2 minutes,@ normal power @ sea level.
2. Climb to cruise ceiling with 2 - 1000 lb. bombs.
3. Cruise at cruise ceiling with 2 - 1000 lb. bombs.
4. Descend to sea level (no distance credit or fuel penalty).
5. Penetrate 50 nautical miles at military power and return.
6. Climb to cruise ceiling.
7. Cruise back @ cruise ceiling.
8. Reserve fuel allowance is 5% of initial fuel plus 20 minutes @ maximum endurance @ sea level.

NOTE: In Mission "D" External (Droppable)Fuel Tanks are not carried as they were in Mission "C".

[redacted]

WEIGHT SUMMARY

F-102 FIGHTER BOMBER

(with external stores)

Structure	8,394
Propulsion	6,631
Fixed Equipment	3,591
WEIGHT EMPTY	18,616
Fixed Useful Load	530
Internal Fuel-Usable	9,555
Armament	
Bombs	2,000
Rockets	216
Installation	2
NORMAL GROSS WEIGHT	30,919
External Tanks, Pylons, Trapped Fuel	334
External Fuel-Usable (430 Gal.)	2,795
LONG RANGE GROSS WEIGHT	34,048
External Armament	
(2) 1000 Lb. Bombs	2,000
Pylons & Ejection Mech.	125
LONG RANGE GROSS WEIGHT with EXT. ARMAMENT	36,173

[redacted]

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TRAINING

Convair is prepared to conduct training programs to familiarize ground crews and pilots with the F-102 and its modes of operation to develop their skills for the levels demanded to successfully operate the airplane and its allied equipment.

SERVICE ORGANIZATION

To back up its training program and provide for unforeseen problems arising in the field Convair maintains a world wide service organization. Convair is capable of providing service anywhere in the world.

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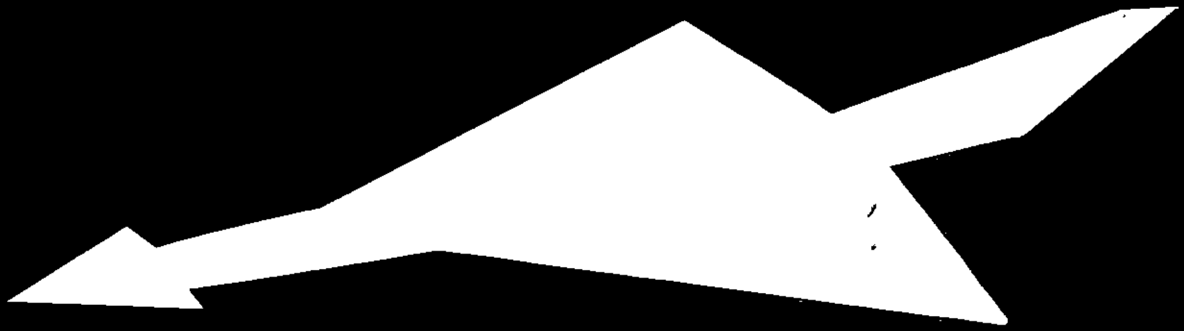

SUPERSONIC CRUISING ATTACK/RECONNAISSANCE WATER BASED AIRCRAFT - STUDY

CONTRACT NO. NOas 59-6018-c

REPORT NO. ZP-285

VOLUME I

31 JULY 1959

CONVAIR SAN DIEGO
A DIVISION OF GENERAL DYNAMICS CORP.

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A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-285
DATE July 31, 1959
MODEL PD-111

TITLE

VOLUME I
SUPERSONIC CRUISING ATTACK/RECONNAISSANCE (U) WATER BASED AIRCRAFT - STUDY

ZP-285

CONTRACT NOas 59-6018-c

FINAL REPORT

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GROUP Preliminary Design

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1.0 Summary - Volume I

The configuration studies performed under Bureau of Aeronautics Contract NOas 59-6018-c are outlined in this volume of the final report as performed on a supersonic cruising water-based attack/reconnaissance airplane.

The recommended configuration resulting from these studies is described in Section 4.0, Proposed Configuration. Design features of this Mach 4.0 Attack/Reconnaissance Seaplane include:

1. Delta type wing of low Aspect Ratio;
2. Tilt wing-hull combination;
3. Axisymmetric internal compression inlet;
4. Individually mounted engines;
5. Canard control surfaces;
6. Extensible ventrals also serve as floats.

The design lift-off weight for a Mach 4.0 cruise mission radius of 2,000 nautical miles and a military payload of 10,000 lbs. is 189,000 lbs.

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2.0 Summary - Volume II

The component selection studies performed under Bureau of Aeronautics Contract NOas 59-6018-c are outlined in this section of the report as applicable to the supersonic cruising-water-based attack/reconnaissance airplane.

Results of wing planform studies indicate an optimum wing with leading edge sweep of 55°, AR = 1.0, taper ratio of zero and a thickness of less than 2%. Modifications necessary to improve low speed performance result in higher taper ratio and leading edge sweep at Mach 4.0 and at Mach 4.75.

Results of the propulsion system studies indicate an optimum selection (minimum take-off weight) of the W.A.D. SE-109 turboramjet engine with a relative ramjet sizing of 0.65 times the nominal size.

Sustention systems are compared with consideration given to resistance, landing impact and system complexities.

Material evaluation studies with respect to environmental exposure, life and corrosion properties, and structural weight factors result in the selection of Ti-4Al-3Mo-1V alloy for the Mach 4.0 application and Rene' 41 at Mach 4.75.

Data is also summarized on the relative effects of methods of providing aircraft trim and on the effects of fuselage shapes.

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3.0 Introduction

The studies summarized in this report were conducted under Bureau of Aeronautics Contract NOas 59-6018-c, "Analytic and Design Study of a Supersonic Cruising Water-based Attack/Reconnaissance Aircraft".

This contract, covering the period from 12 November 1958 to 31 July 1959, stipulated a study to evaluate the basic aircraft parameters required for a supersonic attack/reconnaissance water-based aircraft capable of flight in the speed regime from Mach 4.0 through Mach 5.0 at altitudes from 60,000 to 100,000 feet.

In fulfilling this objective, the study was logically divided into two phases. The first phase, chronologically, was the analysis and study of suitable propulsion system installations, wing, body and tail configurations, and hydrodynamic sustention systems. The second phase consisted of the integration of the optimum results of Phase I into a series of realistic configurations and the comparative evaluation of these configurations.

Phase I - Component Selection Studies

The Component Selection Studies, which are summarized in Volume II of this report, include the investigations of propulsion systems, wing planform characteristics, and hydrodynamic sustention gear most suited to high speed cruising application. Consideration was also given to the effects on the cruise performance of body shapes and the method of trim control.

Supporting data generated, such as thermal environment, structural design criteria, material selection, etc. are summarized as part of the Component Selection Studies.

In general, the criteria utilized for selection of the optimum component was that of minimum weight and/or maximum range. For example, the engine selected (SE-109) under the propulsion system studies, was the one that realized the minimum gross take-off weight. In the case of the wing planform studies, the selection was on the basis of a maximum Breguet type range parameter.

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In other cases, the selection is dependent on the specific configuration concept. To afford a selection and comparison basis for these applications, generalized data was prepared around a spectrum of limit parameters. Such was the case in the hydrodynamic sustention system study, where the data was readily adaptable to the configuration studies.

Phase II - Configuration Studies

The basic configuration studies were conducted for a 2,000 nautical mile mission radius with a military payload provision of 10,000 pounds. Mission performance was considered only with hydrocarbon fuels.

Because of the water-basing requirement (Sea State 2), the utilization and integration of the optimum components from the Selection Studies often resulted in modification and compromise of these components.

Four configuration concepts were developed for comparison. These consisted of, at Mach 4.0: (A) submerged inlet, (B) roll over, (F) outboard tail, and (G) tilt-wing types. Additional variations were developed from concept A to determine effects with Mach number and Aspect Ratio changes. These consisted of: (C) Design Speed M = 3.0, (D) Design Speed M = 4.75, and (E) Design Speed M = 4.0, AR = 1.75.

Thrust loading and wing loading variations were performed by means of an IBM 650 Synthesis Program to determine the optimum conditions leading to minimum weight for the configuration.

These results, in conjunction with an assessment of the problems peculiar to each configuration were used as a basis for selecting the recommended configuration.

Other synthesis studies were conducted to determine the effects of:

1. Payload variation (20,000 and 30,000 lb. military load).
2. Structural and systems weight variation.
3. Difference in drag.
4. Design radius.
5. Design radius with high energy fuel.

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4.0
PROPOSED CONFIGURATION

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4.0 Proposed Configuration

The three view of Figure 4.0-1 represents the "most optimum" Mach 4.0 Airplane derived under the analytic and design studies performed on the Supersonic Cruising Water-based Attack/Reconnaissance.

Design features include:

1. Delta type wing of low Aspect Ratio;
2. Tilt wing-hull combination;
3. Axisymmetric internal compression inlet;
4. Individually mounted engines;
5. Canard control surfaces;
6. Extensible ventrals also serve as floats.

Advantages derived from these design features are numerous. The engine-wing arrangement, with an aft pivot point results in a seaplane engine-inlet-exit location that is virtually free of salt water ingestion. The pivot point is located aft of the trailing edge by means of an internally housed chordwise beam which, as well as distributing the chordwise bending loads, provides housing for the crew compartment and certain aircraft equipment.

An aft pivot point combined with a swept forward trailing edge permits maximum static submergence for a floating type seaplane thereby approaching the minimum aerodynamically desired fuselage volume.

The singly mounted engine arrangement permits utilization of the axisymmetric internal compression type inlet with its potentially higher inlet pressure recovery at design Mach number and also with minimum cowl drag. The full circle passages of this type inlet are also highly desirable from a structural arrangement standpoint in dealing with the sustained inlet temperatures and pressures of Mach 4.0 cruise operation. Maintenance problems are eased with individually mounted engines and localized hot spots associated with stacked engines are eliminated.

Certain advantages are also derived in terms of equipment installation as noted from Figure 4.0-2 (Inboard Profile). Visual surfaces of the crew compartment and the crew compartment itself must be thermally isolated from the imposed temperature environment. This is easily achieved in this design by integrating the crew compartment into the tilting wing.

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At design speed, during which time the aircraft will be under automatic control with radar and inertial inputs, forward vision is absent. The forward vision task during cruise, primarily that of collision avoidance, can best be handled electronically since for a reasonably high probability of detection, the time of visual detection range at design speed will approximate the pilot-aircraft reaction time.

Side and downward vision will be provided at design speed, the extent of which will be determined by progress in high temperature optical materials. Forward vision at design speed to fulfill the psychological need could be provided by indirect vision means.

At low speed, for take-off, approach and landing, the tilt wing located crew compartment yields the maximum possible visibility with direct vision over the aircraft nose at angles well in excess of normal approach and landing altitudes. This location of the crew near the C.G. also reduces the severity of the transient vibrations transmitted during high speed water operation.

4.0.1 Design Requirements

The basic design requirements for this configuration that were contractually specified and/or discussed with the contracting agency are:

1) Sustained cruise in the flight regime from Mach 4.0 to 5.0 at altitudes from 60,000 to 100,000 ft.;
2) Attack/Reconnaissance Mission of 2,000 nautical mile radius;
3) Water Based Operation - Sea State 2;
4) 10,000 lb. military load; and
5) 3.0g combat load factor and 2.5g load factor at take-off.

4.0.2 Configuration Parameters

The specific design parameters derived to fulfill these requirements are outlined in the following table (Table 4.0-I). The particular portion of the study wherein these parameters were derived, and the corresponding section of this report discussing the parameter selection are both noted in the table also.

Weight estimates for the proposed configuration are given in Table 4.0-II.

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TABLE 4.0-I

TILT WING DESIGN FACTORS

Design Parameter(s)	Value	Study Leading to Selection of	Report Section
Wing Planform	-	Wing Selection Study	Vol. II, Para. 7.1.4
Wing Loading Start of Cruise (Take-off)	55 (72)	Tilt Wing Synthesis Analysis	Vol. I, Para. 5.4.1
Engine	SE-109 Turboramjet	Propulsion System Selection Study	Vol. II, Para. 7.2.3
Relative Ramjet Size Engine Size	.65 X Nominal Cruise Power Setting = .85	Propulsion System Selection Study Tilt Wing Synthesis Analysis (Thrust to weight ratio at lift-off requirement yields 322 pps engine with SLS thrust of 31,300 lbs. for 0.90 static pressure recovery)	Vol. II, Para. 7.2.3 Vol. I, Para. 5.4.1
Structure Design Life Material	 1,800 hrs. Titanium Ti-4Al-3Mo-1V	 Design Criteria Material Section	 Vol. II, Para. 8.3.2 Vol. II, Para. 8.4.0
Configuration	Tilt Wing	Configuration Selection	Vol. I, Para. 5.3.3
Design Speed	M = 4.0	Design Speed Comparison	Vol. I, Para. 5.3.4

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M=4.0 Attack
Reconnaissance
Seaplane

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TABLE 4.0-II MACH 4.0 TILT WING - WEIGHT SUMMARY

Structure	37,030
Wing	12,100
Tail and Ventrals	3,130
Canard	1,500
Hull	13,100
Sustention	3,700
Surface Controls	1,400
Nacelle	2,100
Propulsion	12,910
Engine	8,060
Air Induction	2,100
Cooling	100
Lube System	100
Fuel System	2,100
Engine Controls	100
Starting	100
Water Injection	250
Equipment	7,700
Instruments and Navigation	300
Hydraulics, Pneumatics and Electrical	1,900
Communications and Navigation Electronics	1,500
Armament Installation	800
Air Conditioning	2,000
Auxiliary Gear	100
Furnishings and Equipment	1,100
Military Load	10,000
Electronics	4,160
Armament	5,000
Armament Auxiliaries	300
Crew	540
Useful Load	121,360
Oil	100
Fuel	120,860
Trapped Fuel and Oil	400
Total (Lift Off Weight)	189,000
Take-off Allowance	7,500
Fuel	2,300
Water	5,200
Ramp Weight	196,500
Weight Empty	61,800

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4.1 Mission Performance

4.1.1 Mach 4.0 Cruise Out and Return

The weight of the Mach 4.0 Attack/Reconnaissance Seaplane during the sustained cruise mission is shown in the Weight vs. Distance Plot of Figure 4.1-1. Altitude vs. Distance and Time vs. Distance are shown in Figure 4.1-2. The mission profile consists of a maximum power climb to design speed and altitude for L/D_{max} followed by $M = 4.0$ cruise at L/D_{max} with an engine power setting of 85% and descent and loiter. Fuel for 20 minutes loiter and 5% of initial fuel are maintained for reserves and SFC's are marked up by 1.05.

Distance credit is taken for the climb and descent with the stores released at mission midpoint of 2,000 nautical miles.

4.1.2 Mach 4.0 Cruise Out and Subsonic Return

The radius capability of the Mach 4.0 Attack/Reconnaissance Seaplane with Mach 4.0 cruise out, subsonic sea level combat and subsonic return is 1,680 nautical miles. The aircraft weight vs. distance is shown in Figure 4.1-3 for a return Mach number of 0.9 at altitude for L/D_{max}. The return Mach number has not been optimized for maximum distance. Altitude vs. Distance and Time vs. Distance are shown in Figure 4.1-4.

4.1.3 Mach 4.0 Mission with HEF

The mission profile of paragraph 4.1.1, with HEF used for ramjet cruise operation only, results in a radius capability of 2,585 nautical miles.

Synthesis results were obtained with HEF but are not included since the resulting weights were well below the limits considered acceptable for simulation by the synthesis equations.

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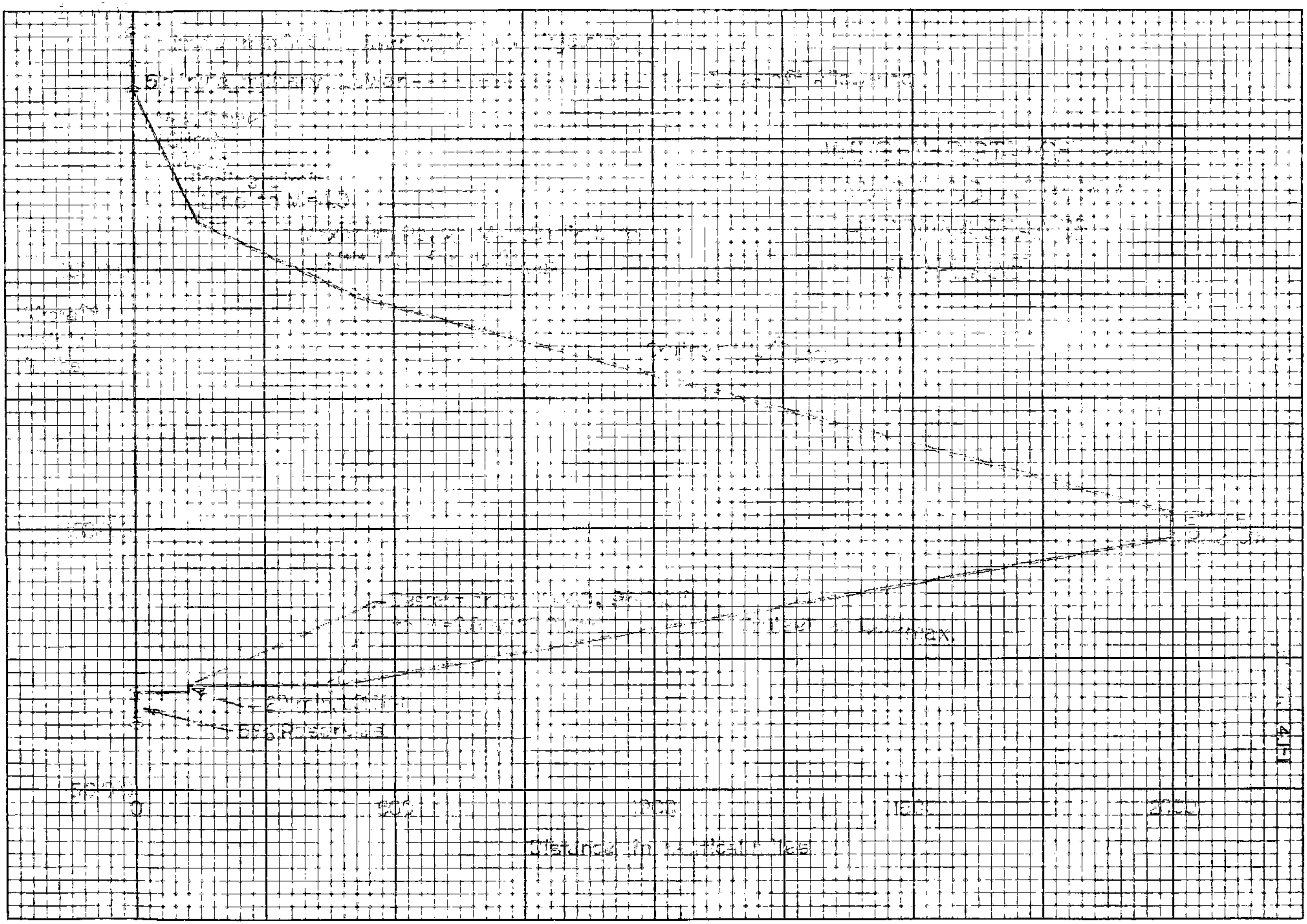

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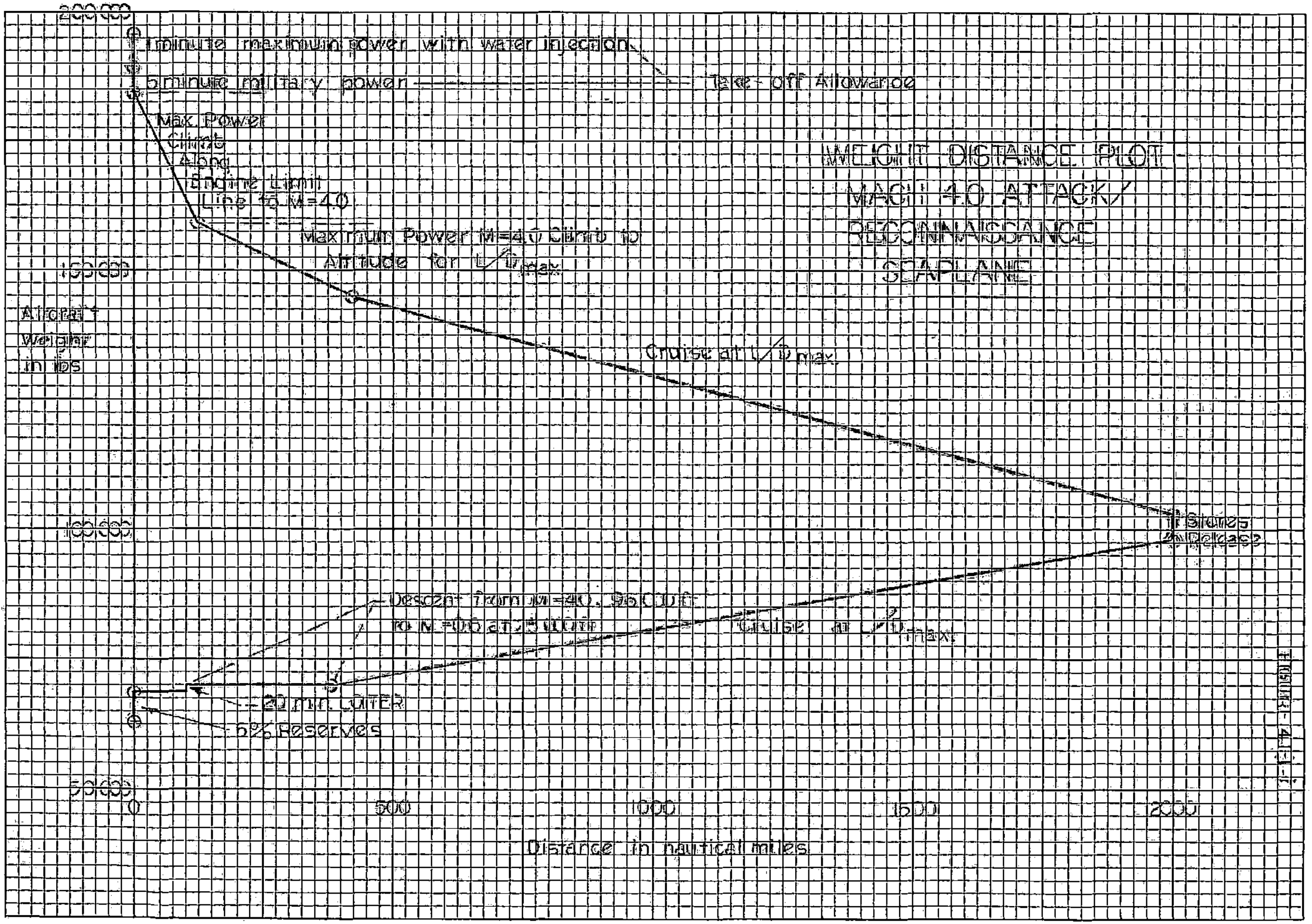
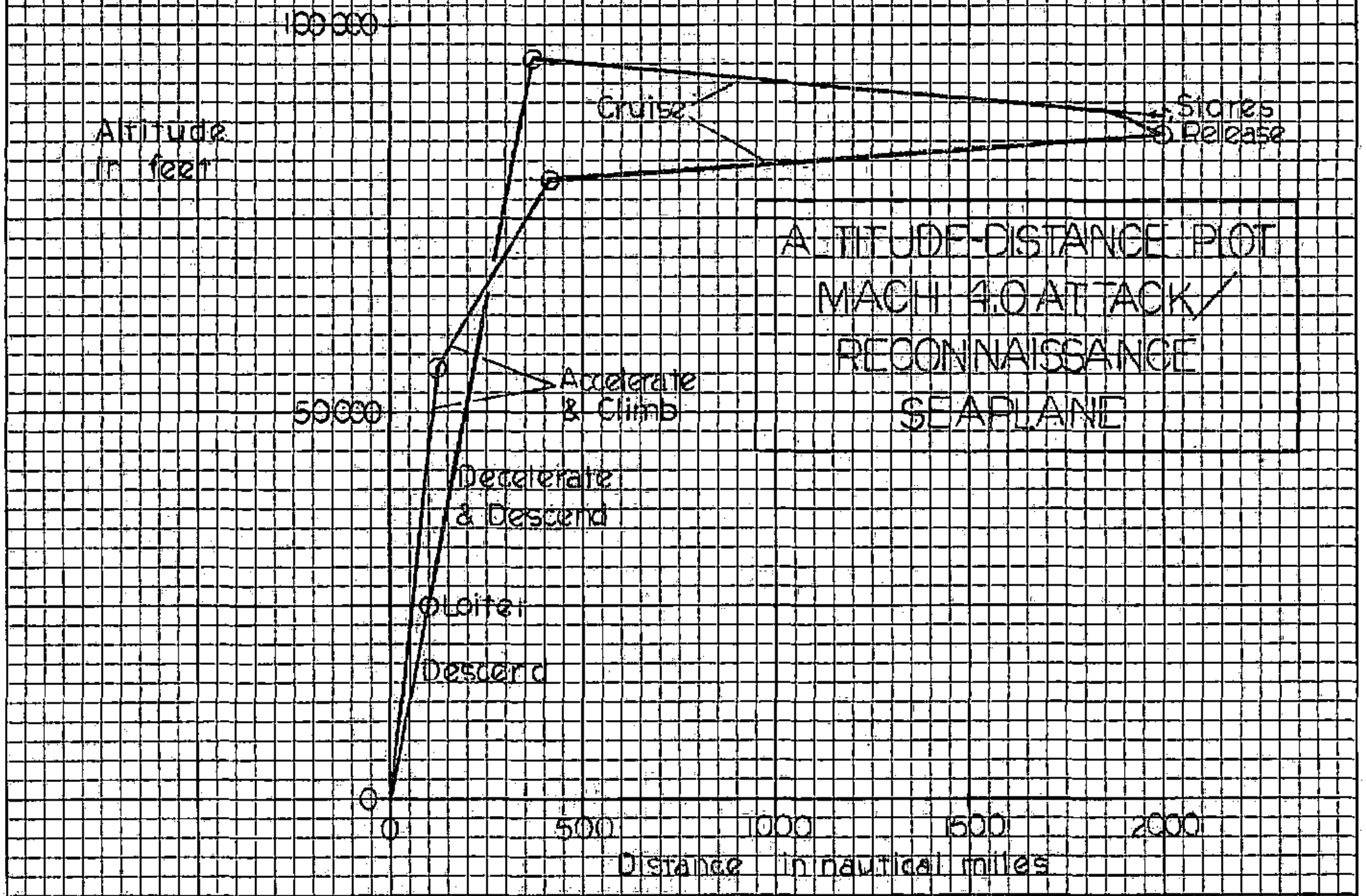


FIGURE 4.1-27



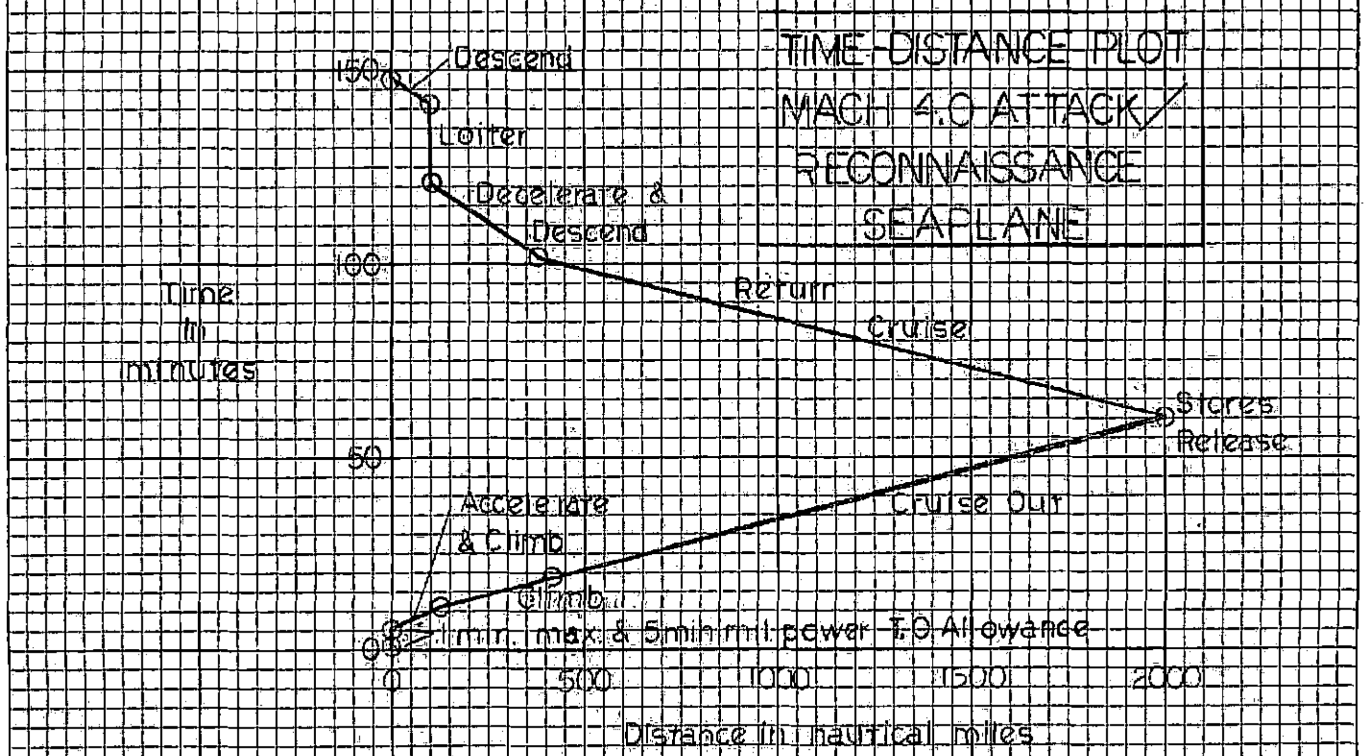


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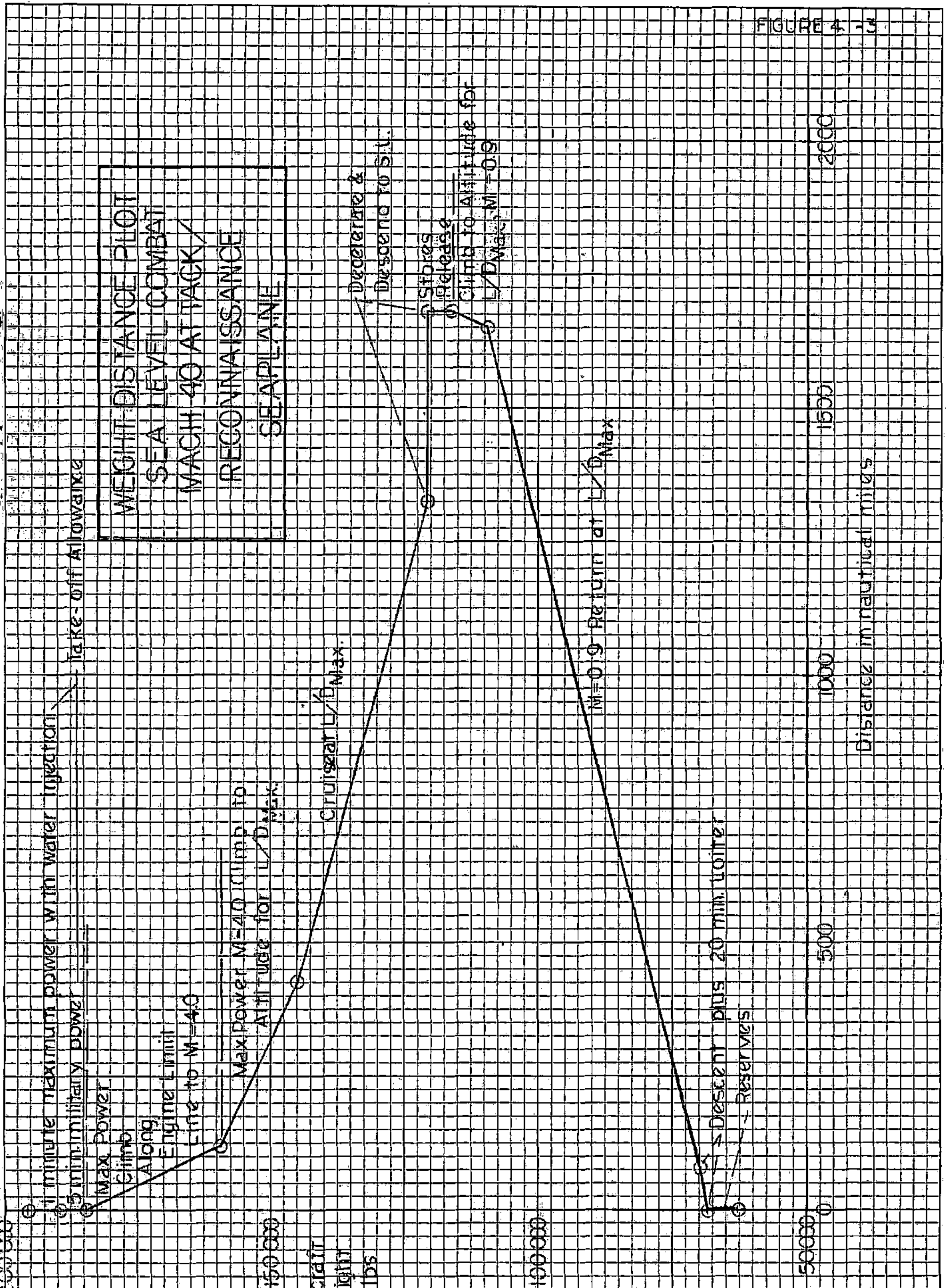
FIGURE 4-3



WEIGHT DISTANCE PLOT
SEA LEVEL COMBAT
MACH .40 ATTACK/
RECONNAISSANCE
SEAPLANE

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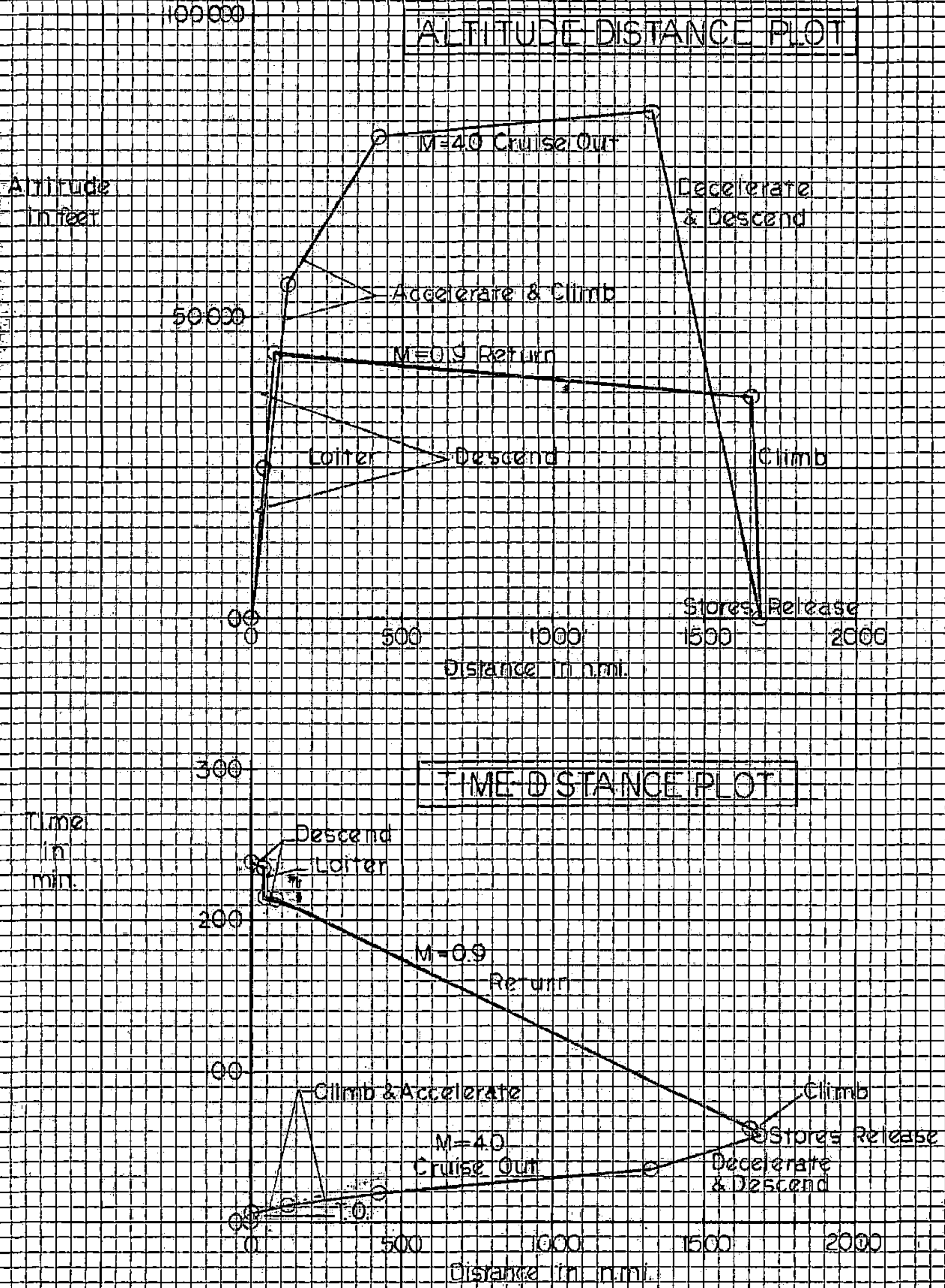
FIGURE 4.1-4

MAC-4.0 ATTACK RECONNAISSANCE SEAPLANE
SEA LEVEL COMBAT



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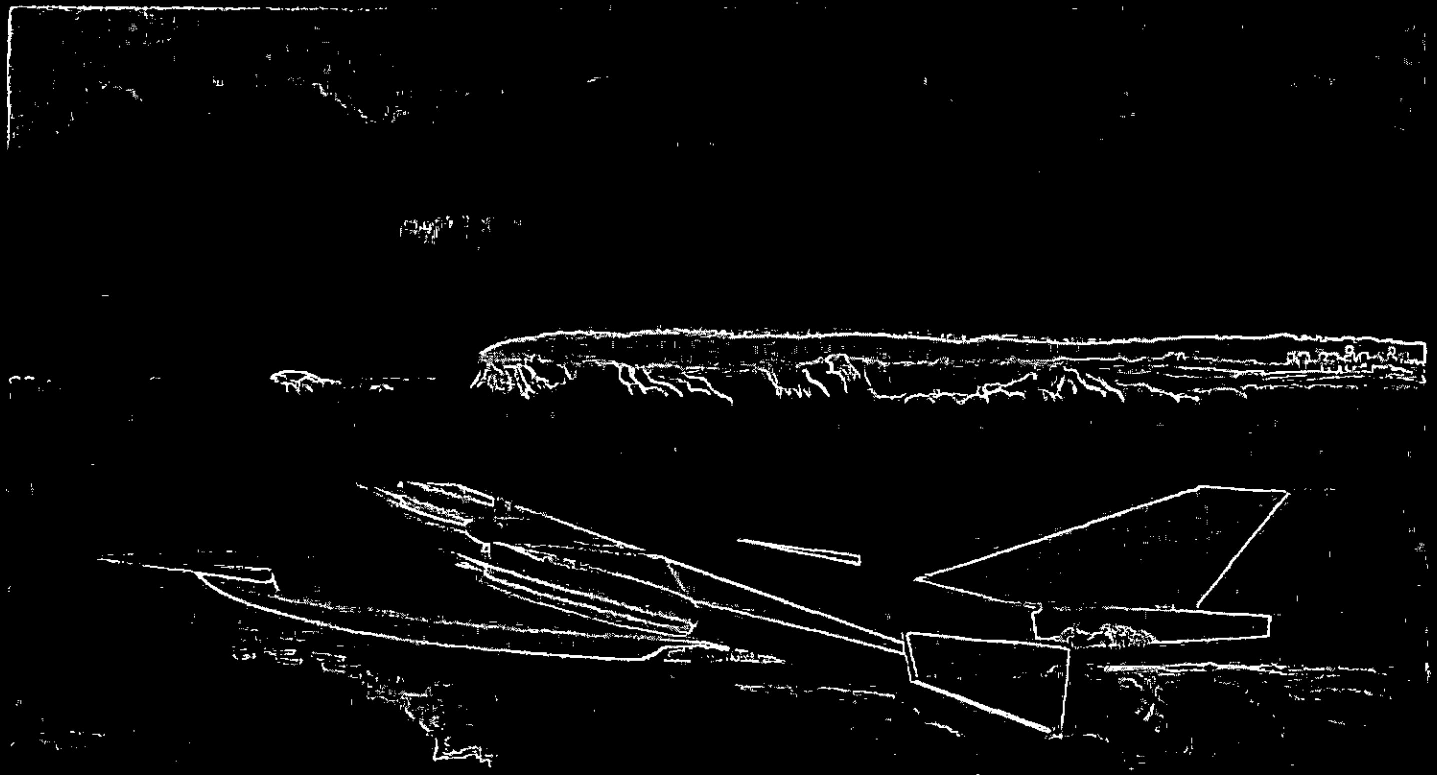
4.2

LOW SPEED CHARACTERISTICS



4.2 Low Speed Characteristics

The tilt wing concept relies on a high wing incidence to achieve take-off and landing lift thereby reducing requirements for high stern-post angles. The estimated low speed lift, drag and pitching moment characteristics of the configuration with and without wing incidence are shown in Figure 4.2-1 and are based on methods reported in references I-1 and I-2. The effect of wing lift carryover on the fuselage was omitted from estimates of ΔC_L and ΔC_m due to the high wing and large wing incidence. The pitching moment data includes an estimated $\Delta C_{m_0} = + .006$ due to the fuselage nose upsweep. The data shown are for a canard programmed to maintain constant angles of attack of 0°, 5° and 15° with respect to the free stream.

A take-off lift coefficient of $\approx .38$ is available at a body angle of 7° with no flaps and a required canard $\alpha = 3.5^\circ$. The effect of flaps was estimated using the span loading theory of reference I-3 and from experimental data on flap effectiveness on low aspect ratio wings. The combination of 20° of flap deflection (S_f/S_w) = .096, C_f/C_{mac} = .14), 9° wing incidence, and 7° fuselage angle results in an estimated C_L of 0.55 requiring 22° canard angle of attack for trim. Increased flap deflections will be limited due to water spray problems and will require canard boundary layer control to maintain effectiveness up to higher canard angles.

The incidence angle of the power plants contribute a considerable vertical component of lift, which for the thrust to weight ratio of this airplane corresponds to a $\Delta C_L \approx .05$. This effect is also shown on Figure 4.2-1.

With an effective lift coefficient of 0.6 and a lift-off wing loading of 72[illegible] psf, the take-off velocity is 187 knots. Resistance and trim variation with speed are shown in Figure 4.2-2. The take-off time is 58.7 seconds and distance is 9,271 ft. Minimum acceleration, which occurs at hump, is .128 g's. Basic factors used in deriving the take-off performance are as follows:

Thrust	=	71,000 lbs. (with water injection)
Ramp Weight	=	196,500 lbs.
Lift-off Weight	=	189,000 lbs.
(Thrust/Weight) at Start of Take-off	=	.363
(Thrust/Weight) at Lift-off	=	.378

$L_H/L_B = 27.5$ $L_F/L_A = .95$ $\beta_F = 30^\circ$	$C_{\Delta o} = 10.37$ $K_2 = .0137$

The shock-mounted step discussed in the Sustention System Study (Section 7.3) has not been incorporated into the proposed configuration since the weight penalty associated with the installation is much higher than the weights associated with the local beef-up required to withstand the higher impact load factors of the rigid hull with tilt wing. The relatively light landing weight at impact load factors will generally be less critical than for flight conditions particularly with honeycomb and sandwich type construction except for fittings, attachments, etc.

Further study of take-off loadings is required to determine the final trade-off of weights in this area.

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MACH 4.0 ATTACK/RECONNA

LOW SPEED CHARAC

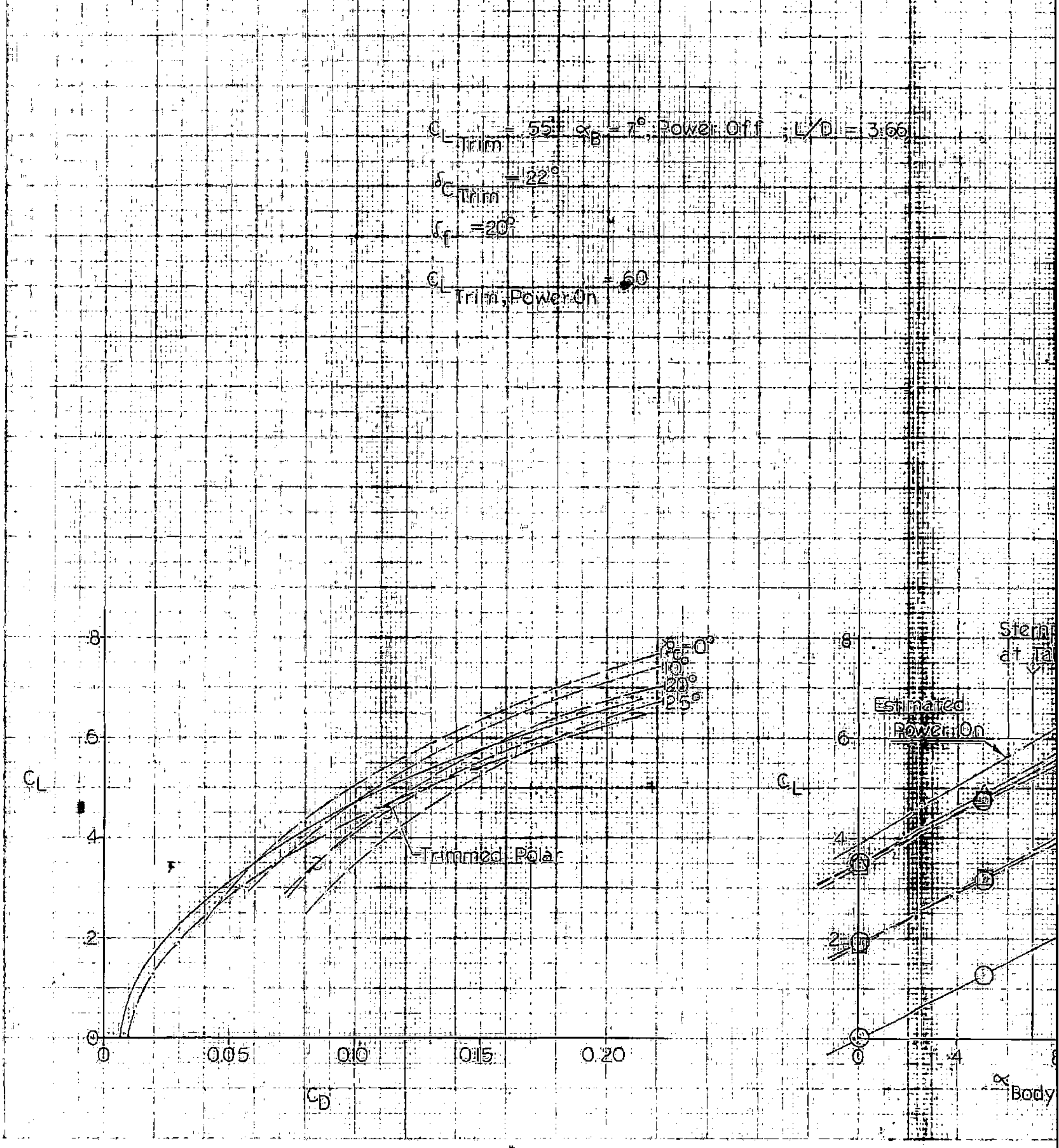
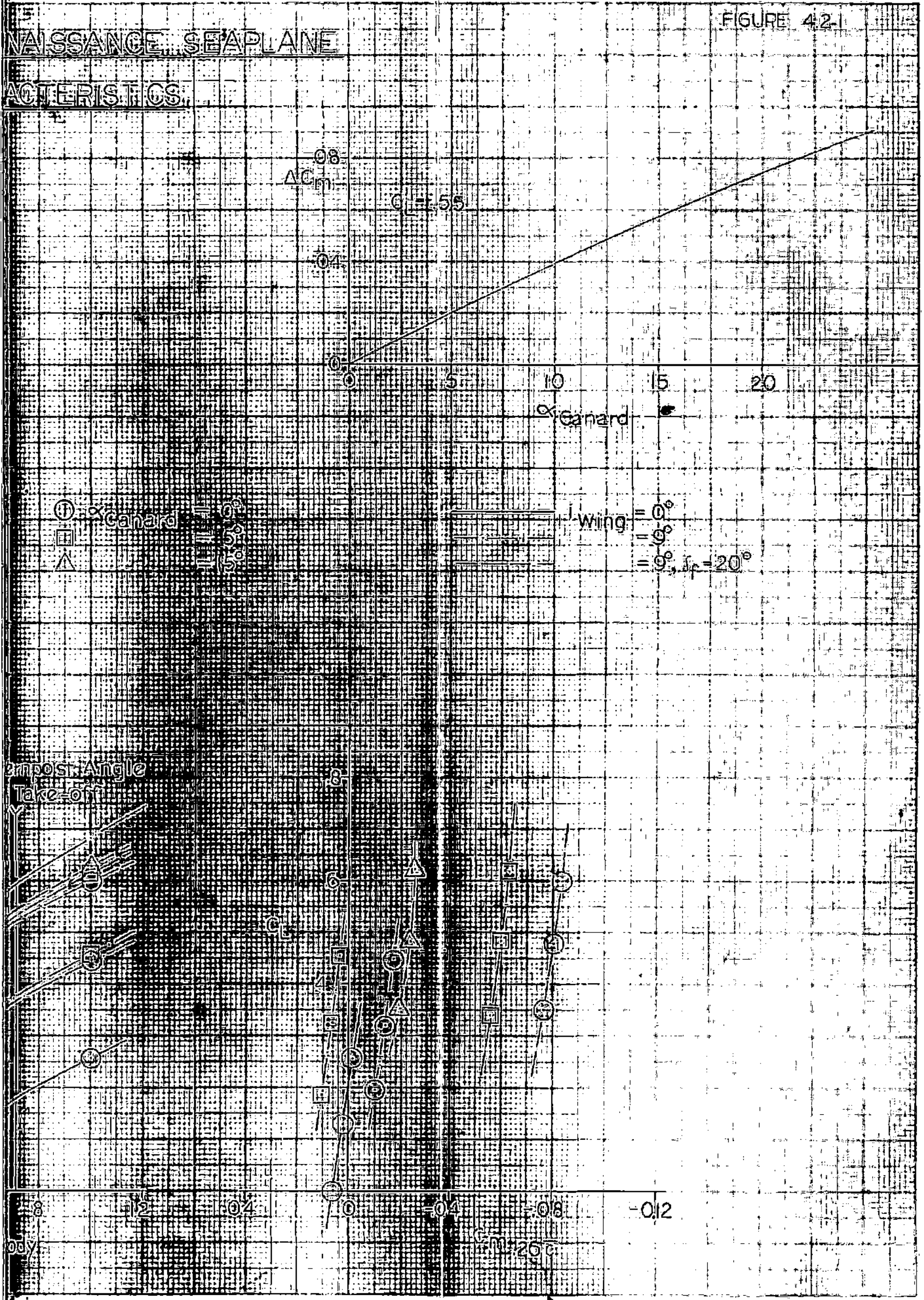


FIGURE 42-1


NAISSANCE SEAPLANE
ACTERISTICS
.08
ΔCm
CL = .55
.04
0
0
5
10
15
20
α Canard
① α Canard = 0°
⊡ = 5°
△ = 15°
i Wing = 0°
= 9°
= 9°, δf = 20°
ernpost Angle
Take-off
.8
.6
CL
.4
8
.12
.04
0
-.04
-.08
-0.12
ody
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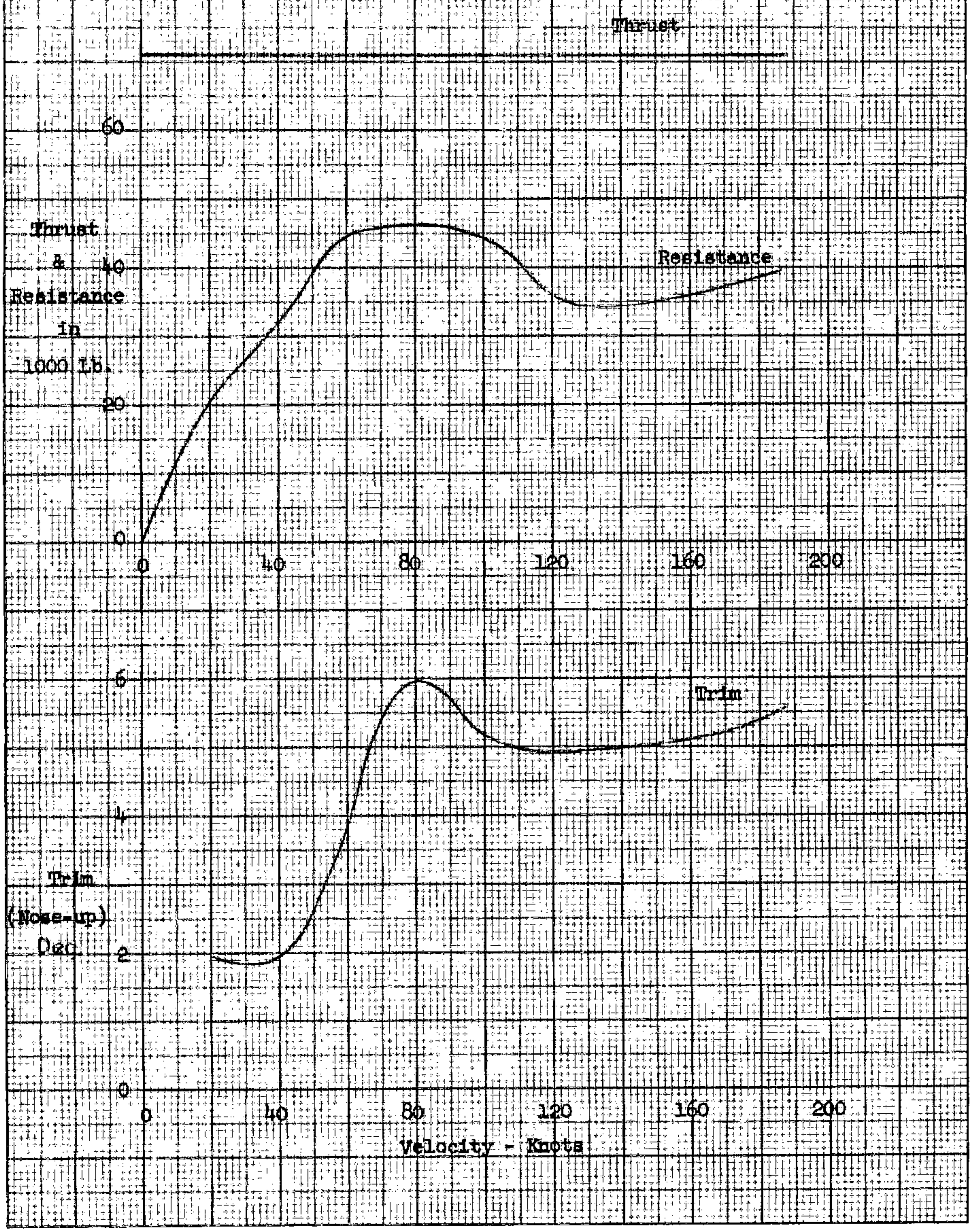
MACH 4.0 ATTACK/RECONNAISSANCE SEAPLANE

FIGURE 4.2-2

TAKE-OFF CHARACTERISTICS



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4.3 Design Speed Characteristics

Drag estimates on the Mach 4.0 Attack/Reconnaissance Seaplane were based on the sum of the estimates of skin friction drag, wave drag, and drag-due-to-lift.

Longitudinal stability characteristics were estimated based on linearized theory and checked using published results on several similar configurations.

4.3.1 Skin Friction Drag

The skin friction drag of the total configuration is computed as the sum of the friction drags of its components assuming a fully turbulent boundary layer. The component Reynolds numbers were calculated at several altitudes corresponding to the Mach number-altitude flight path in order to minimize the error due to large altitude variations in the proposed flight schedule.

Resulting values of skin friction parameters are as follows:

Component	Total Wetted Area - Sq. Ft.	Characteristic Length - Ft.
Wing	4,750	59.8
Fuselage	2,430	150.0
Pods and Pylon	1,240	27.6
End Plates	484	16.8
Vertical Tail	680	27.0
Canard	365	16.8
	9,949	

Von Karman's equation for the coefficient of skin friction with a fully turbulent boundary layer ($C_f = 0.455/(\log_{10}RN)^{2.58}$, reference I-4) was used to estimate the incompressible skin friction coefficients. The compressible skin friction coefficients were determined by the method described in reference I-5. The skin friction ratio C_f/C_{fi} was obtained from a curve plotted versus Mach number for various wall temperature ratios, T_W/T_1, where C_f is the average turbulent skin friction coefficient, C_{fi} is the incompressible turbulent skin friction coefficient, T_W is the wall temperature and T_1 is the initial temperature. The skin friction ratio increases with decreasing wall temperature ratio and decreases with increasing Mach number. This curve was obtained by using the T' method of Rubesin and Johnson and applying the Karman' Schoenher incompressible flow equation. The equilibrium surface temperature (T_W) was determined from reference I-6 for an all turbulent boundary layer and an emmisivity of 0.9, one foot aft of the leading edge.

The results of the analysis are as follows:

Mach Number	Altitude (Ft.)	$(R_N/ft)x10^{-6}$	C_{fi}	C_f/C_{fi}	C_f	$C_{D_{sf}}$
0.9	25,000	3.04	.00198	1.000	.001980	.00746
2.0	60,000	1.46	.00215	0.751	.001618	.00610
3.0	70,000	1.36	.00218	0.614	.00134	.00505
4.0	80,000	1.12	.00227	0.506	.001149	.00433
5.0	90,000	0.83	.00247	0.429	.001060	.00400

The skin friction drag is plotted versus Mach number on Figure 4.3-1.

4.3.2 Wave Drag

A wave drag correlation procedure (reference I-7) based on the configuration normal area distribution was used to estimate the wave drag at M = 1.0 and M = 1.2 and the supersonic wave drag variation with Mach number was determined from a comparison of available data on several similar configurations.

The maximum wave drag was calculated to be .0058. The variation with Mach number is also shown on Figure 4.3-1.

4.3.3 Drag-Due-To-Lift:

The subsonic drag-due-to-lift was estimated from available test data on low aspect ratio wings and wing-body combinations. A spanwise efficiency factor of 0.73 was determined which results in a K_{sub} of 0.467. The supersonic drag-due-to-lift (Figure 4.3-2) was estimated from an examination of data gathered in a correlation of various theories with experimental data (reference I-8), from the theoretical results of the Harmonic Method outlined in Volume II of this report and from test results of similar wings reported in the literature. No attempt was made to account for scale effects in predicting basic drag-due-to-lift.

4.3.4 Longitudinal Stability Characteristics

The longitudinal stability characteristics of the Mach 4.0 Attack/ Reconnaissance Seaplane were estimated based on linearized theory and checked using published results on several similar configurations. The component center of pressure variations with Mach number, up to a Mach number of ≈ 1.2, were estimated using the method of reference I-1. The estimated aerodynamic center variation with Mach number is shown in Figure 4.3-3 for the configuration with and without the canard surface. The theory indicates that the a. c. should continue to move aft with increasing supersonic Mach number as a result of wing body interference effects, while the opposite trend has been obtained experimentally on many configurations (Reference I-9 through I-11). Accordingly, the a.c. variation with Mach number for the wing-body combination in the range 1.2 - 4.0 was estimated from available test data on similar wing-body configurations and is shown as a dotted line in Figure 4.3-3. The forward shift in aerodynamic center due to the canard surface at supersonic speeds was determined from linearized theory.

As can be seen from the estimated a. c. curve for the complete configuration with a fixed canard, the allowable low speed center of gravity is forward of 15% mac. With a maximum forward fuel c.g. of 25% mac, a significant forward c. g. travel and attendent cruise trim drag penalty results requiring consideration of various means of altering the stability margin variation with Mach number.

Since from the standpoint of directional stability at high Mach numbers it is desirable to have some vertical surface outboard and/or under the wing, a natural solution to both the longitudinal and directional problems is folding wing tips. The estimated rearward a. c. shift subsonically with the addition of an outboard wing panel formed by extending the wing leading and trailing edge to a pointed tip is shown on Figure 4.3-3. With the tip folded 90° to the wing chord plane at high subsonic speeds and maintained there for high speed acceleration and cruise, the improvement in high speed directional stability is provided and the a. c. variation from subsonic to supersonic speeds is reduced. In addition to these benefits, the 55% increase in effective wing aspect ratio at subsonic speeds would improve the available lift at take-off and landing, and improve the subsonic cruise and loiter performance.

Although the use of folding wing tips appear to provide substantial aerodynamic benefits, the addition of an auxiliary device required to improve the static lateral stability of the configuration in the water is considerably complicated when folding wing tips are used. For this reason the proposed configuration is shown with fixed vertical surfaces at the wing tips for directional stability considerations and the longitudinal problem is solved by relying on the use of a "free floating canard" (or a canard to act as a free floating surface) for low speed flight. With this system the configuration longitudinal stability can be essentially that of the wing body combination except for the destabilizing effects of canard-body and canard-wing interference. One possible stability augmentation device which can be used with such a canard is the so-called "q - bob" which was studied in some detail for application to an advanced F-106 version (reference I-12). In this device an equivalent angle of attack signal (i.e., approximately equal to n_Z/q) is generated mechanically by a q-spring suspended bob-weight, and the canard driven based on this signal to provide the desired degree of stability. At some high subsonic Mach number the system then reverts to direct pilot control with the configuration stability determined by the fixed canard a. c. variation with Mach number shown in Figure 4.3-3.

The $M = 4.0$ cruise characteristics of the tilt-wing-fixed-canard configuration are shown in Figure 4.3-4. From available test data on the effects of fuselage nose deflection the configurations upswept nose was estimated to provide $\Delta C_{M_O} = +.007$ at $M = 4.0$. As can be seen from Figure 4.3-4, this is sufficient to provide trim at a $C_L = .10$ with a stability margin of 3%. The effect of canard deflection on lift, drag, and pitching moment is also shown in the figure.

Figure 4.3-1

MACH 40
ATTACK/RECONNAISSANCE
SEAPLANE

SKIN FRICTION AND WAVE DRAG VS. M

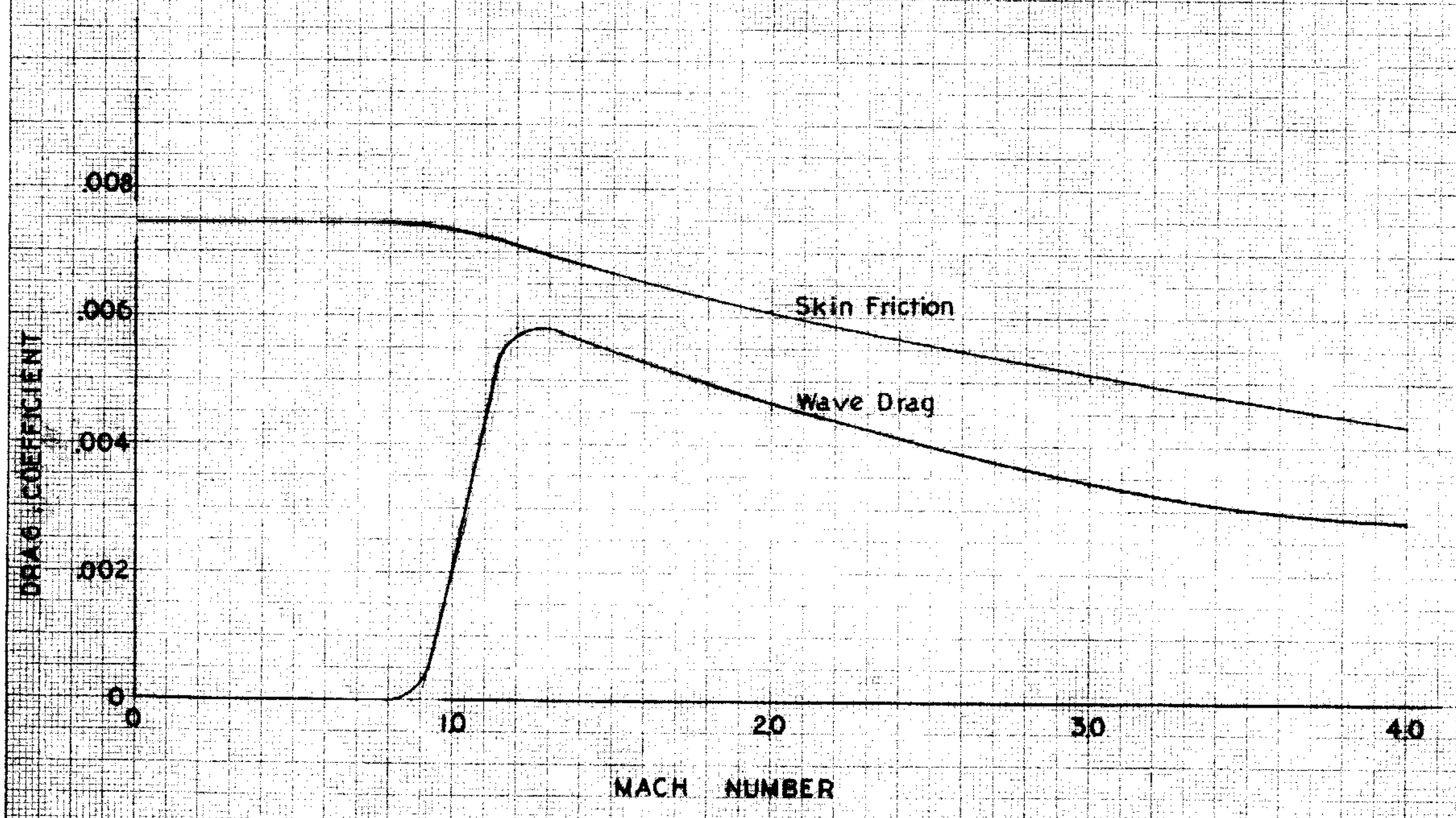

FIGURE 4.3-2

MACH 4.0
ATTACK RECONNAISSANCE SEAPLANE

DRAG-DUE-TO-LIFT

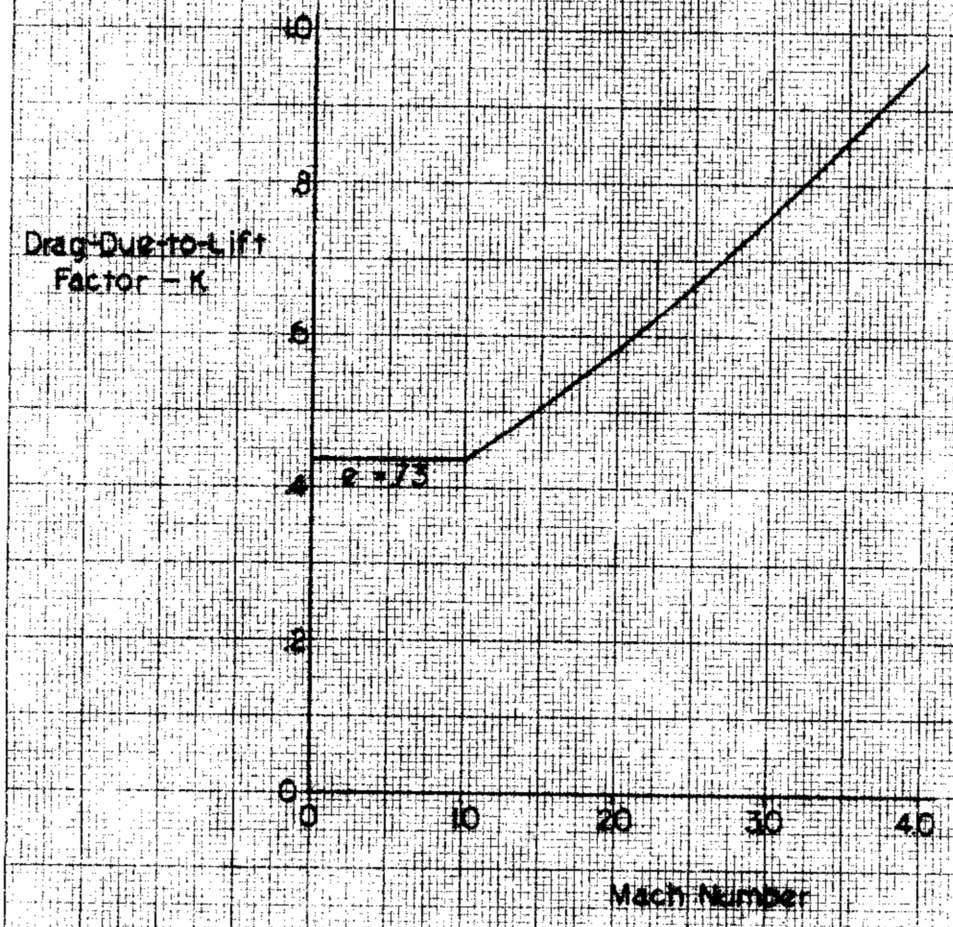

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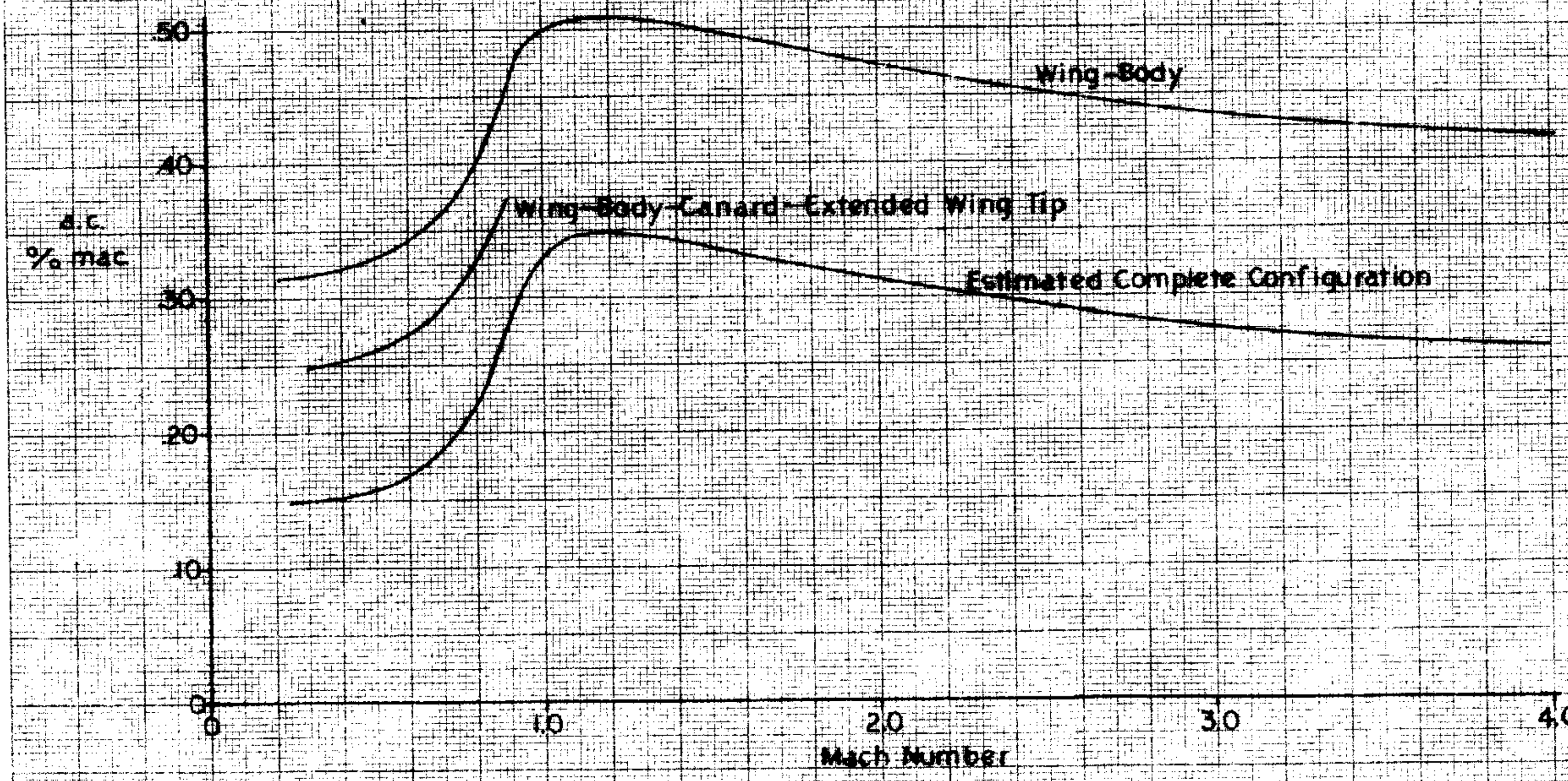
FIGURE 43-3

MACH 4.0 ATTACK RECONNAISSANCE SEAPLANE

A.C. VARIATION WITH MACH NUMBER



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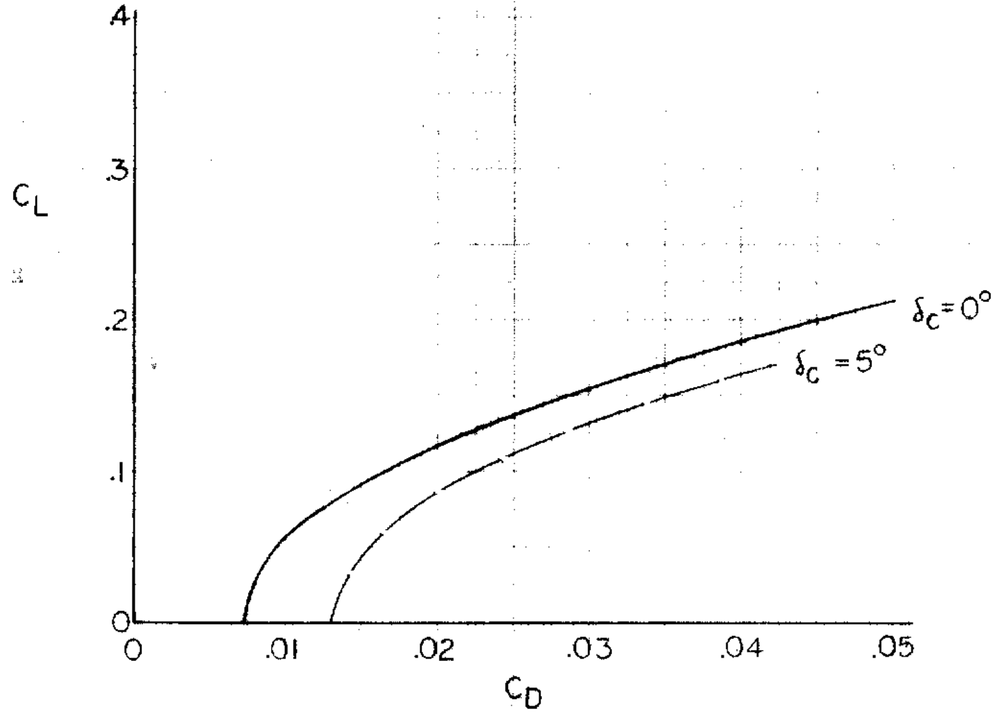
MACH 4.0 ATTACK/RECONNAISSANC

M=4.0 CRUISE CHARACTERIS



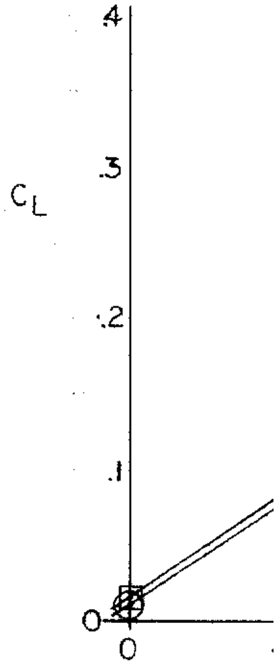
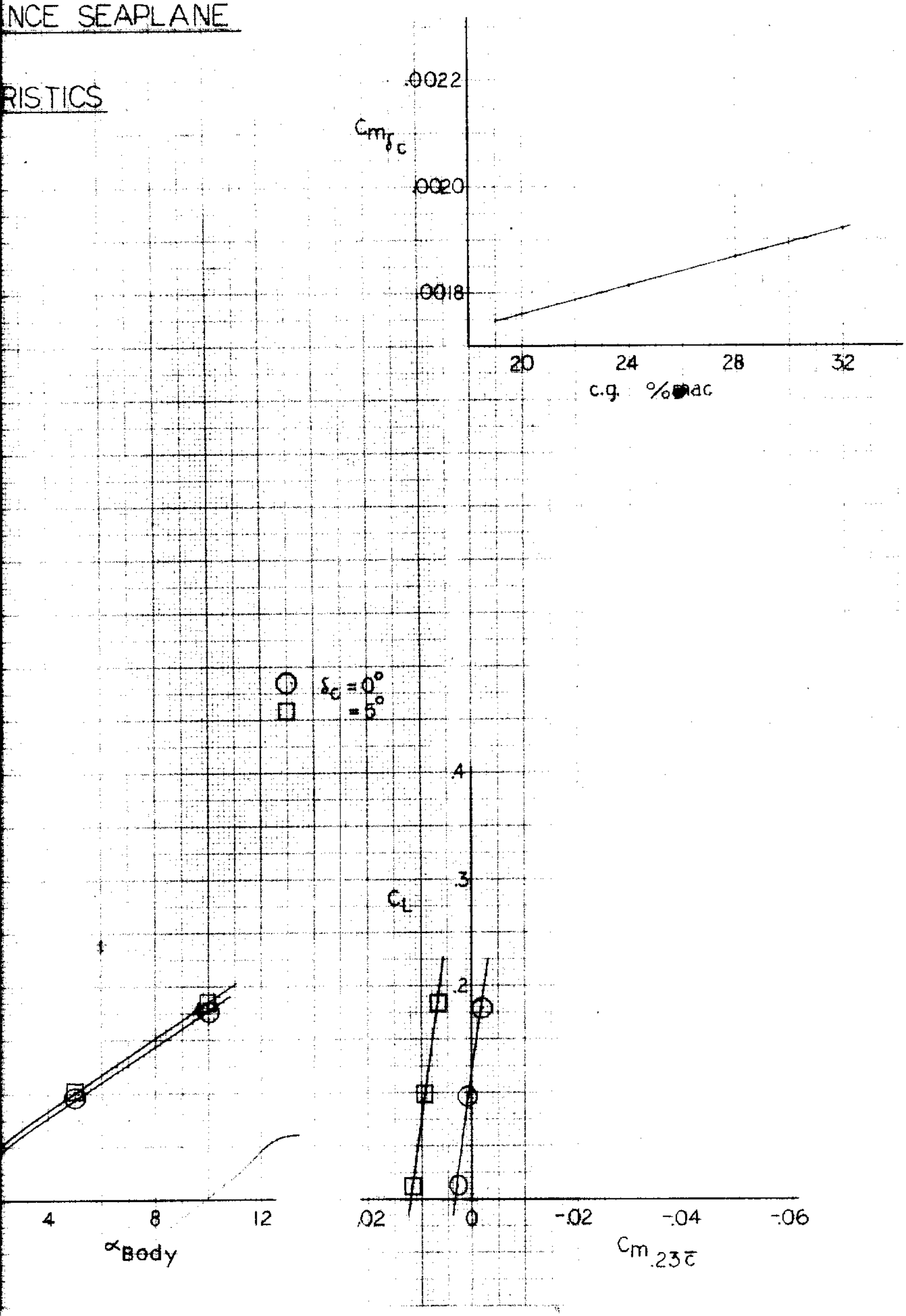


FIGURE 43-4

NCE SEAPLANE

RISTICS



4.4 Mach 3.0 Configuration

The Mach 3.0 Attack/Reconnaissance Seaplane shown by the three view of Figure 4.4-1 and the Inboard Profile of Figure 4.4-2 was prepared prior to the development of the Mach 4.0 Tilt Wing Configuration. The version shown was therefore selected on the basis of the minimum weight indication at that time. That was the leading-edge submerged inlet version in comparison with the roll-over version.

As is noted in Section 9.0 of Volume II, the basic engine selection studies performed at Mach 3.0 were directed towards commercial type application. The engines considered were primarily those under development at the present time or study refinements of these engines. As a result, the Mach 3.0 engine selection (STJ-19A, 2100° F. turbine inlet temperature; proposed by Pratt & Whitney) is based on a more conservative basis with respect to calendar time and also with respect to the operational time within an engine development cycle.

As a result, a direct comparison of the estimated take-off weights for the Mach 4.0 Configuration and the Mach 3.0 Configuration is based on different engine development levels.

The sustained cruise mission profile for this configuration is the same as for the Mach 4.0 seaplane of Section 4.0 except for the difference in operational speed and altitude.

TABLE 4.4-I

MACH 3.0 VERSION WEIGHT SUMMARY

Structure	40,175
Wing	17,400
Tail	2,860
Canard	1,650
Hull	10,750
Sustention	4,380
Surface Controls	1,650
Nacelle	1,485
Propulsion	26,860
Engine	17,400
Air Induction	6,280
Cooling	100
Lube System	200
Fuel System	2,480
Engine Controls	200
Starting	200
Equipment	7,700
Instruments and Navigation	300
Hydraulics, Pneumatics and Electrical	1,900
Communications and Navigation Electronics	1,500
Armament Installation	800
Air Conditioning	2,000
Auxiliary Gear	100
Furnishings and Equipment	1,100
Military Load	10,000
Electronics	4,160
Armament	5,000
Armament Auxiliaries	300
Crew	540
Useful Load	144,065
Oil	200
Fuel	143,365
Trapped Fuel and Oil	500
Total	228,800
Weight Empty	78,895

4.5 Development Program

A broad outline of the development program for the Mach 4.0 Supersonic Cruising Attack/Reconnaissance Seaplane is given in Figure 4.5-1. Initial fleet availability is estimated to be 6 1/2 years from the initiation of the program. A two year research and development program is included for coping with the expected problem areas.

Development programs for the SE-109 engine and the air induction system are indicated in Figure 4.5-2. The engine program is for the development of the JP fuel version considered primarily in this study. An additional three months (total = 54) is required for development of the HEF version.

4.5.1 General Problem Areas

Certain problem areas concerning materials, design, fabrication, and structural proofing are present independent of the specific configuration. These problem areas are primarily a function of the environmental temperature. Of primary concern is the development of materials and material fabrication techniques to realize an efficient structural to gross take-off weight ratio.

4.5.1.1 Materials

Considerable progress has been made in the immediate past on the development of new materials to satisfy the need for strength and stability at high temperature. This work is continuing with further progress to be expected. Dependable mechanical properties of the materials necessary to support this Mach 4.0 Seaplane should be available in the near future.

4.5.1.2 Design

Designing for sustained flight at the temperatures and times considered will require a change in design philosophy. Considerable design information has been gained through research and development programs, but the majority of aircraft engineers have had no need for delving into this information. Thus, the education, or rather, re-education, of the design engineers to design for high temperatures appears to be a major problem.

As the size of the specimen to be tested increases and the test requirements become more stringent, the test cost and time factor increases at a greater rate. No definite value can be assigned to this increase at this time.

4.5.1.5 Material and Fabrication Status

Many of these problems are currently under intensive investigation as related to Mach 3.0 aircraft in development. It is expected that some of the problems mentioned here will have been solved within the time period of this Attack/Reconnaissance airplane through research, knowledge and training gained from related projects.

4.5.2 Potential Problem Areas - Tilt Wing Configuration

Some unique relationships of the tilt-wing configuration indicate the requirements for preliminary testing to verify the engineering estimates that have been made. Other factors of potential difficulty can be resolved through careful and judicious design. In the first group are aerodynamic and hydrodynamic model testing requirements. In the second group are factors such as sonic fatigue effects on the wing lower surface as caused by the engine and the mechanization of the tilt wing.

4.5.2.1 Aerodynamic Testing

The proposal submitted by Convair on this Supersonic Cruising Attack/Reconnaissance Aircraft Study portrayed a two phase study. Phase A, consisting of analytical and design studies, is summarized in this report. Phase B of the proposal outlined a wind tunnel test program.

This wind tunnel program is required to confirm engineering estimates in two specific areas. At low speed, very sparse information is available on low aspect ratio tilt wings. Experimental work in this area will be in addition to that described in Phase B of the proposal.

The effect of wave drag on the airplane gross take-off weight is discussed in Section 6.4 and is shown for a configuration other than the tilt wing in Figure 6-5. In this case, a 10% increase in wave drag resulted in an increase in take-off weight of almost 20%. Although some increase in aerodynamic efficiency will be realized during the development period of this airplane, a verification of the design estimates is necessary to assure the characteristics of the airplane presented herein.

The adverse effects of high temperature on the strength and rigidity of an aircraft are not fully known. Some predictions can be made in this area, but the problems associated with each specific configuration will be different dependent on the airplane and the utilization of the airplane.

In addition to the basic material and design problems, the determination of an economical balance between airframe weight and strength, although necessary for any new aircraft, is expanded due to the magnitude of the step forward.

4.5.1.3 Fabrication

In the fabrication area, one of the most imposing problems will be "learning to live" with the materials to be used. The extensive use of any new material, or a new alloy of the same basic material, inviariably present manufacturing problems. Also, changes in design philosophy, due to the temperatures involved, will require the solving of a great many unknowns before a useful article can be produced.

Some of the manufacturing problems requiring investigation include forming, machining and joining of the materials selected, processing methods, quality control standards and inspection procedures. Fabrication sequences will require revision to accommodate the changes in design and joining methods used. This will also affect all tools used, from those used for the manufacture of the detail parts to those used for the mating of major components.

Welding, both resistance and fusion, is expected to be the joining method used for aircraft of this type. The development of welding techniques and the training of a large number of people as welders also will be required prior to fabrication.

4.5.1.4 Testing

Considerable progress has been made in the amount of testing involving load-temperature-time relationships. Component testing to obtain design information under these conditions is now being performed. The basis on which to project the requirements for testing a major component of this aircraft is known. Necessary test equipment and power supply would be the limiting factors.

4.5.2.2 Hydrodynamic Testing

Two specific areas of concern appear to warrant hydrodynamic model testing of the tilt wing configuration. The first is the generation of spray by the engine exhuast impingement on the water. Considerable thrust-generated water splash-up will occur in the vicinity of the wing trailing edges. Although this may have an adverse resistance effect at pre-hump speeds, the engine effects may reduce any tendency of water roll-up on the afterbody at high speeds. This would be favorable from the standpoint of resistance and directional stability.

The second area is the relatively undesirable location of the wing tip floats. The floats, which also serve as ventral surfaces, are located approximately 30 feet aft of the airplane C.G. With such a location, the floats are vulnerable to damage in yawed landings. Adverse bow-down pitching accelerations at landing may also be expected with this float location. From the standpoint resistance, the floats will be additive as they are either immersed or will be spray targets throughout the take-off run.

The floats will directional stabilize the airplane to the extent of adversely affecting low speed water handling characteristics, however, the extension feature of the float lends itself to a control system where the pilot could steer the seaplane by varying float extension.

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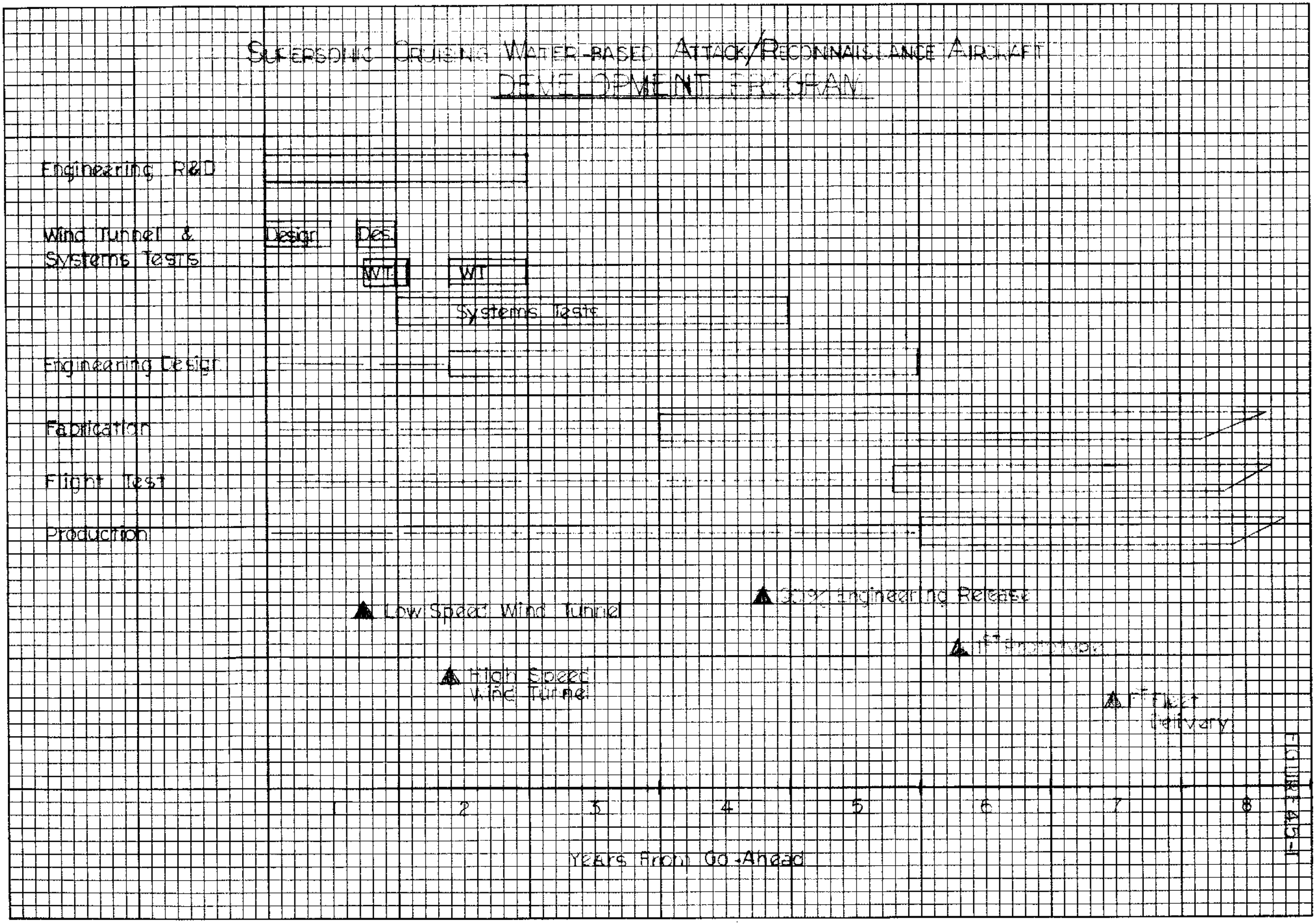
FIGURE 4.5-I

SUPERSONIC CRUISING WATER-BASED ATTACK/RECONNAISSANCE AIRCRAFT
DEVELOPMENT PROGRAM



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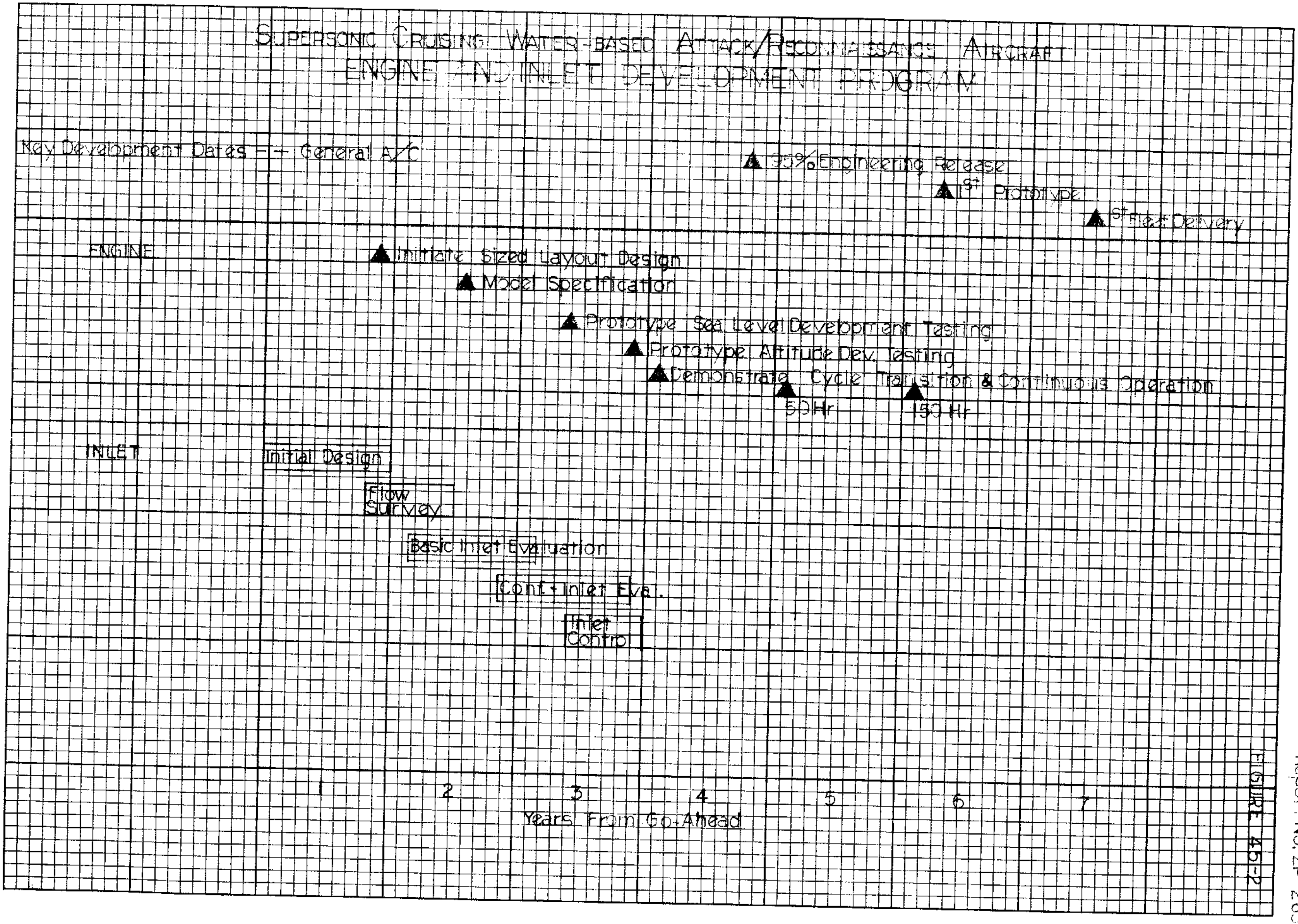

SUPERSONIC CRUISING WATER-BASED ATTACK/RECONNAISSANCE AIRCRAFT
ENGINE AND INLET DEVELOPMENT PROGRAM

FIGURE 45-2

4.6 Vehicle Evaluation

The inherent abilities of the seaplane, and the geographical position of the major threat couple together to indicate a potential advantage for the Attack/Reconnaissance System.

The advantages accruing from water basing lie in the inherent ability of seaplanes to employ any suitable water area for landings and take-offs. Specifically, this ability results in increased dispersal, survival and operational flexibility of the system.

The geographical distribution of water and land masses, and the influence of the North Pacific Current and the Gulf Stream provide a peripheral zone surrounding the threat suitable for year around seaplane operation with the exception of the limited Northern Arctic sector. The outlying islands, the extensive inland seas and sheltered waters afford a multitude of suitable sites for seaplane operations.

Realization of this well dispersed operation requires the development of mobile, independent and self-sustaining tender basing complexes in conjunction with the Attack/Reconnaissance System. Further flexibility, permitting deeper penetration in many cases, can be realized by submerged refueling sites at strategic locations.

Previous studies (Reference I[illegible]-13) have indicated over 400 potential mobile basing sites. These sites are distributed among various operational theaters as West Europe, Mediterranean, Middle East, South East Asia and Japan. Additional sites would be available in seasonally favorable weather.

The many water basing sites available for operation not only forces the enemy to dissipate defenses to counter the omnidirectional threat, but poses the difficult problem of initiating a coordinated world-wide surprise attack on all bases.

In general, these sites are located in a band extending from 600 to 1500 nautical miles from the boundaries of the threat. Closer locations are included when area defense are present such as in Japan and the British Isles.

This peripheral basing arrangement, coupled with the 2000 nautical mile range of the Attack/Reconnaissance Seaplane will yield penetration distances of from 500 to 1400 nautical miles and greater from those bases protected by local defenses. Greater penetration will also be realized through utilization of submerged refueling sites located in nearest return friendly waters.

With this penetration capability, a high percentage of targets lie within striking distance. The potential targets within the sphere of operation of the Attack/Reconnaissance System include airfields, naval installations such as bases, ports, and shipyard facilities, naval forces and missile launching installations.

Operational capability in the reconnaissance mode would be expected to be high with the exception of missile launching sites and possibly the naval forces. The relatively small sizes of these targets require high resolution capability of the sensing device and may limit detection to very clear weather operation.

With pre-reconnaissance information on missile launching installations and a limited area location of naval forces, operation in the attack/reconnaissance mode should result in a good capability.

5.0 Configuration Development

The evolutionary process of arriving at an optimum airplane to perform the design requirements of mission radius capability and payload capacity requires the numerical investigation of the performance capabilities of a large number of configurations and configuration variations.

The Synthesis Program, described within this section, in utilizing digital computing techniques greatly facilitates this type of evaluation. Advantages are derived not only in the decreased analysis time, but in the capability of arriving at optimum design points for each of the configurations under consideration.

Three basic configurations were developed and investigated in the synthesis studies. These consisted of: (A) a partially submerged underwing inlet version where the inlets contribute to static flotation and auxiliary inlets are used for take-off and landing; (B) a roll-over concept where the crew would be located in a rotating capsule; and (C) a hull version with a tilt-wing to provide take-off altitude. The fourth version, (F) Outboard Tail-NASA Configuration , initially prepared for synthesis analysis, was discontinued prior to machine investigation.

Other variations compared in the configuration studies include a $M = 3.0$ and a $M = 4.75$ version of the (A) configuration and a $M = 4.0$, aspect ratio = 1.75 version of the (A) configuration for determining the airplane effects for improved low speed performance.

5.1 Parametric Study Factors

The nominal configuration variables required for optimizing a given design consist of the wing loading and the thrust loading. These are in addition to the wing planform parameters and engine sizing factors developed from the component selection studies.

The development of a family of curves in these two variables requires a minimum of 9 configuration designs with corresponding engine airframe sizing for satisfying the mission requirements. Additional investigations may be required if the chosen limits do not produce an optimum result.

When this development is applied to a number of distinct configurations and to a number of improvement variations on each configuration, the design and analysis task becomes unwieldy, and in attendance with this, subject to error.

This complexity is considerably reduced by the use of automatic computing machines.

5.2 Synthesis Program

The designation of "Synthesis Program" refers to the digital computing machine processes used to size an engine and an airplane combination to meet a certain set of design requirements. Because of the nature of digital machine functions, extensive investigations into the relationship of design variables may be made. Design and parameter variations can be examined to determine those combinations resulting in the most "optimum" over-all configuration.

In this study, the synthesis approach was used to compare configurations, determine the best thrust and wing loadings for these configurations, and to derive incremental gross weights as a result of changes in design radius, design payload and fuel type.

5.2.1 Synthesis Program - General Description

The digital machine synthesis program utilized in this study was originally prepared to permit three distinct modes of operation. These modes are:

1) Synthesis Program: For the given design mission, both the engine and airplane size are variables, values of which are determined by the machine to satisfy the mission.

2) Fixed Engine Program: Same as 1) except that the airplane size only is variable.

3) Analysis Program: The engine and airplane are both fixed with the variable being mission radius capability.

A simplified flow chart of the mission calculations for the synthesis analysis is given in Figure 5.2-1. This synthesis analysis mode was used almost exclusively throughout this study for both the parametric and the propulsion studies.

Airplane weight and volume, aerodynamic characteristics and engine performance are expressed in equation form for machine processing. These equations are such that the airplane and engine size can be variables determined by the machine to accomplish a given mission. The accuracy or completeness of the analysis equations is limited to some extent by the storage limitations of the IBM 650 machine utilized. However, machine calculated mission performance results have shown good agreement with hand calculations, and the program is considered adequate for preliminary design studies.

A numerical description of each configuration is prepared on the basis of a preliminary design layout. The drag, weight and volume constants are described for the machine equations. Constants for the engine equations are determined from engine manufacturer's specification performance reports, adjusted for pressure recovery and additive drag losses (Section 7.2.1.1).

Constants which describe the mission, i.e., climb path, design speed, initial cruise altitude, engine cruise power setting, etc., are also required as input data.

The basic mission specified for this study consists of (Figure 5.2-2):

1) Warm up, take-off, and accelerate to climb speed - fuel allowance based on one minute's maximum power operation (plus assist when required), and five minutes of normal rated power operation;

2) Climb and accelerate on engine limit line to design speed;

3) Climb at design speed to altitude for maximum L/D ratio;

4) Cruise to 2,000 nautical miles, distance of 2) + 3) included, at altitude for maximum L/D;

5) Weapon release;

6) Return (at design speed) at altitude for maximum L/D;

7) Descent (6 + 7 = 2,000 nautical miles)

8) Fuel reserves for 20 minute loiter, plus 5% initial fuel.

5.2.2 Synthesis Program - Equations

The equations used to represent the "rubberized" configuration consist of expressions for the drag, weight and volume distribution and the engine performance.

5.2.2.1 Drag Basis

The total drag coefficient is expressed in the program as the sum of minimum drag and drag due to lift (induced drag, including trim drag);

$$C_{D_{TOT}} = C_{D_{min}} + K\,(C_L - \Delta C_L)^2$$

where,

$C_{D_{min}}$ = minimum drag coefficient

K = induced drag factor - $C_{Di}/C_L{}^2$ - includes mark-up to account for trim drag

C_L = trimmed lift coefficient

ΔC_L = C_L at $C_{D_{min}}$

The minimum drag consists of skin friction and wave drag. Skin friction drag, including camber drag, is assumed constant with Mach number up to the start of the drag rise, and decreases with Mach number in the supersonic region. The subsonic value is determined from:

$$C_{D_{f_{suf}}} = C_f\left[2\left[\frac{S_{W_{wett}}}{S_W} + \frac{S_{HT_{wett}}}{S_W} + \frac{S_{VT_{wett}}}{S_W}\right] + G_{fus}\,\frac{D^2}{S_W} + G_{nac}\,\frac{F_o{}^{2/3}}{S_W}\right] + \Delta C_{D_C}$$

C_f = over-all equivalent friction coefficient

$2\left[\frac{S_{W_{wett}}}{S_W} + \,..\, + .\right]$ = wetted area ratios of wing, horizontal and vertical tail

$G_{fus}\,D^2/S_W$ = wetted area ratio of the fuselage

$G_{nac}\,F_o{}^{2/3}/S_W$ = wetted area ratio of the nacelles

ΔC_{D_C} = camber drag

The constants describing the wetted area relationships of airplane components are determined from preliminary design layouts of the study configurations. This expression for the wetted area ratios allowed the wing area, S_W, equivalent fuselage diameter, D, and sea level static thrust, F_o, to be "rubberized" in the synthesis process. The terms of the expression were derived to allow the configuration to vary in size while still maintaining its geometric similarity.

The camber drag is estimated from available wind tunnel tests on similar cambers. The friction drag decrease with Mach number in the supersonic range is expressed by:

$$C_{D_{f_{sup}}} = C_{D_{d_{sub}}} \left[C_0 + C_1 M + C_2 M^2 + C_3 M^3 \right]$$

where C_0, C_1, C_2 and C_3 are based on the friction drag fall off with Mach number estimated using the method outlined in Section 4.3.1. Reynolds numbers are based on the initial configuration layout, assuming no large effect from changing airplane size.

The maximum wave drag of the configuration is calculated in terms of a frontal area to wing area ratio by:

$$C_{D_{W_{max}}} = C_W \left[\frac{\pi D^2/4 + G_W S_W + G_p F_o}{S_W} \right]$$

where,

C_W = wave drag coefficient based on frontal area

$\frac{\pi D^2}{4}$ = frontal area of the fuselage at the maximum cross-sectional area of the total configuration (Station x)

$G_W S_W$ = frontal area of the wing at Station x

$G_p F_o$ = frontal area of the nacelles at Station x

The constants in this equation are determined from M = 1.0 area distributions of the initial configuration layout. In this study, $C_{D_{W_{max}}}$ was determined from M = 1.0 area distributions by the empirical method of Reference I- 7 , and was assumed to occur at M = 1.2. The variation of C_{D_W} with Mach number is calculated by:

$$C_{D_{W_{sup}}} = C_{D_{W_{max}}} (C_4 + C_5M + C_6M^2 + C_7M^3)$$

$$C_{D_{W_{trans}}} = C_{D_{W_{max}}} (C_9 + C_{10}M)$$

where the variation with Mach number is determined from wind tunnel tests of similar configurations.

The estimated drag-due-to-lift was largely based on available wind tunnel tests of similar configurations. No scale effect was applied to the basic wind tunnel data. The effects of trim were included as a mark-up of the untrimmed drag-due-to-lift, and were determined from preliminary longitudinal stability analyses of the configurations using the method of Reference I- 1 . The equations used in the computer are:

$K = K_{sub}$ = constant for subsonic speeds

$K_{trim} = C_{11} + C_{12}M + C_{13}M^2$ for supersonic Mach numbers

$\Delta C_L = \Delta C_{L_{sub}}$ = constant for subsonic speeds

$\Delta C_L = C_{14} + C_{15}M$ - transonic

$\Delta C_L = \Delta C_{L_{sup}}$ - supersonic

where the constants were determined for the initial configuration.

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5.2.2.2 Weight and Volume Relationships

The total weight and volume relationships, derived from preliminary design layouts, are expressed in equation form to allow the configuration to be "rubberized" in the synthesis process. The take-off weight equation is:

$$W_{To} = W_{fix} + W_F + W_B + W_{pp} + W_{fs} + W_w + W_t + W_h + W_{ski} + W_{sc}$$

where,

W_{fix} = total weight of items that remain fixed independent of aircraft size for a given configuration and mission, i.e., crew, furnishings, electronics, etc.

W_F = weight of available fuel

W_B = weight of weapon load

W_{pp} = powerplant installation weight = $K_{pp} F_o + K_{an} F_o^{n_1} S_w^{n_2}$

where F_o = sea level maximum installed thrust, K_{pp} = engine weight constant, K_{an}, n_1 and n_2 = constants describing variations in air induction and nacelle weights.

W_{fs} = weight of fuel system = $K_{fs}W_f + K_{fsk}$

where $K_{fsk} = \dfrac{\text{basic fuel system weight}}{2}$ and $K_{fs} = K_{fsk}/W_{Fo}$

W_W = wing weight = $K_w \left[\dfrac{Wc}{S_w}\right]^{.570} (S_w)^{1.144}$

where $\dfrac{Wc}{S_w}$ = design wing loading and K_w = constant for a given wing planform.

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W_t = weight of horizontal and vertical tail = $[K_{HT} + K_{VT}]\, S_w$

where $[K_{HT} + K_{VT}] = \dfrac{\text{tail weight of basic design}}{S_{wb}}$

W_h = hull weight = $K_h D^3 + K_{hf}\,(K_c - W_c)$

where K_h = weight of basic hull/$D_b{}^3$, D = equivalent hull diameter at station of maximum cross sectional area of the total configuration, K_c = theoretical combat weight $= W_{to\ basic} - 0.40\, W_{f\ basic}$, K_{hf} = hull weight factor based on weight of the basic hull.

W_{ski} = weight of sustention gear = $K_{ski}\, D^3$

where K_{ski} = ski weight of basic design/$D_b{}^3$

W_{sc} = weight of surface controls = $K_{sc}\,(S_w)^{.52}$

where K_{sc} = surface control weight of basic design/$(S_{wb})^{.52}$

Two separate equations are available for the hull volume relationships, one when the hull size is set by the volume requirements of fuel, payload, etc. (volume limited case) and the other when the hull size is determined by static flotation requirements (flotation limited case).

The hull volume equation for the volume limited case is:

$$V_h = V_{hf} + V_{ski} + V_{pp} + V_{un} + V_{fix}$$

V_h = total hull volume = $C_h D^3$

where C_h = hull volume of basic design $/(D_b)^3$

V_{hf} = volume of fuel in hull with allowance for structure, expansion space, and fuel system = $C_f W_f - C_{wf} S_w{}^{3/2}$

where C_f = fuel volume constant = 1/0.9 x fuel density (lb/Ft^3),

C_f = wing fuel volume constant

$= \dfrac{\text{gross volume for wing fuel in basic design}}{S_{wb}^{\;3/2}}$

V_{ski} = volume of full cavity for sustention system

$= C_{ski}\, D^3$

where C_{ski} = ski cavity volume in basic design/$(D_b)^3$

V_{pp} = volume of power plant installed in the hull = $C_{pp}\, F_o$

where C_{pp} = power plant volume in basic design/F_o

V_{un} = unusable volume in the hull = $C_{un}\, D^3$

where C_{un} = unusable volume in basic design/$(D_b)^3$

V_{fix} = fixed hull volume requirements independent of aircraft size including items as crew compartment, electronics, etc.

The flotation limited case is described by:

$D^3 = C_B\, W_{TO}$

where

W_T = take-off weight

$C_B = (D_b)^3/W_{TO}$ of basic design

The actual operation performed in the synthesis process is the simultaneous solution of two of the following equations:

$$\text{(i)}\quad W_E = (K_{ski} + K_h)D^3 + K_{sc}\,(S_w)^{.52} + K_w \left[\frac{W_c}{S_w}\right]^{.57} S_w^{\,1.144} +$$

$$+ (K_{HT} + K_{VT})\,S_w + K_{pp}\,F_o + K_{an}\,F_o^{\,n_1}\,S_w^{\,n_2} + K_{fs}\,W_f$$

$$+ K_{hf}\,(K_c - W_c) + K_{fsk} + W_{fix}$$

$$\text{(ii)}\quad D^3 = C_B\,W_{TO} = C_B\,(W_E + W_F + W_B) \text{ - flotation limited case}$$

$$\text{(iii)}\quad D^3 = \frac{C_f W_F - C_{wf}\,(S_w)^{3/2} + C_{pp}\,F_o + V_{fix}}{C_h - C_{ski} - C_{un}} \quad \text{Volume limited case}$$

5.2.2.3 Engine Performance

The engine performance, corrected for installation losses (Section 7.2.4), is expressed in equation form by polynomial representation as determined through an IBM 704 curve fitting program.

Maximum power performance is expressed as:

$$\frac{F_N}{F_{o_{max}}} = \alpha_0 + \alpha_1 M + \ldots + \alpha_5 M^5 + \alpha_6 A + \alpha_7 AM + \ldots + \alpha_{10}\,AM^4 + \alpha_{11} A^2$$

$$+ \alpha_{12} A^2 M + \ldots + \alpha_{14} A^2 M^3 + \alpha_{15} A^3 + \alpha_{16} A^3 M + \alpha_{17}\,A^3 M^2$$

$$+ \alpha_{18} A^4 + \alpha_{19}\,A^4 M + \alpha_{20} A^5$$

$$SFC_{max} = \beta_0 + \beta_{1M} + \ldots + \beta_{14} A^2 M^3 + \ldots + \beta_{20} A^5$$

where A = altitude/10,000

The descent performance utilizes the same form of equations as for the maximum power operation.

Partial power cruise performance is expressed as:

$$\left(\frac{SFC_{pp}}{SFC_{max}}\right)_A = \beta_0'' + \beta_1''\left(\frac{D}{F_{N_{max}}}\right) + \beta_2''\left(\frac{D}{F_{N_{max}}}\right)^2 + \beta_3''\left(\frac{D}{F_{N_{max}}}\right)^3$$

Loiter partial power performance is expressed as:

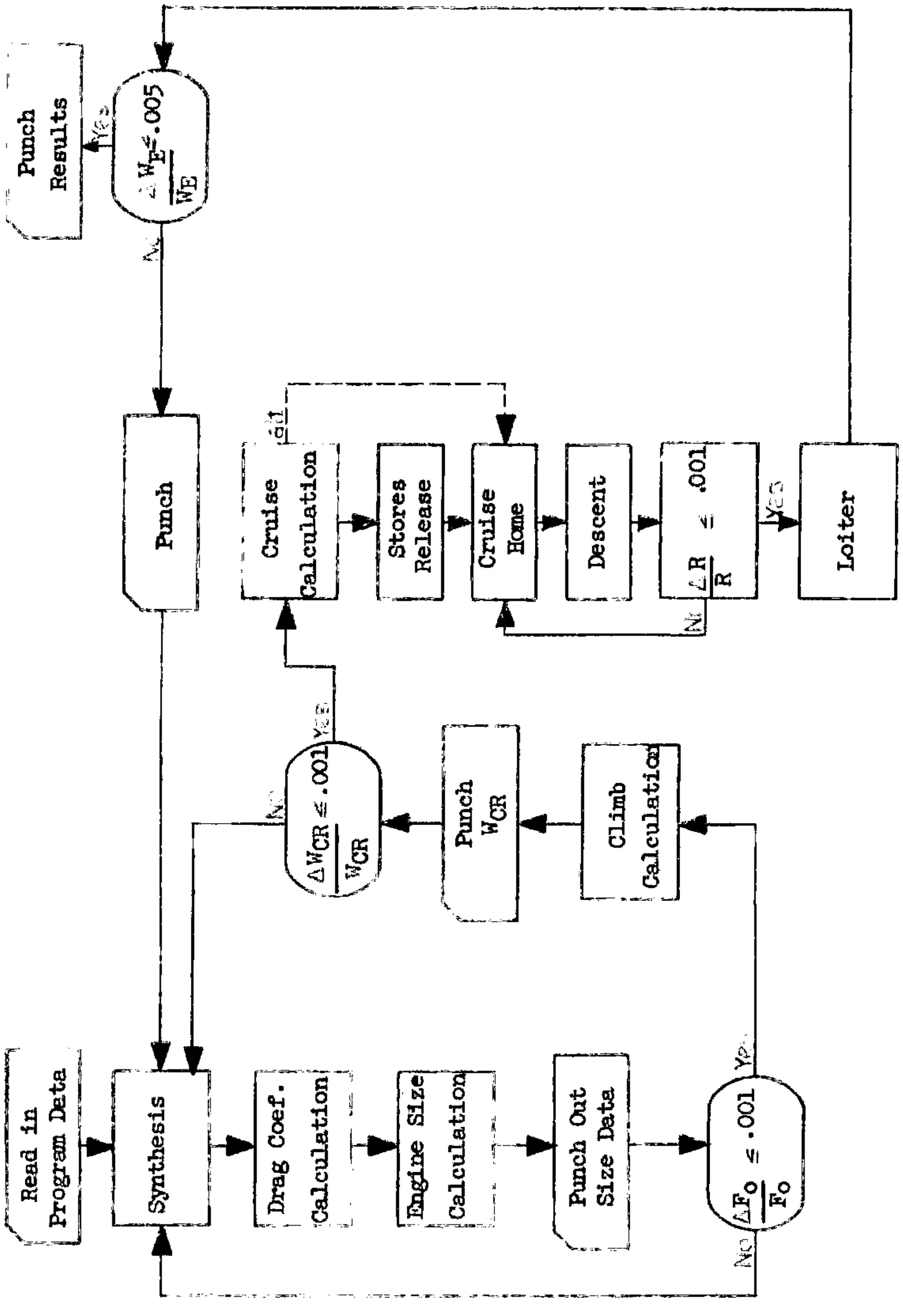
$$\left(\frac{SFC_{loiter}}{SFC_{descent}}\right)_A = \beta_0' + \beta_1'\left(\frac{D}{F_{N_{max}}}\right) + \beta_2'\left(\frac{D}{F_{N_{max}}}\right)^2 + \beta_3'\left(\frac{D}{F_{N_{max}}}\right)^3$$

5.2.3 Synthesis Program - Results

Output data from the IBM 650 are punched onto IBM cards at the end of each loop (Figure 5.2-1) so that the operator can monitor the progress of the calculation. The final mission analysis results are also punched out.

The results from each machine run are sufficient to provide a complete weight - distance and altitude - distance mission profile for the configuration generated to perform the mission.

FIGURE 5.2-1

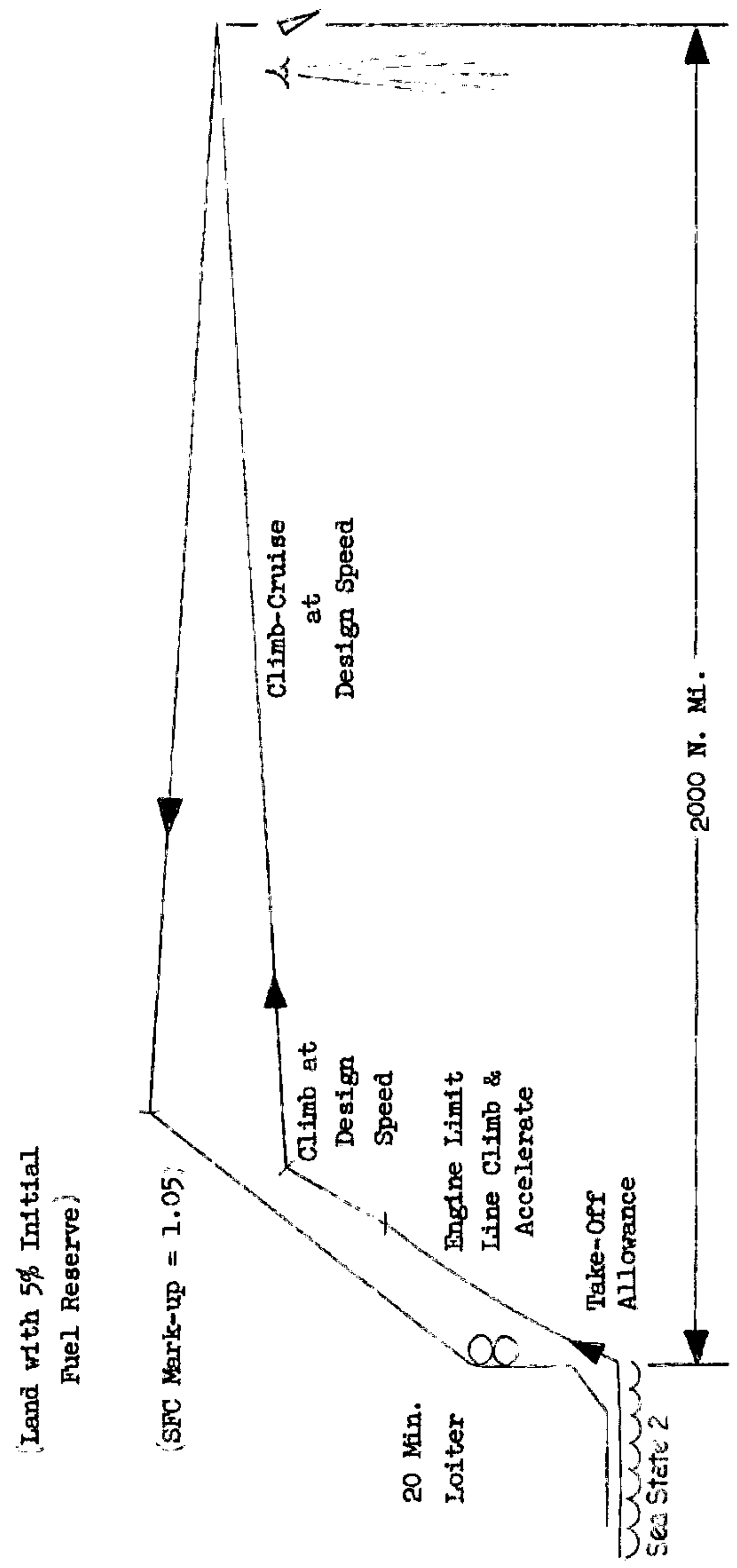


Synthesis Program General Flow Chart

FIGURE 5,2-2

SUPERSONIC CRUISE ATTACK/RECONNAISSANCE

MISSION PROFILE



5.3 Configuration Investigation

The process of development of supersonic seaplane configurations can be based on two distinct design philosophies. The first is to develop a purely aerodynamic configuration until an optimum design is realized. In this case, the effects of water basing are ignored throughout the design's preliminary stages. The design is then modified for water basing.

The second approach develops basic configurations for water basing throughout the study with every attempt made to minimize the water basing effect.

In general, the first approach is not satisfactory because of the inherent inter-relationships between the hydrodynamic requirements and the aerodynamic and other design requirements. In the case of the component selection studies, this method was applied since the items of major interest (engine and wing performance) are relatively independent of the basing concept.

The configuration developments, however, have been based upon utilizing the component selection results with the incorporation of minimum water basing modifications. The primary problems involved do not lie in the components themselves, but rather in the geometric arrangement of the components. In particular, the ever present problem of engine location with seaplanes exists.

A comparison of synthesis investigation results along with the inherent problems and disadvantages of three basic configurations has resulted in the selection of the tilt-wing version presented in Section 4.

5.3.1 Configuration Concepts

The configurations prepared for synthesis investigation have been developed around the SE-109 engine as determined from the Propulsion System Studies (Section 7.2.3) and a wing planform of AR = 1.0, leading edge sweep of 68.1° and a taper ratio of 0.20, as indicated by the Wing Planform Studies (Section 7.1.4). However, some additional wing compromises beyond that for low speed arise as a result of the specific application.

The basic configurations developed include a submerged inlet version (Configuration A, Figure 5.3-1), where the inlet is located under the wing leading edge. The inlet is sealed off for waterborne operation when auxiliary inlets would be used. Particular advantages occur with this arrangement in the reduction of hull cross-section required for flotation by utilizing a portion of the large air induction system volume for the static flotation requirements.

A positive wing incidence of 3° is used to increase the available take-off lift coefficient of the low AR (1.0) wing. The wing planform choice of Section 7.1.4 was further modified to accommodate the leading edge type inlet installation.

A forward control surface (canard) is used to reduce the trim drag during the supersonic cruise phase and to make possible consideration of wing flaps for take-off and landing. The canard is placed high on the upswept body nose to take advantage of the favorable pressure field created by the expanding fuselage nose, to reduce the effect of the canard wake on the inlet, and to relieve canard-wing interference.

The positive C_{m_o} shift due to the upswept body nose has been shown by tunnel test results to provide 5% to 6% improvement in trimmed L/D at cruise for configurations of this type. However, the subsequent incorporation of direct vision capability for low speed manual aircraft control will result in less improvement.

Fuselage cross-sectional shape, and, to some extent, fuselage volume are dictated by hydrodynamic considerations and represent weight and drag penalties due to water basing. No attempt was made to isolate these penalties since it would involve many design considerations, including body cross-sectional and volume changes, wing and power plant relocation, ski vs. landing gear weight, etc., requiring a completely new configuration layout.

Twin vertical tails (tail volume coefficient 0.1) are placed outboard of the canard span to take advantage of favorable side wash outboard of the canard wake.

An alternate arrangement of Configuration A, not developed here, which permits a further decrease in aircraft volume, is a version with a lowered engine (with respect to the water line) with tilt-up of the engine axis for landing and take-off. The problems associated with an inlet duct joint at the temperatures and pressures of the flight regime, as well as the mechanisms involved in the engine tilting and control linkages yielded preference to the version considered.

The second version developed for the configuration synthesis investigation solves the difficult seaplane power plant location problem by placing the inlet-engine package above the wing in the take-off and landing regimes, and "rolling-over" or inverting the airplane for normal flight (Configuration B, Figure 5.3-2). The crew compartment remains upright during conversion by an internally rotating capsule.

In addition to the improved inlet location for cruise, the vertical tails are in a more favorable flow field beneath the wing. However, these configuration advantages over version A are somewhat offset by the increased fuselage volume required for flotation and by the fact that the favorable upswept fuselage nose-canard arrangement is not possible. The aircraft inversion also requires designing the structure for negative load factors comparable to the positive normal requirements thus resulting in increased structural weight. With the use of honeycomb type of construction envisioned for this class of vehicle, this weight increment amounts to a 2.5% factor applied to the basic bending material. More detailed analysis would also introduce other factors in the dual systems requirements where liquids are involved.

Partially offsetting the trend to higher weight is the shorter propulsion system installation.

A third version considered in the synthesis studies is the tilt-wing arrangement (Configuration G, Figure 5.3-3). A lower sweep - higher aspect ratio wing was initially used to reduce the root chord and, hence, the travel of the tilt mechanism.

The tilt-wing concept permits a greater potential percentage of submergence with the engines underslung forward of the wing and also permits consideration of the hull sustention system with lower hump speed and resistance.

The upswept nose advantage can be retained with location of the crew compartment under the center wing leading edge. Such a location provides excellent low speed visibility with no high speed penalty. From a water-basing standpoint, the crew is located in a more preferable location near the C.G. with respect to imposed transient loadings.

High speed vision is limited to the side and downward regions. Foreward vision could be provided by auxiliary means, however, at design speed, object detection and collision avoidance cannot be done optically. Electronic means must be provided in any case.

The shortened forebody and single aft vertical tail result from balance considerations. Additional directional stability must be provided by ventral fins.

A fourth concept, developed around the NASA "Outboard Tail Configuration" was prepared initially for synthesis investigation (Configuration F, Figure 5.3-4). The weight penalty incurred in the wing, boom and tail structures to withstand boom and tail water loads on take-off and landing appear to be large enough to offset the increase in cruise L/D with this configuration (Section 7.4). Dynamic considerations and non-uniform flow across the inlet for off design performance would also introduce development problems in this design.

Other concepts were briefly investigated, but were not developed further because of complexity, varying aerodynamic parameters (pitch-up) or severe spray problems. Included here were configurations utilizing concepts of rotating wing tip-engine-ventral sections, pod-pylon engine rotating from above the wing for take-off and landing to below the wing for flight, and tilt pod-pylon engine packages projected forward of the leading edge.

5.3.2 Configuration Comparison

The configurations developed as input for the synthesis investigations into thrust loading and wing loading were nominally sized for a gross take-off weight of 200,000 lbs., thrust to weight ratio at take-off of 0.4 and a design wing loading at start of cruise of 50 psf. These values were chosen as a result of the indications of the Propulsion System Studies at Mach 4.0, the speed at which the configuration studies were conducted. Some variation was introduced as results became available.

Mach 1.0 area distributions and wetted area ratios were prepared and weight and volume allocations were estimated for defining Configurations A, B and G in equation form for the synthesis investigation. The weight comparison for the synthesis input configurations is given in Table 5.3-I.

Summary results of the machine runs with these configurations are shown in Figure 5.3-8, with variations of take-off weight with wing loading and thrust loading. The mission profile used in these runs consists of an engine Mach limit climb to Mach 4.0, a Mach 4.0 climb to altitude for $(L/D)_{max}$, and $(L/D)_{max}$ altitude cruise to the 2,000 nautical mile radius point, a 5,000 lb. weapon release, an $(L/D)_{max}$ altitude return cruise, an $(L/D)_{max}$ power off descent, and a 20 minute loiter at 25,000 feet. Five percent reserves were retained and engine specification SFC's were marked up by 1.05.

As indicated in Figure 5.3-8, Configuration A results in a lower weight airplane and optimizes at a lower thrust to weight ratio. Higher power settings on Configuration G giving a lower thrust to weight ratio resulted in insufficient thrust for climb and acceleration. The lower weight and thrust loadings of Configuration A are directly due to the lower wave drag permitting a smaller power plant selection because of the improvement in climb-acceleration performance. The increased wave drag of Configurations B and G is due to the increased volume needed for flotation. The higher aspect ratio-thicker trailing edge of Configuration G contributes to the wave drag. Drag-due-to-lift for the Configuration G wing is also increased primarily due to the lower leading edge sweep. Both of these effects result from the shorter root chord desired for the tilting arrangement.

At the power settings corresponding to the "optimum" thrust to weight ratio for versions A and B, and particularly the Configuration A where the "optimum" occurs at the 95% power setting limit chosen to provide maneuver capability at cruise speed, the thrust loadings are too low for unassisted take-off consideration.

5.3.3 Configuration Selection

A general configuration comparison requires, in addition to the relative design weights, consideration of the problem areas associated with each design as well as a critical review of the potential improvements possible with each configuration. Table 5.3-II lists the projected take-off gross weights, specific problems and advantages of each configuration.

The projected take-off gross weights are based on the synthesis results with consideration given to the take-off requirements of fuel and augmentation. Synthesis results on Configuration A and B were also reviewed for the addition of a forward located crew compartment projection in terms of drag and the increased weight required.

During this review, an inconsistency of input data on Configurations A and B was noted. A machine re-run on Configuration A yielded the point A_c of Figure 5.3-8. (The comparative data of Figure 5.3-8 remains valid, but the absolute magnitudes for Configuration A and B data are in error by this increment). This increment is included in Table 5.3-II for Configurations A and B.

A comparison of the specific problem areas and potential advantages, as noted in Table 5.3-IV, indicates the tilt-wing version to be preferable. However, this is not sufficient to compensate for the greater weight of the tilt-wing configuration.

Re-examining the factors contributing to the higher gross weight of Configuration G indicates that a more optimum wing is required and that the volume requirements of flotation must be reduced. Incorporating the "Compromise Wing" of Section 7.1.4 as used on Configuration B results in an increased travel required for the tilt mechanism. This, however, can be lessened by building incidence into the wing relative to the body which in turn provides a more adverse effect of the contracting sternpost fuselage area under the wing. Since a step is required hydrodynamically and must be retracted in flight, this adverse effect can be eliminated by extending the step further.

The critical requirement of static flotation of Configuration G is considered to be the clearance of the wing trailing edge. The specific clearance required must be determined from powered model tests considering the posting area necessary for spray and the engine exhaust. In preliminary work, a value of approximately 20 inches has been used at the inboard trailing edge.

A reduction in fuselage cross section can be obtained by moving the wing pivot point aft while holding the clearance on the trailing edge. Preliminary investigations indicated that modifications in these two areas, the wing and the hull flotation can result in a take-off gross weight competitive with that of Configuration A.

5.3.4 Design Speed Comparison

The primary range of speed for this investigation was from Mach 4.0 to Mach 5.0. In addition, information generated under Convair studies has been included at Mach 3.0. The investigation results, derived on Configuration A prior to completion of the configuration selection studies, are shown in Figure 5.3-9 for Mach numbers of 3.0, 4.0 and 4.75. The reduction from Mach 5.0 to 4.75 is discussed in Section 7.2. The engines considered at Mach 5.0 are limited due to requirements of water injection or to fixed altitude operation in the case of the engine selected, the SE-109. At $M = 4.75$, SE-109 engine operating limits permit an $(L/D)_{max}$ cruise rather than the constant altitude (90,000 ft.) at Mach 5.0.

The results of Figure 5.3-9 cannot be cross-plotted to provide gross weight versus speed because of inherent differences in component characteristics.

At Mach 3.0, (Configuration C, Figure 5.3-5), the engine resulting as optimum (Section 9.2) is not directly comparable to the SE-109 series engine. Under the Convair funded studies in the Mach number range from 2.0 to 3.0, the turbojet counterpart of the SE-109 (SE-105) was not considered. Therefore, a comparison between Mach 3.0 and Mach 4.0 is on the basis of a different propulsion system.

At Mach 4.75, (Configuration D, Figure 5.3-6), initial analysis of material capabilities indicated a working stress level for titanium of 85,000 psi vs. the 100,000 psi level at Mach 4.0. Further investigation of material properties, while confirming the value at Mach 4.0, resulted in a design level of 45,000 psi for titantium at Mach 4.75. On a strength weight basis, where titanium is desirable at Mach 4.0, a material such as Rene' 41 becomes desirable at Mach 4.75. The structural weight allocations of Table 5.3-III for the $M = 4.75$ require an additional 7% increment to account for these material properties.

The weight ratio of 1.68 of the Mach 4.75 to Mach 4.0 design from Figure 5.3-9 will therefore be further increased. Because of this rather large increase in design weight for only a small increase in design speed, the final design studies were conducted at Mach 4.0.

5.3.5 Aspect Ratio Comparison

Results of the wing planform studies (Section 7.0) have indicated a decrease in the range parameter for an increased aspect ratio wing. Specific configuration results on the submerged inlet version were

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obtained for an aspect ratio of 1.75 (Configuration E, Figure 5.3-7) to determine the trade-off for improved low speed performance. The weight breakdown for Configuration E is given in Table 5.3-III.

The aspect ratio variation results for a design speed of Mach 4.0 are shown in Figure 5.3-10. The actual difference in gross take-off weight will be further increased on the higher aspect ratio version by additional take-off assist requirements due to "optimizing" at a lower take-off thrust to weight ratio.

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Synthesis Configuration Input Weights

TABLE 5.3-I

Configuration	A	B	G
Structure	40,065	42,240	40,415
Wing	15,450	15,375	14,400
Tail	2,880	2,300	2,420
Canard	1,740	1,780	1,510
Hull	11,160	14,555	14,600
Sustention	5,450	5,450	3,800
Surface Controls	1,530	1,530	1,665
Nacelle	1,855	1,250	2,020
Propulsion	19,510	18,060	12,945
Engine	11,480	11,480	8,000
Air Induction	4,850	3,400	2,265
Cooling	100	100	100
Lube System	200	200	100
Fuel System	2,480	2,480	2,280
Engine Controls	200	200	100
Starting	200	200	100
Equipment	7,700	7,700	7,700
Instruments & Navigation	300	300	300
Hydraulics, Pneumatics & Electrical	1,900	1,900	1,900
Communication & Navigation Electronics	1,500	1,500	1,500
Armament Installation	800	800	800
Air Conditioning	2,000	2,000	2,000
Auxiliary Gear	100	100	100
Furnishings & Equipment	1,100	1,100	1,100
Military Load	10,000	10,000	10,000
Electronics	4,160	4,160	4,160
Armament	5,000	5,000	5,000
Armament Auxiliaries	300	300	300
Crew	540	540	540
Useful Load	122,725	122,000	118,940
Oil	200	200	200
Fuel	122,025	121,300	118,240
Trapped Fuel & Oil	500	500	500
Total	200,000	200,000	190,000
Weight Empty	71,435	72,160	65,220

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TABLE 5.3-II

TABLE 5.3-II

GENERAL CONFIGURATION COMPARISON

Configuration GTOW	Problems	Advantages
A. Submerged Inlet 205,000	1. Sealing rectangular inlets. 2. Auxiliary inlet. 3. Inlet shape not hydrodynamically desirable. 4. Inlet operation in yawed flight. 5. Wing carry through with honeycomb construction. 6. Salt water in engines. 7. 15-20 ft. ski strut and mechanism. 8. Poor visibility at high angle of attack.	1. Inlet flotation results in more dense aircraft (reflected in weight).
B. Roll-Over 225,000	1. Inlets blanketed at low speed. 2. Rotating capsule, controls and connections. 3. Inversion of engines and liquid systems. 4. Salt water in engines. 5. 15-20 ft. ski strut and mechanism. 6. Poor visibility at high angle of attack.	1. Verticals in favorable field at cruise. 2. Stores eject and load door common. 3. Water seal not required on stores door. 4. Engine inlets in favorable flow field.
G. Tilt-Wing 241,000	1. Wing tilt mechanism. 2. Controls, lines, etc. through wing pivot. 3. Spray drag from engine exhaust.	1. Circular inlets better pressure shape. 2. Crew near C.G. 3. Good low speed visibility. 4. May not require T.O. assist.

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[illegible]

TABLE 5.3-III

Configuration	C-M=3.0	A-M=4.0	D-M=4.75	E-AR=1.75
Structure	38,220	40,065	55,590	45,070
Wing	15,750	15,450	21,275	20,700
Tail	2,300	2,880	3,820	2,880
Canard	1,770	1,740	2,400	1,740
Hull	9,850	11,160	16,400	10,915
Sustention	4,800	5,450	7,100	5,450
Surface Controls	1,530	1,530	1,730	1,530
Nacelle	2,220	1,855	2,365	1,855
Propulsion	23,265	19,510	23,160	19,200
Engine	16,000	11,480	13,500	11,480
Air Induction	4,085	4,850	6,090	4,580
Cooling	100	100	120	100
Lube System	200	200	200	200
Fuel System	2,480	2,480	2,850	2,440
Engine Controls	200	200	200	200
Starting	200	200	200	200
Equipment	7,700	7,700	7,700	7,700
Instruments & Navigation	300	300	300	300
Hydraulics, Pneumatics & Electrical	1,900	1,900	1,900	1,900
Communication & Navigation Electronics	1,500	1,500	1,500	1,500
Armament Installation	800	800	800	800
Air Conditioning	2,000	2,000	2,000	2,000
Auxiliary Gear	100	100	100	100
Furnishings & Equipment	1,100	1,100	1,100	1,100
Military Load	10,000	10,000	10,000	10,000
Electronics	4,160	4,160	4,160	4,160
Armament	5,000	5,000	5,000	5,000
Armament Auxiliaries	300	300	300	300
Crew	540	540	540	540
Useful Load	120,815	122,725	123,550	118,030
Oil	200	200	200	200
Fuel	120,115	122,025	122,850	117,330
Trapped Fuel & Oil	500	500	500	500
Total	200,000	200,000	220,000	200,000
Weight Empty	73,345	71,435	90,610	76,130

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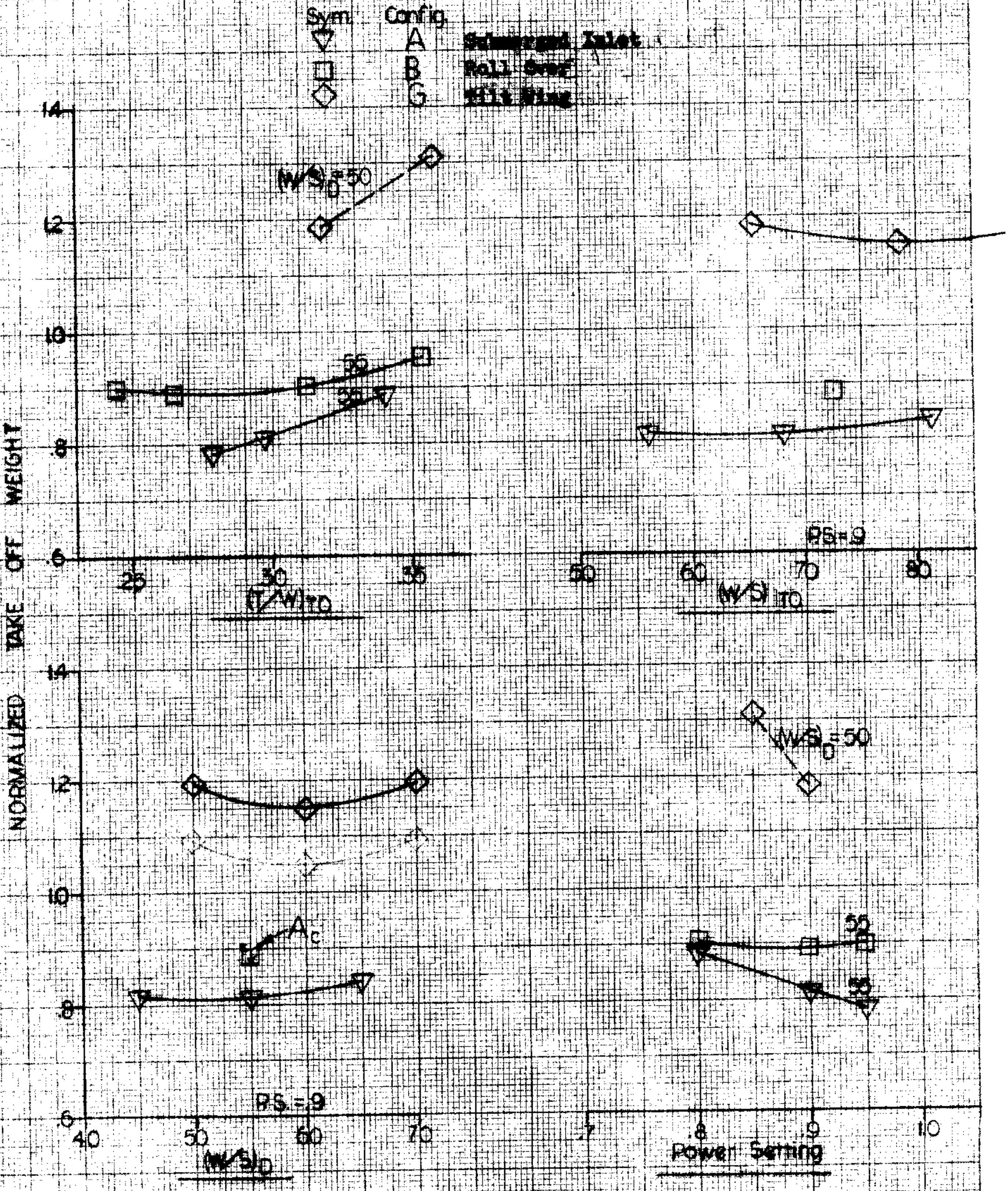
FIGURE 5.3-8

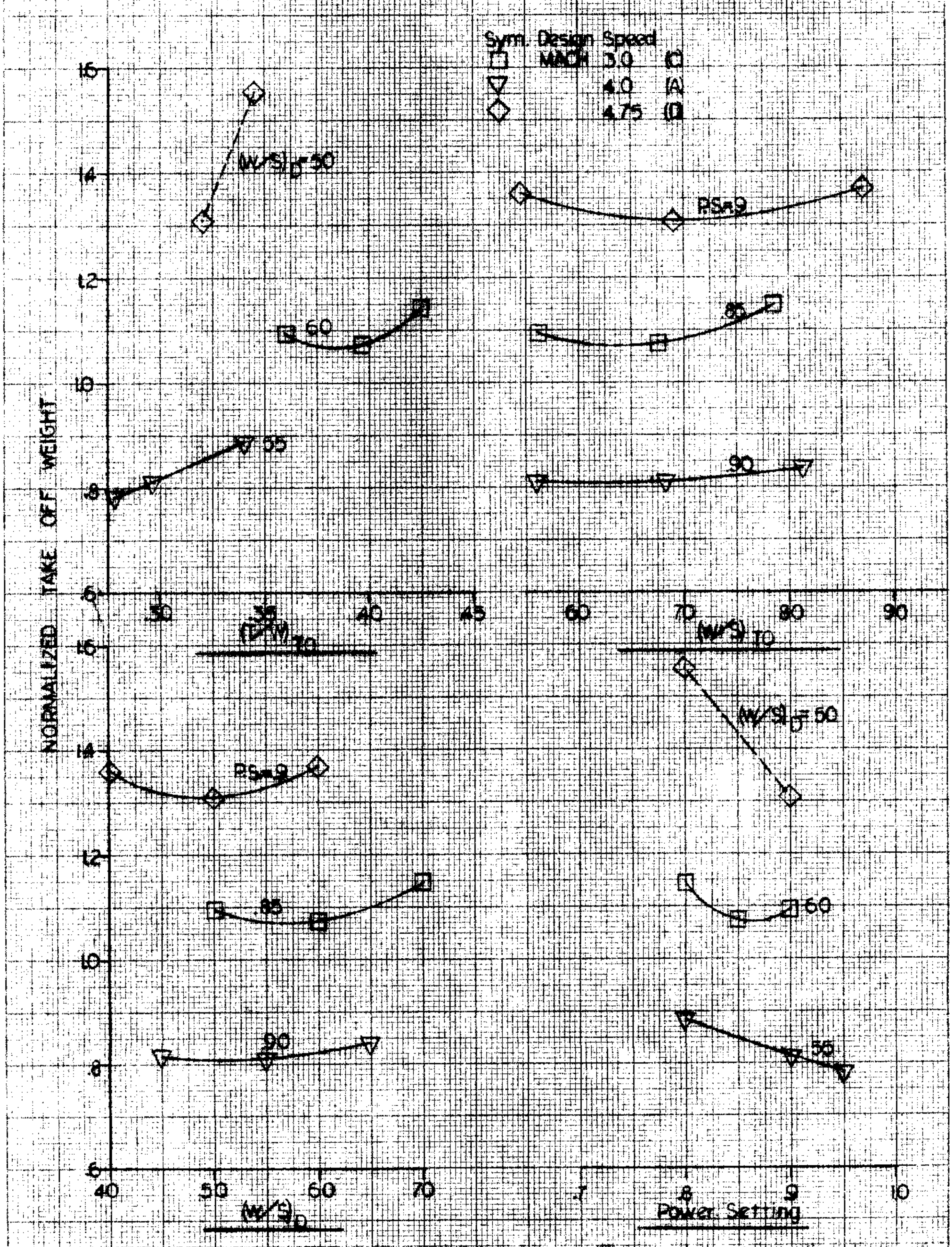
CONFIGURATION STUDIES



Configuration Comparison Results
Mach 3.0

CONFIGURATION STUDIES

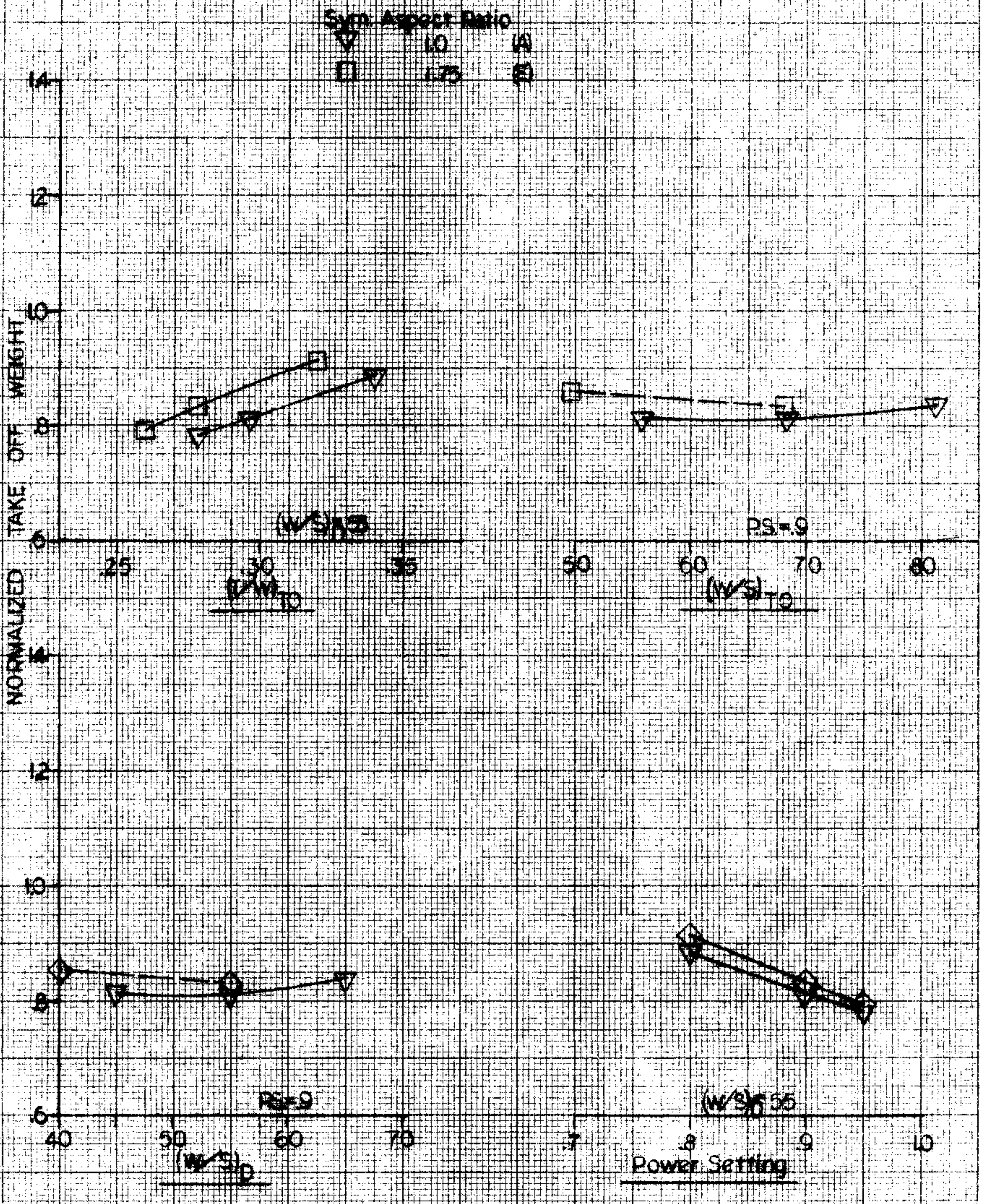
FIGURE 5.3-9



Design Speed Comparison Results - Configuration A

FIGURE 5.3-10

CONFIGURATION STUDIES



ASPECT RATIO COMPARISON RESULTS - ([illegible]) INLET CONCEPT

5.4 Tilt Wing - Synthesis Analysis

The improvements in the tilt wing configuration discussed in Section 5.3.3 are incorporated into the version shown in Figure 5.4-1. These improvements consisted of: a lower aspect ratio (1.0) wing with greater leading edge sweep (68°), actually the modified wing from the wing planform studies (Section 7.1); and, moving the wing pivot point aft.

In conjunction with these changes, the fuselage volume was reduced as far as possible with the retention of approximately 20 inches clearance at the inboard trailing edge with respect to the static water line.

A design speed of Mach 4.0 was used for this final version as indicated in Section 5.3.4.

The weight breakdown derived for synthesis input of this configuration is given in Table 5.4.1. Provisions were included for water injection at take-off, since initial runs indicated a low thrust to weight ratio at take-off as a result of the decreased drag of this configuration.

Results of the synthesis studies on this Mach 4.0 tilt wing configuration are shown in Figure 5.4-2. The mission basis for this investigation consisted of:

1) 5 Minute military power fuel allowance (added);
2) Engine Mach limit climb from Mach 0.9 to Mach 4.0 at maximum power;
3) Maximum power climb at Mach 4.0 to the altitude for L/D_{max};
4) Cruise at L/D_{max} to 2,000 nautical miles;
5) Weapon release;
6) Return L/D_{max} cruise;
7) Decelerate and descent (power off);
8) 20 Minute loiter at 25,000 ft.;
9) Descent and land with 5% reserves.

The take-off allowance of one minute maximum power liquid consumption consisting of fuel and water is treated as an overload condition during static and taxi operation.

Data of Figure 5.4-2 as plotted against the ratio of nominal SLS thrust to lift-off weight includes effects of weapon weight and effects of reduced tolerance on the $\Delta W_{CR}/W_{CR}$ and $\Delta W_E/W_E$ loops of the synthesis program (Figure 5.2-1).

Initial analysis points on this tilt wing configuration were obtained with a weapon weight of 4,000 lbs., compared to the design value of 5,000 lbs. Repeating the analysis with the design weight resulted in a series of gross weight points which indicated that curves drawn through the respective weapon weights would cross.

A close inspection of the computed weights during the machine analysis of the mission profile indicated that the scatter of the take-off weight points was in the direction of the tolerance on the aircraft weight empty. A reduction in the tolerance on $\Delta W_E/W_E$ from .01 to .005 was introduced into the machine program and points were obtained at a stores weight of both 4,000 lbs. and 5,000 lbs. as also shown in Figure 5.4-2. The curves of the figure are drawn through these "improved" points.

It is considered that the variation obtained with the initial tolerance was due to the relatively low empty weight in comparison with the lift off weight (a ratio of .332).

5.4.1 Tilt Wing Design Parameter Selection

Prior to the selection of the "optimum" or minimum weight configuration, the take-off performance of the airplane was investigated. The resistance ratio at hump speed was estimated to be 0.245, while the "optimum power setting selection" point yields a Nominal SLS Thrust to Lift-off Weight ratio of 0.28. This "optimum power setting" is at 95% of maximum power, used as an upper limit to provide a maneuvering margin. This ratio of resistance to thrust at hump is unacceptable. It results in take-off times in excess of 100 seconds and in considerably less than .1g acceleration at hump.

Corresponding to the power setting selection of 0.95 of maximum power for cruise, the optimum design wing loading is 60 psf at start of cruise. Two direct methods of improving take-off performance are decreasing the wing loading and increasing the engine size.

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Investigation indicated that a variation in wing loading from 60 psf to 55 psf would result in an increase in take-off gross weight of less than 1% for a 4% decrease in take-off speed along with a decrease in take-off time. A wing loading at start of cruise of 55 psf was therefore chosen.

In order to attain a take-off time of 60 seconds or less, the take-off-thrust to weight ratio required was estimated to be of the order of 0.36. A direct increase in engine size to attain this thrust loading results in an airplane growth of 1.13 over the optimum engine size selection (Figure 5.4-2). In terms of lift-off weight, where in Figure 5.4-2, the normalizing factor of 200,000 lbs. has been used, the optimum aircraft with a thrust to weight ratio of .28 grows from 184,000 lbs. to approximately 206,000 lbs. to satisfy take-off requirements.

Because of this rather large penalty, engine performance capabilities and assist methods were reviewed. Direct augmentation by means of liquid rocket assist was considered unsatisfactory primarily from the standpoint of logistics. The use of auxiliary turbojet engines for take-off add to aircraft complexity with the requirement of retraction for high speed flight which is compounded with respect to a satisfactory location on a seaplane. In addition, the auxiliary engine installation weight will cause a substantial growth in aircraft weight.

The fixed weight variation curve of Figure 5.4-2 was generated to provide an assist in comparing take-off augmentation methods. With a thrust requirement of the order of 15,000 lbs. and an estimated installed thrust to weight ratio for an auxiliary turbojet of 12 to 1, an airplane growth of 1.065 results. The penalty is not as large as in the case of primary engine increase since the optimum primary engine performance basis is unchanged and the aircraft drag basis is not changed.

The static installed engine and inlet performance was reviewed to determine potential areas of improvement. Inlet tests have shown that the static pressure recovery of 0.85 for sharp lip inlets can be increased to 0.90 by means of blow-in doors or a translating cowl. For the SE-109 engine, this represents an increase in SLS installed thrust of 8.5%.

Additional data received from the engine manufacturer on the SE-109 with water injection shows an augmentation factor of 13.3%.

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Incorporation of the improved pressure recovery and of water injection for take-off results in a basic nominal SLS thrust (.85 pressure recovery) to lift of weight ratio requirement of .305 to yield a take-off time of the order of 60 seconds. The thrust to weight curve of Figure 5.4-2 includes allowances for plumbing, spray-bars, etc. and shows a relative take-off weight of .945 or a lift off weight of 189,000 lbs. for a growth of 1.025. Liquid requirements of fuel and water for the one minute take-off allowance are treated as an overload condition.

5.4.2 Design Selection Summary

The selection of the design parameters for the proposed configuration results in a small compromise in high speed parameters to realize acceptable take-off performance. An engine larger than that required for cruise is used with take-off augmentation provided by water injection. The wing area is also increased to reduce the take-off velocity.

Selected design parameters are:

1) Wing loading at start of cruise = 55 psf

(Wing loading at lift-off = 72 psf)

2) Power setting at start of cruise = 85%

This results in a lift-off weight of 189,000 lbs. compared to slightly less than 184,000 lbs. for the optimum cruise parameter selection of:

1) Wing loading at start of cruise = 60 psf

(Wing loading at lift-off = 76 psf)

2) Power setting at start of cruise = 95%

The major portion of the increased weight is due to the lower power setting.

Liquid allowance for take-off, treated as an overload during water-borne operation, consists of one minute operation at maximum power. This allowance consists of 2,300 lbs. of fuel and 5,200 lbs. of water giving a ramp weight of 196,500 lbs.

The lift-off weight variation of this configuration with design mission radius is also shown in Figure 5.4-2. (The relative take-off weight is normalized at a value of 200,000 lbs.)

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TABLE 5.4-I

MACH 4.0 TILT WING SYNTHESIS INPUT WEIGHTS	
Structure	39,985
Wing	13,600
Tail	2,420
Canard	1,570
Hull	14,600
Sustention	4,200
Surface Controls	1,575
Nacelle	2,020
Propulsion	14,164
Engine	8,969
Air Induction	2,265
Cooling	100
Lube System	100
Fuel System	2,280
Engine Controls	100
Starting	100
Water Injection	250
Equipment	7,700
Instruments and Navigation	300
Hydraulics, Pneumatics and Electrical	1,900
Communications and Navigation Electronics	1,500
Armament Installation	800
Air Conditioning	2,000
Auxiliary Gear	100
Furnishings and Equipment	1,100
Military Load	10,000
Electronics	4,160
Armament	5,000
Armament Auxiliaries	300
Crew	540
Useful Load	138,151
Oil	200
Fuel	137,451
Trapped Fuel and Oil	500
Total	210,000
Weight Empty	66,009

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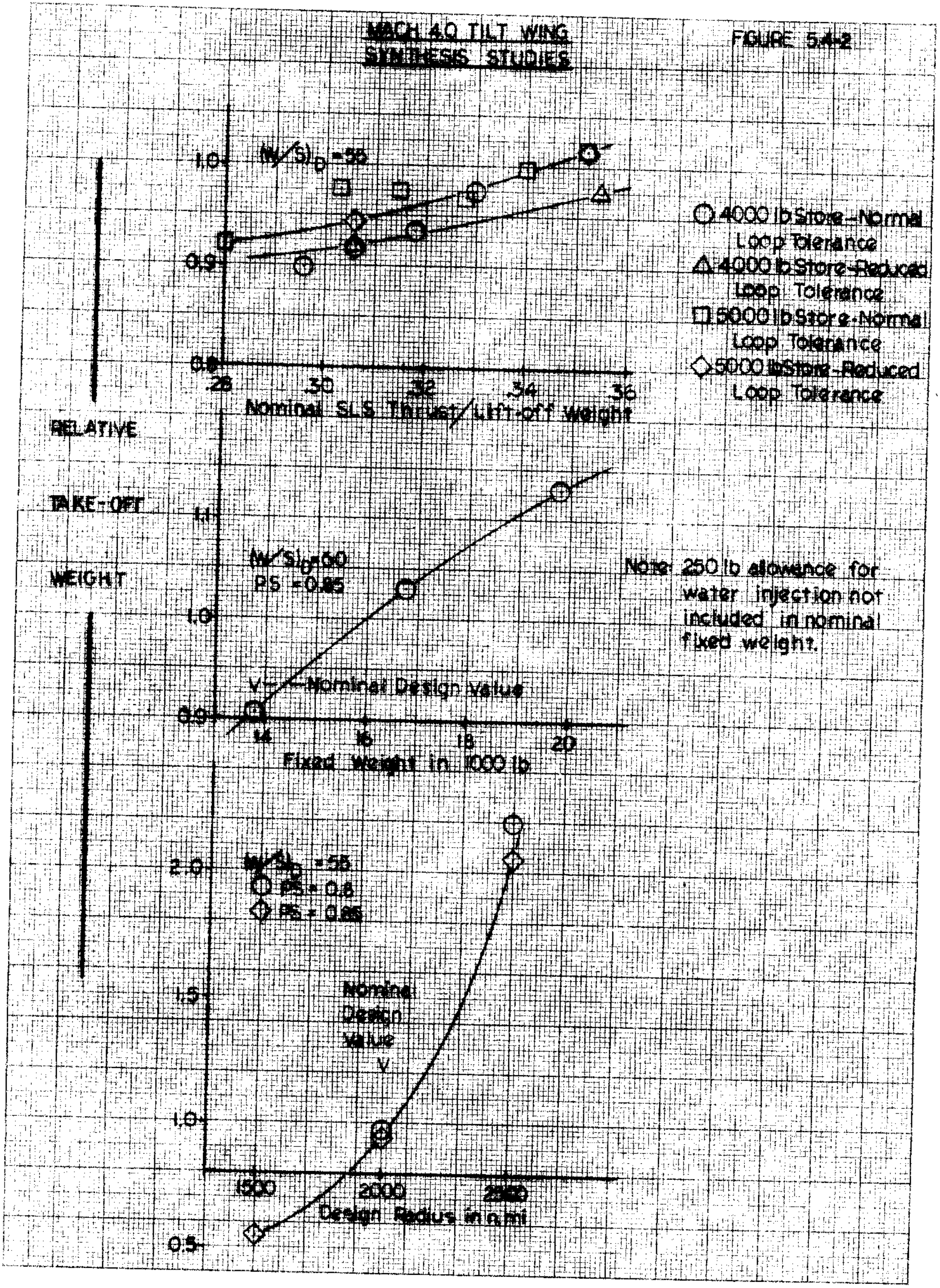
MACH 4.0 TILT WING SYNTHESIS STUDIES

FIGURE 5.4-2



6.0 Mission Parametric Variation

Parametric mission studies were conducted in addition to the configuration studies described in Section 5.0. Specific requirements included an examination of the designs to determine the effects of increasing the military payload to 20,000 and 30,000 lbs. from the basic 10,000 lbs. Other investigations were made into the effects of mission radius, incremental weight increases in either systems or structural weight values and in wave drag increments.

All data derived in this section was based on the submerged inlet configuration. This work was done during the configuration selection studies to provide directional data in the final selection.

6.1 Military Payload

The basic military payload of 10,000 lbs. was divided into allowances for crew, attack/reconnaissance electronics and equipment, electronic-countermeasure equipment, armament and armament auxiliaries of 540, 2060, 2100, 5000 and 300 lbs. respectively. For the 20,000 and 30,000 lb. military payloads, the weight increases were assumed to be in the armament group.

Volumetric requirements of the armament loading, required as input to the synthesis program, were derived from Figure 6-1. Installation volumes required for missiles ranging in weight from 500 to 30,000 lbs. were determined and a line was faired through the known points.

Actual volumetric requirements were not critical for the configurations studied, however, specific missile shapes could require changes in configuration lines.

The volume requirements from Figure 6-1 indicate 350 cu. ft. for a 5,000 lb. armament load and approximately 700 cu. ft. for both the 15,000 and 25,000 lb. armament. This equal volume requirement at the higher weights results from changing the missile loading from a cruise type ASM to an air-launched ballistic missile.

Effects of increased payload on gross take-off weight are shown in Figure 6-2. In each case the armament is dropped at the 2,000 nautical mile radius point.

6.2 Mission Radius

The mission radius requirements are, as yet, undefined for the time period under consideration. The nominal 2,000 nautical miles was chosen as it provides a high percentage of target coverage for peripheral seaplane basing (Section 4.6).

Gross take-off weight variation with mission radii from 1,500 to 2,500 nautical miles is shown in Figure 6-3.

6.3 Fixed Weight

The effects of increasing fixed weight on gross weight are shown in Figure 6-4. This data was derived specifically to provide an indication of the effects for any required take-off augmentation on the over-all take-off gross weight. In addition, the data provides an indication of the airplane growth with systems changes.

6.4 Wave Drag

The effect of increased wave drag for a given configuration on the gross take-off weight is shown in Figure 6-5. An increase in wave drag requires an increase in the engine size and a rather large increase in take-off weight. This effect is also noted in the power setting variation results of Figures 5.3-8 through 5.3-10.

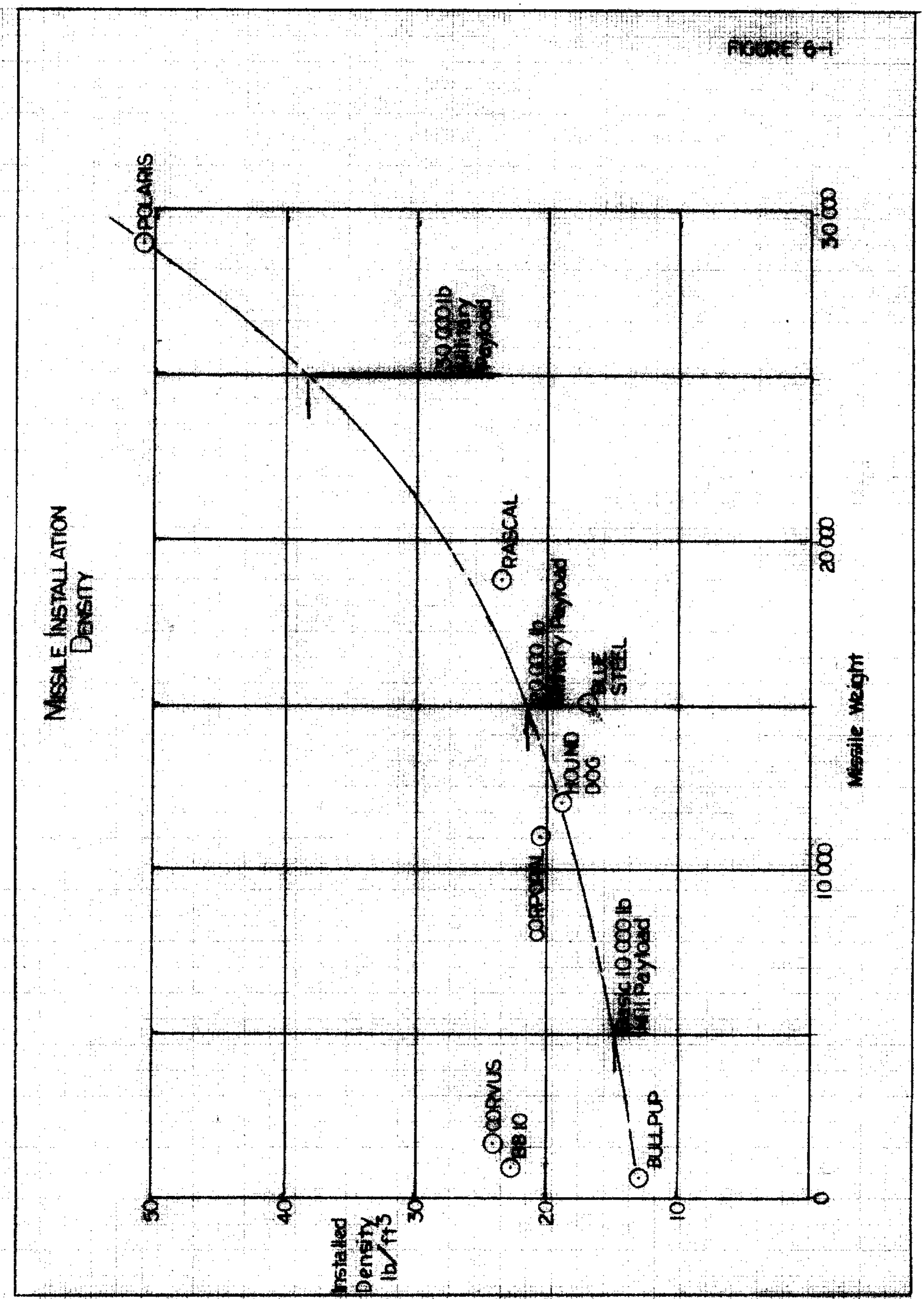


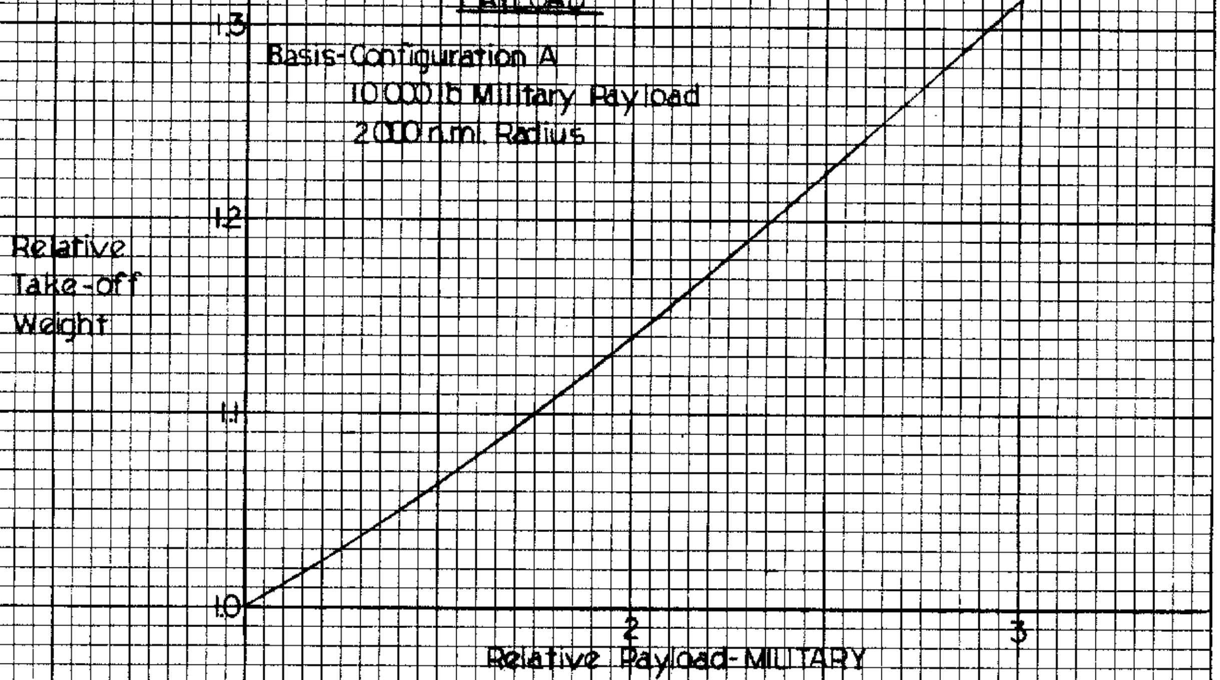
FIGURE 6-1

MISSILE INSTALLATION DENSITY

FIGURE 6-2&3

FIGURE 6-2

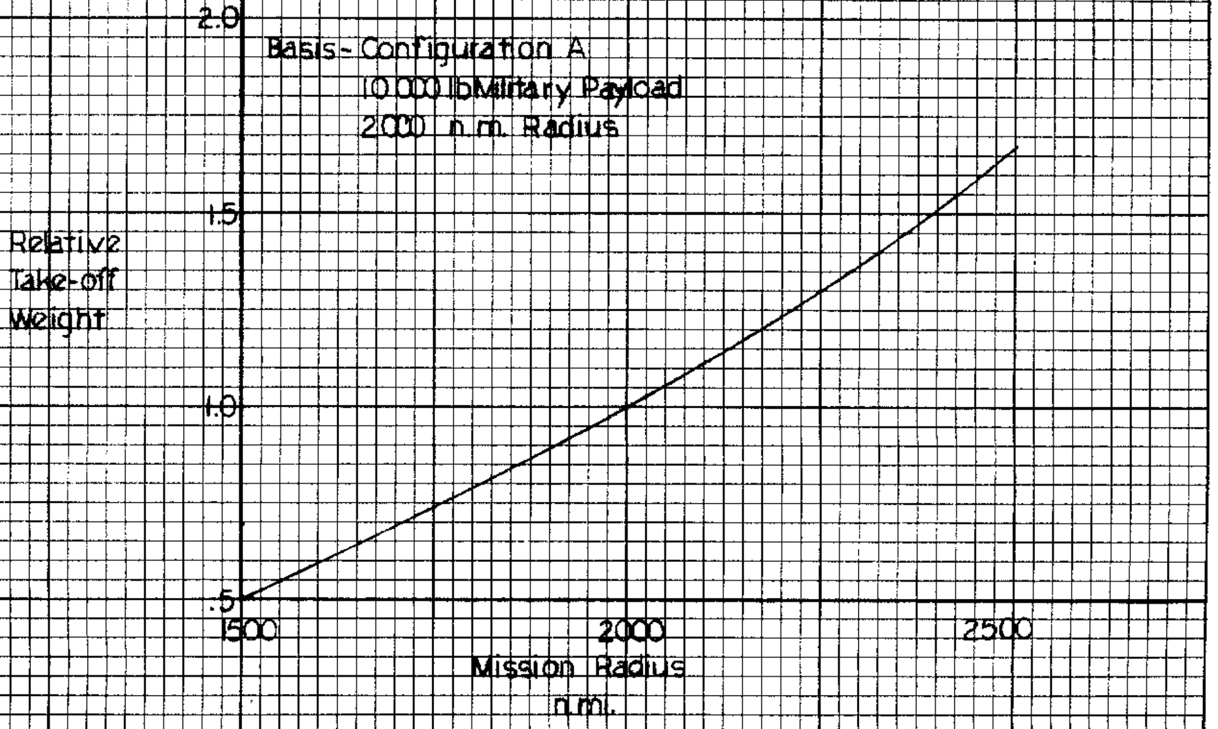
EFFECT OF INCREASED PAYLOAD



Basis-Configuration A
10,000 lb Military Payload
2000 n.mi. Radius

FIGURE 6-3

EFFECT OF MISSION RADIUS



Basis- Configuration A
10,000 lb Military Payload
2000 n.mi. Radius

K&E 10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO. MADE IN U.S.A.

FIGURE 6-4&5

FIGURE 6-4

EFFECT OF INCREASED FIXED WEIGHT

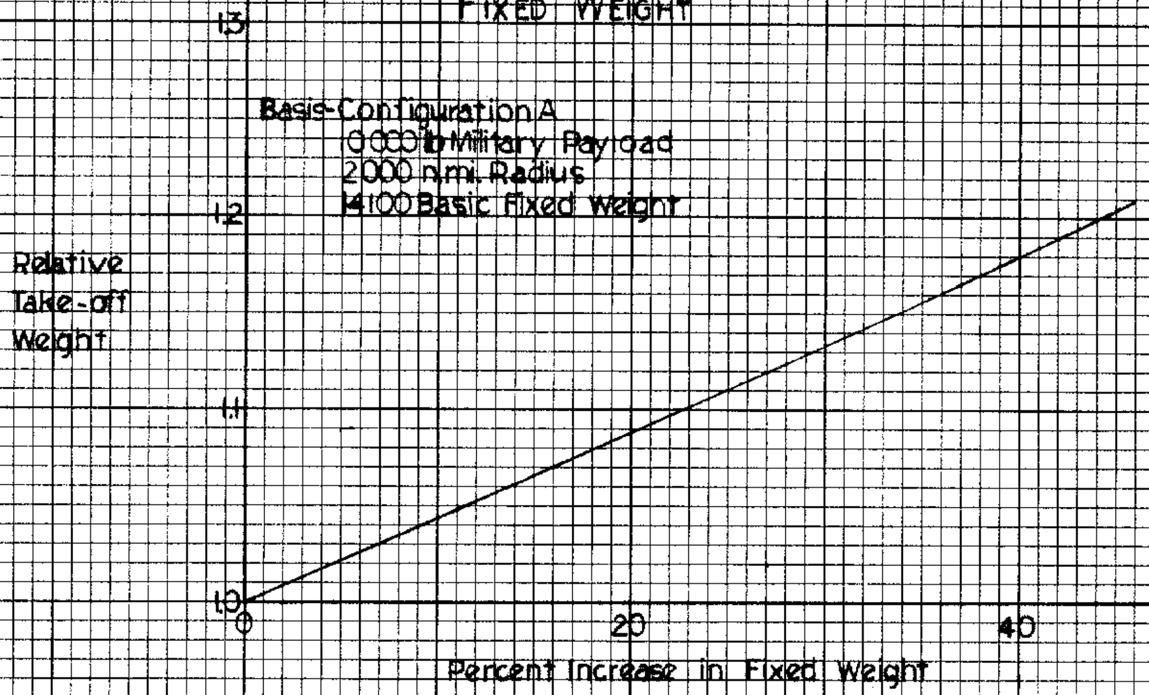
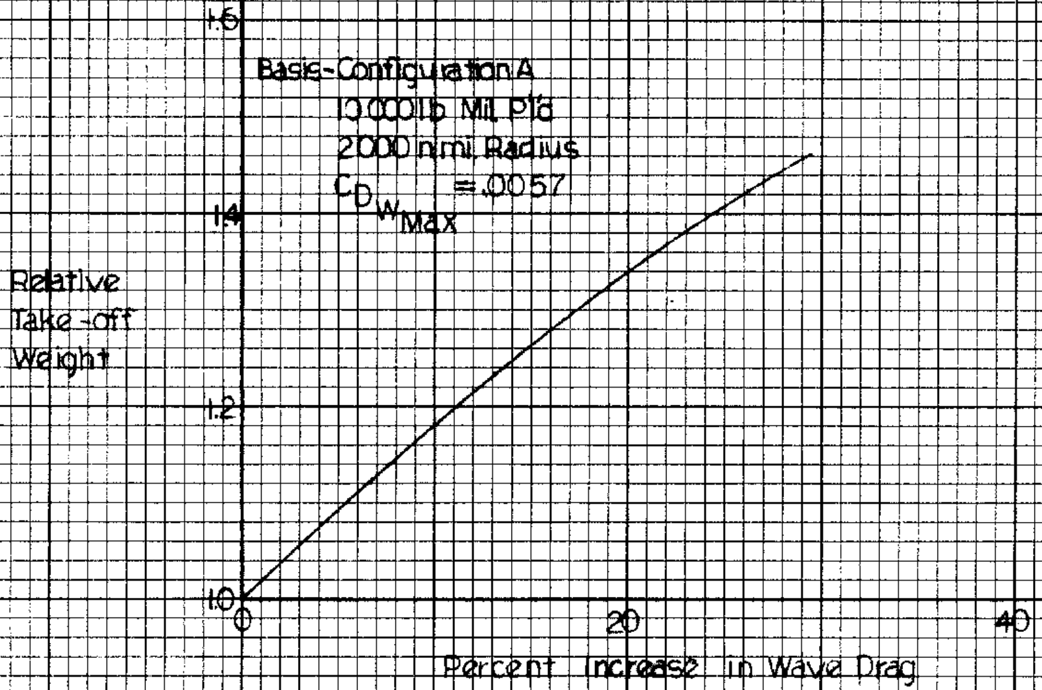


FIGURE 6-5

EFFECT OF INCREASED WAVE DRAG



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REPORT ZP-8-045
DATE 20 August 1956
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TITLE

FUEL REQUIREMENTS FOR THE F-106A INTERCEPTOR IN AIR DEFENSE OPERATIONS

(Including Combat Tanks)

ZP-8-045
Including Addendum Report OR-8-43

PREPARED BY R. C. Lingley
R. C. Lingley

GROUP

REFERENCE

CHECKED BY

APPROVED BY E R Tuttle for
A. R. Lambert

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INTRODUCTION

The existing F-106A airplane is designed to meet the requirements of a 375 n. mi. area intercept mission, as well as certain speed and altitude guarantees. The design basis of this aircraft has been to accomplish the above requirements with internal fuel only. External tanks may be carried, however, as defined by model specification ZD-8-005. These tanks are for the "ferry" condition only and their use is presently restricted by the following conditions:

1. Maximum design speed is Mach No. = 0.95
2. Load factors are reduced in symmetrical and unsymmetrical maneuvers when tanks are carried.
3. Tanks are Class II, non-combat tanks.

These tanks may be used for range extension of the area-intercept mission provided the tanks are dropped during the subsonic cruise out, prior to accelerating to supersonic speeds, or start of combat.

If this airplane is used for a "supersonic dash" or point interceptor type of mission, it is to be expected that the tactical radius will be seriously reduced or if the present Royal jet external tanks are used that they will of necessity have to be dropped at the top of the climb regardless of whether there is any fuel remaining.

In view of the above, Convair was requested to make an informal study to investigate the tactical advantage of range extension, on an all out mission, through the means of additional fuel, either internal or external, and the possible benefit of using "so-called" supersonic tanks. For the purpose of the study, the following ground rules were adopted:

1. A "supersonic dash" mission which includes take-off, climb to 40,000 ft. and cruise out to interception point, using full military plus A. B. power.

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INTRODUCTION (Continued)'

2. If the interception point occurs at a greater radius than can be obtained by supersonic dash only, then the distance would be covered by an optimum combination of subsonic cruise and supersonic dash.

3. Return by most economical means.

4. If external tanks are to be carried in the combat mission, they shall not be dropped in day-to-day training or tactical identification missions.

5. It is assumed that combat tanks would be compatible with supersonic flight but that they would be dropped prior to actual combat in the case of hostilities.

Various combinations of internal and external tanks, both subsonic and supersonic have been studied to determine the optimum configuration for a typical air defense problem. An operations analysis study (Convair Report OR-8-43) has been made and is included as an addendum in the back of this report. A typical SAGE sector was used with the expected 1959-60 interceptor squadron deployment against Mach 0.9 bombers at 40,000 ft. The effect of radar range of 390 n. mi. and an extended radar range of 600 n. mi. was examined.

In order to provide for supersonic flight C.G. control, the existing internal tanks of the F-106A have a greater fuel capacity than required for the area intercept mission. By increasing the capacity of Tank #1 and including the capacity of the aft transfer tank, it is possible to increase the internal fuel load of the airplane by 300 gallons at a small cost of minor changes to the fuel system. The importance of the additional internal fuel is shown in this study.

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INTRODUCTION (Continued)

No allowance is made in this study for increased strength due to higher combat weights or increased stiffness as a result of external tanks carried at supersonic speeds. It is assumed that reduced load factors would be acceptable in the early part of a flight wherein the fuel is used at a much greater rate than would be the case in the area-intercept mission.

Allowance has been made, however, for the increased internal fuel plumbing; stronger ejection and local reinforcements for attachment of supersonic tanks; and increased tire and landing gear strength for the higher take-off weights. If adopted, some of these weight items would be reflected in a degradation of the altitude performance of the basic aircraft.

None of the features presented herein are presently being incorporated in the F-106A airplane.

This study is limited in scope; a more detailed study of all aspects of the aircraft and its performance in conjunction with the air defense problem should be made prior to incorporation of any of the revisions to the aircraft discussed by this report.

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SUMMARY

1. Addition of either internal or external fuel to the F-106A increases its radius of action and supersonic dash capabilities.

2. The addition of fuel will however tend to reduce the combat ceiling, time to climb, and zoom performance and the addition of weight to the basic airplane will affect the guaranteed ceiling. Therefore, any consideration of adding external tanks or internal fuel will have to weigh the conflicting performance requirements.

3. 300 gallons of fuel can be added in the existing internal fuel tanks by making minor changes to the fuel system.

4. The existing Royal jet 230 gallon external tanks are for the "ferry" condition only and cannot be used for a tactical mission extending into the supersonic regime because they are restricted to M = .95.

5. A new 230 gallon supersonic tank would allow flight in the supersonic regime. This would be an advantage in allowing training at supersonic speeds without dropping tanks.

6. Addition of supersonic tanks to the airplane would dictate a wind tunnel and flight test program to determine accurately the effects on performance, stability and control.

7. The effect of several possible modifications on the supersonic dash radius at 40,000 ft. can be summarized as follows:

(a)	F-106A	140 N. M.
(b)	F-106A plus two 230 gal. subsonic tanks	190
(c)	F-106A plus 300 gal. internal fuel	195
(d)	Item (c) plus two 230 gal. subsonic tanks	260
(e)	Item (c) plus two 230 gal. supersonic tanks	275
(f)	Item (c) plus two 550 gal. supersonic tanks	385

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SUMMARY (Continued)

Therefore it is evident that the desirable amount of additional fuel cannot be defined on the basis of supersonic dash radius only.

8. For a typical air defense problem (a raid on New York), the following percentages of combat available interceptors can engage in the battle: (based on the 6 configurations studied in this report).

Configuration	Radar Range	
	390 n. mi.	600 n. mi.
1. Clean airplane (existing F-106A)	65%	65%
2. (1) above plus (2) 230 gal. subsonic tanks	74%	91%
3. (1) above plus 300 gal. internal fuel		
4. (3) above plus (2) 230 gal. subsonic tanks	87%	100%
5. (3) above plus (2) 230 gal. supersonic tanks		
6. (3) above plus (2) 550 gal. supersonic tanks	90%	100%

9. A significant increase in tactical effectiveness is obtained by adding fuel up to and including the 300 gallon internal plus (2) 230 gallon external tanks. The gain, by adding fuel beyond this point, is insignificant for the conditions studied.

10. For the conditions studied, the use of larger 550 gal. supersonic tanks appears to be unwarranted because as the radar coverage is increased, more time is available for subsonic cruise out and fuel requirements become less critical.

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SUMMARY (Continued)

11. The percentages shown in (8) above are based on a limited set of conditions. No consideration has been given to such conditions as altitude extremes, target variations, ECM, multiple pass capabilities, and evasive action.

12. Most of the tactical advantages of the heavier configurations are gained from those interceptors based great distances from the defended area. The more conservative approach of limiting recruiting to squadrons based close to the defended area reduces greatly the advantage of the heavier fuel configurations. The external tank equipped aircraft show little or no distribution advantage over the 10, 010 lb. internal fuel clean airplane because their advantages are gained late in the air battle.

13. Though the 230 gal. supersonic tanks appear to have little advantage over the subsonic form in supersonic dash radius or tactical effectiveness, the supersonic form is to be preferred because of the higher level of training permissible during peacetime operations with the aircraft in the alert status configuration.

14. It should be pointed out that the use of supersonic tanks on the F-106A will involve additional programming in the SAGE system. Since all peacetime missions will be flown without dropping tanks whereas wartime missions will drop tanks when empty, two different configurations with substantially different performance characteristics must be controlled. As a result, it will be necessary to have two tactical mission programs stored in the SAGE computer. In addition it may be advisable to have a program using a clean airplane for pilot training at higher performance such as live missile firing practice.

15. This study assumes use can be made of various amounts of subsonic and supersonic speed cruise-out to take advantage of the optimum of maximum radius and minimum time of intercept for distances beyond the supersonic dash radius. Programming of the SAGE computer to utilize this capability may be elaborate and unreasonably complex.

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SUMMARY (Continued)

16. Considerations of the use of any external tank as a standard tactical configuration, under the assumption that the tanks are dropped prior to combat, should include the problem of attaching and refueling new tanks during the required fifteen minute turn-around period, as well as the additional communications required to maintain cognizance over which aircraft are in each configuration.

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DISCUSSION

In order to determine the optimum configuration for the "supersonic dash" mission, various combinations of additional internal fuel and external tanks have been studied. These include both subsonic and supersonic tanks and are shown below together with fuel capacities and takeoff weights.

Configuration	Fuel - Gal. Internal	Fuel - Gal. External	Total Fuel-lb.	T. O. Gross Weight
1. Basic F-106 A Airplane (no tanks)	1240	-	8060	33,125
2. Basic + (2) 230 gal. tanks (existing Royal Jet tanks) (limited fuel usage in climb)	1240	460	10960	36,360
3. Basic + 300 gal. internal	1540	-	10010	35,125
4. Basic + 300 gal. internal + (2) 230 gal. subsonic tanks (improved fuel usage in climb)	1540	460	12909	38,400
5. Basic + 300 gal. internal + (2) 230 gal. supersonic tanks	1540	460	12909	38,550*
6. Basic + 300 gal. internal + (2) 550 gal. supersonic tanks	1540	1100	17134	43,750*

* Based on rough area estimates.

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MISSION BASIS

Performance of the above 6 configurations was determined for the "supersonic dash" mission. A detail breakdown of the mission is as follows:

1. Warm-up, take-off, and accelerate to climb speed (standard Mil. Spec. allowance, i. e. 1 minute at military plus afterburner power plus 2 minutes at normal rated power at sea level).

2. Climb on course to 40,000 ft. altitude, military plus afterburner power.

3. Cruise out at best cruise speed at 40,000 ft. altitude.

4. Accelerate on course to M = 2.0 at 40,000 ft. altitude.

5. Continue at M = 2.0 to point of intercept at 40,000 ft. altitude.

6. Combat 2 minutes at M = 2.0 at 40,000 ft. altitude, military plus afterburner power.

7. Cruise home at best cruise speed and altitude.

8. Reserve fuel as specified for A. D. C. performance charts, i. e. 5 minutes operation at sea level at traffic pattern power plus 5 minutes at cruise altitude at maximum continuous power.

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MISSION DATA

The above mission was plotted on the basis of radius versus time to point of intercept for the six configurations studied.

The sample plot on the following page shows how these plots were constructed. By varying the amounts of subsonic and Mach 2 cruise in the above mission a locus of radius versus time to point of intercept point can be established.

The upper end of the curve represents the case with subsonic cruise out. The lower right hand corner of the solid portion of the curve represents the case with acceleration to Mach 2 cruise out.

For the lower solid portion of the curve the supersonic tanks are dropped when empty, then the airplane accelerates to Mach 2 and continues to the intercept point.

Any intermediate point on the solid portion of the curve can be obtained by various combinations of subsonic and Mach 2 cruise.

The lower dotted portion of the curve represents the case where tanks are dropped at the top of the climb before beginning acceleration to Mach 2.

Intermediate points between the lower dotted and solid portion of the curve can be obtained by dropping the supersonic tanks with some fuel remaining, after varying amounts of acceleration with tanks attached.

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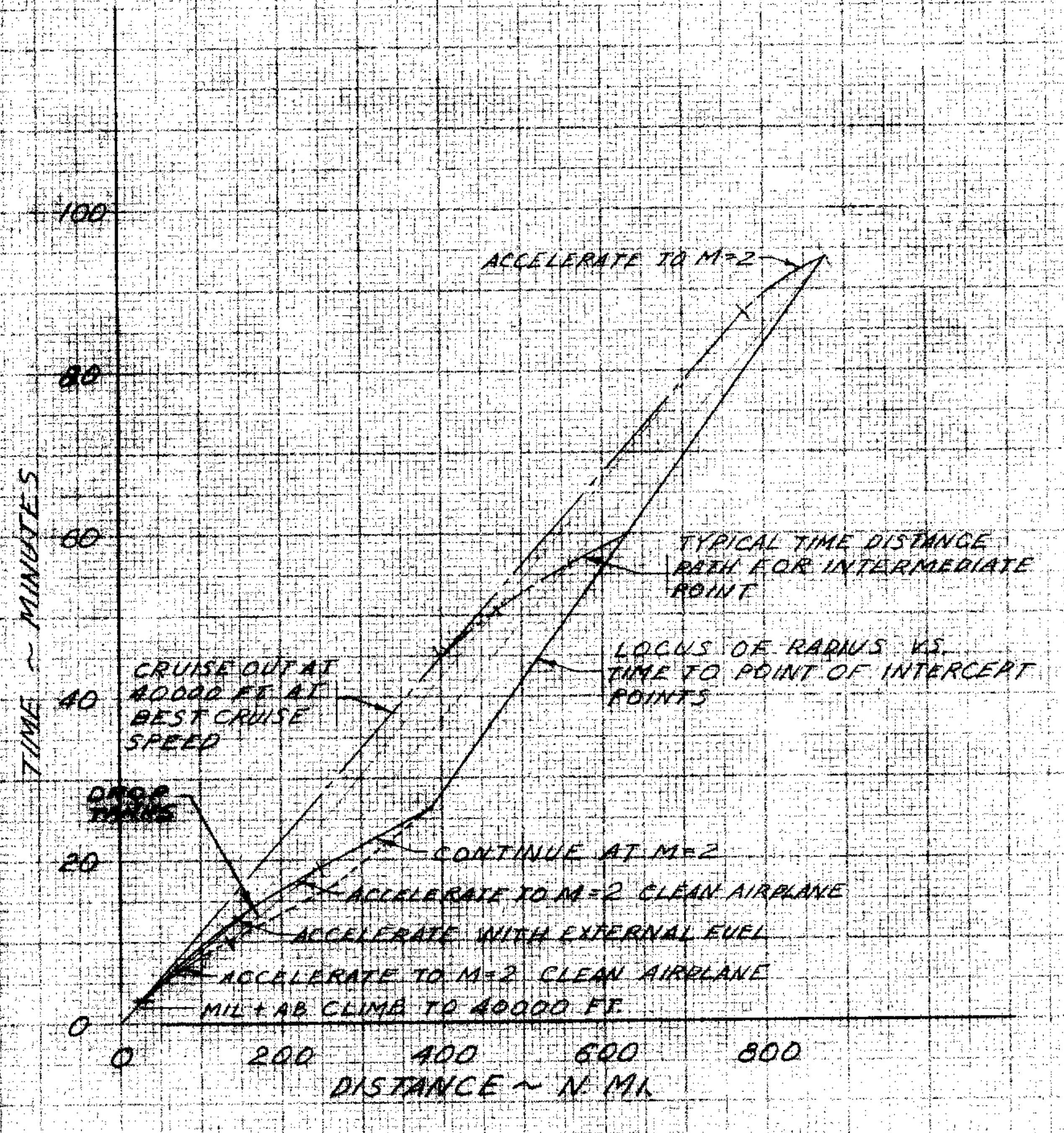
TIME VS DISTANCE

AIRPLANE PLUS 2-550 GALLON SUPERSONIC EXTERNAL TANKS



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The plots on the following two pages show radius versus time to point of intercept. for the six configurations studied.

The subsonic cruise out portion of the curves has been omitted for clarity.

These curves are the basis for the operations analysis study included in the back of this report.

Note that the important point on the curves is the lower right hand corner. This represents the maximum supersonic radius that can be obtained and in the least time.

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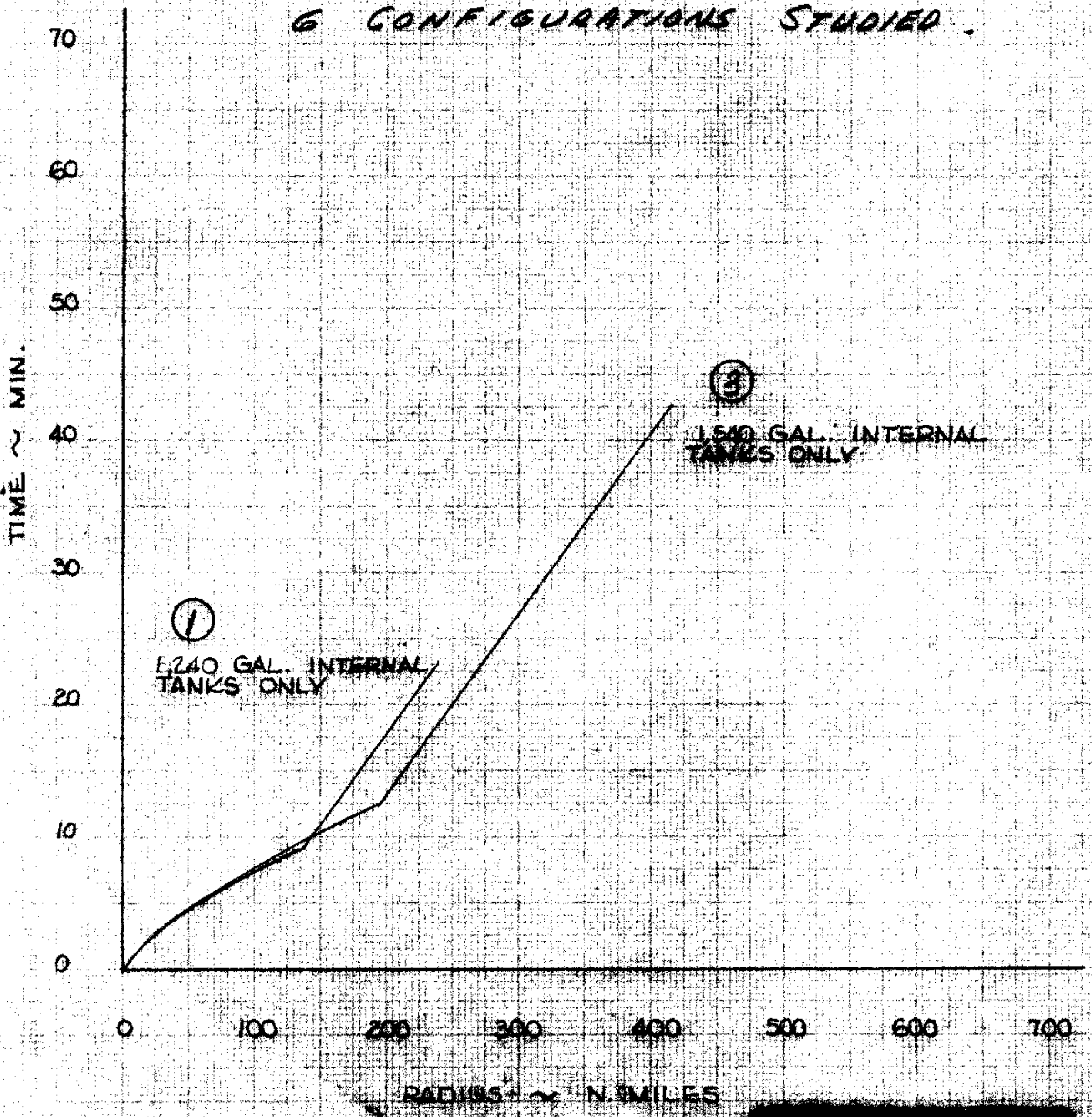
[redacted]F-106A AIRPLANE

TIME VS. DISTANCE PLOTS

NOS. CORRESPOND TO THE 6 CONFIGURATIONS STUDIED.



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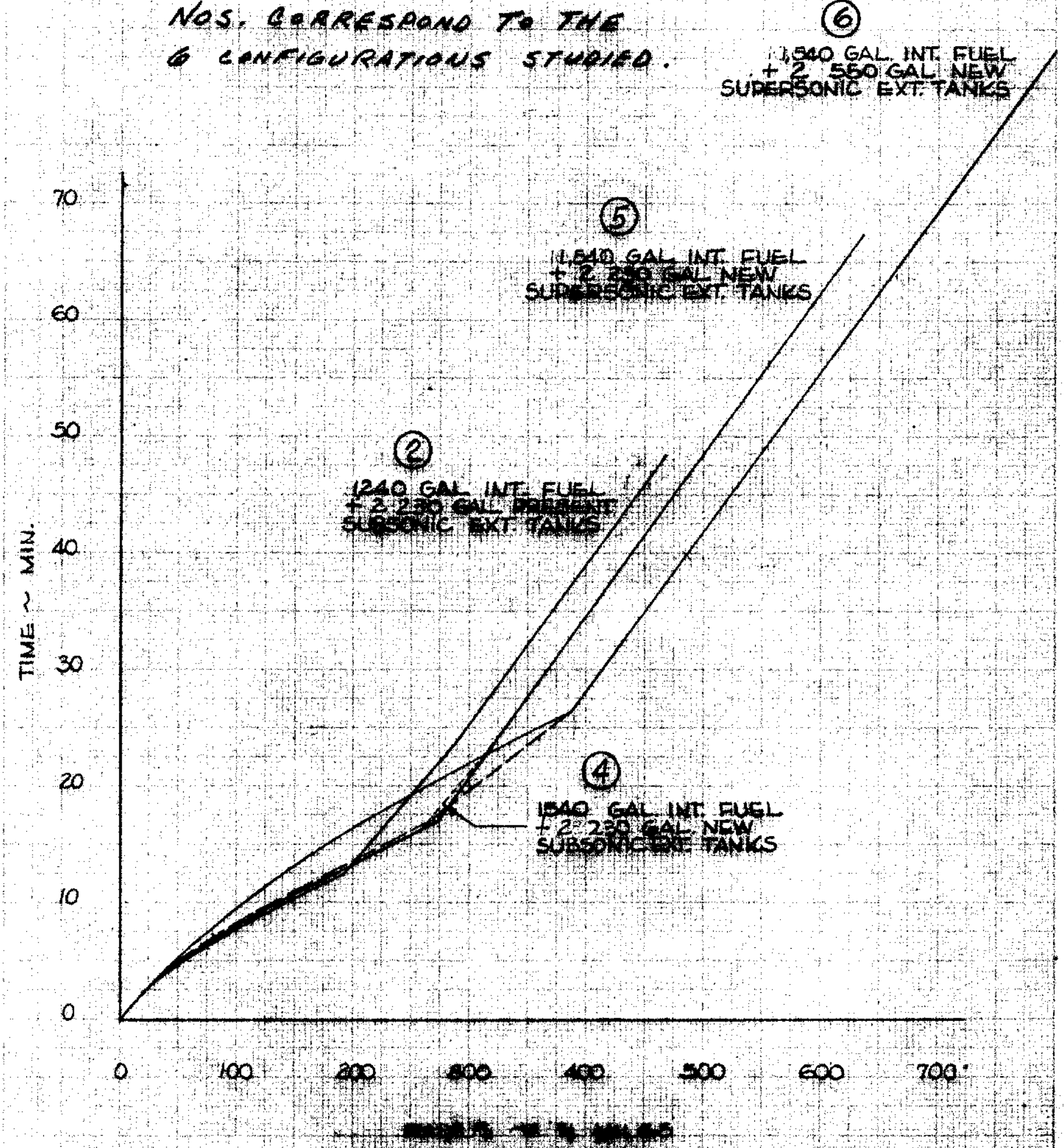
F-106A AIRPLANE

TIME VS. DISTANCE PLOTS



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The following table shows the weights, usable fuel, and tank drag numbers used as a basis for the mission calculations.

Configuration		1	2	3	4	5	6
Take-off Gross Weight	lb.	33125	36360	35125	38400	38550	43750
Internal Usable Fuel	lb.	8060	8060	10010	10010	10010	10010
External Fuel (Usable and Unusable)	lb.	-	2899 (1)	-	2899 (1)	2899 (1)	7124
External Usable Fuel	lb.	-	2808 (1)	-	2808 (1)	2808 (1)	6906 (2)
Droppable Weight (Tanks, Pylons, and Trapped Fuel)	lb.	-	427	-	467	537	1274
External Tank Drag							
ΔC_D, $M < 0.95$		-	.0019	-	.0019	.0019	.0034
ΔC_D, $M > 1.0$		-	-	-	-	.0041	.0073

NOTES:

1. Each 230 gallon Fletcher Royal Jet tank as installed on the F-106A airplane can be filled with only 223 gal. of JP4 fuel and each has 7 gal. of trapped fuel. The tanks of configurations 4 and 5 are assumed to have the same characteristics.

2. The trapped fuel of the 2-550 gal. external tanks was estimated as follows:

$$\text{Trapped Fuel} = \frac{550}{230} \times 14 = 33.5 \text{ gal.}$$

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PERFORMANCE BASIS

These studies have been made without benefit of pre-design layout of the supersonic tanks. For the purpose of this study, it is assumed that all of the external tanks have the same fineness ratio and location as the present Fletcher Royal Jet 230 gallon tanks. This approach is conservative and better performance should be realized with an improved fineness ratio.

The drag increment for the external tanks is based upon wind tunnel tests of the F-106A with (2) 230 gallon external tanks as run in CWT test No. 528. The 230 gallon tank data were changed by a tank frontal area ratio to provide a drag estimate for the 550 gallon external tanks.

The performance used in this study for the clean airplane is based upon Convair Aerodynamics Memorandum A-8-119, "Estimated Performance Characteristics of the F-102B Airplane", dated 1 August 1955.

SPEED VS. ALTITUDE

A plot of Mach Number vs. altitude is shown on the following page. It compares the performance capabilities of the F-106A with both subsonic and supersonic tanks. It shows the performance penalties that can be expected if supersonic tanks must be carried in day-to-day training and tactical identification missions.

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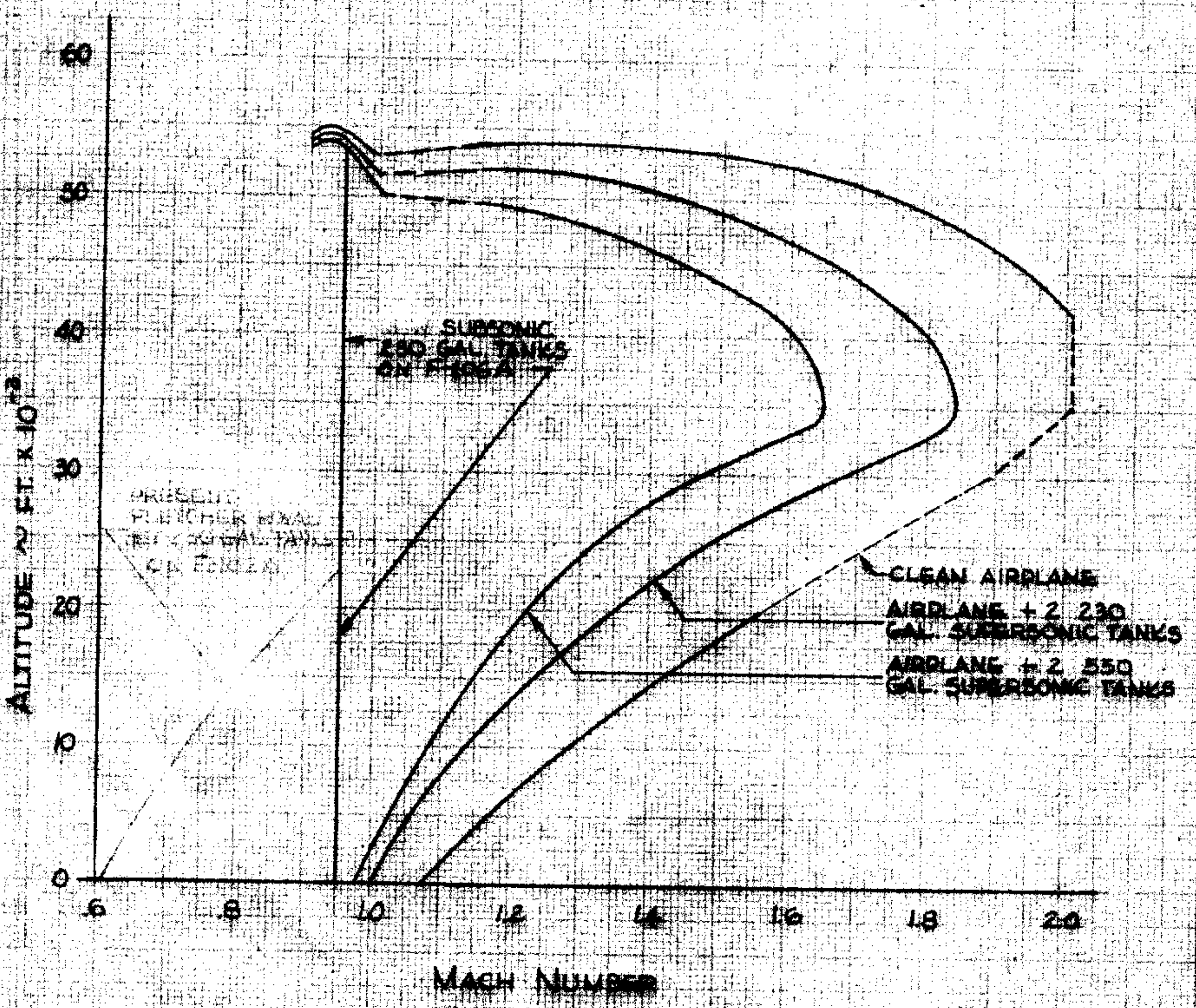
F-106A AIRPLANE
MACH NUMBER VS ALTITUDE
AIRPLANE PLUS SUPERSONIC EXTERNAL TANKS

GROSS WEIGHT = 30,000 LBS.



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FUEL SYSTEM

The following table shows changes required to add 300 gallons of fuel to the internal tanks of the F-106A airplane:

1. Raise pilot valves in tank #1
2. Add the following components in "T" tanks:
   - (2) Air shut-off valves
   - (2) Level control shut-off valves
   - (5) Level control pilot valves
   - (2) Refuel pressure operated pilot valves
   - 60 Feet of 1/4" tubing and fittings

The tank fuel quantities will change as follows:

Tank	Present Gals.	Proposed Gals.
1	235	308
2	316	316
3	440	440
F	227	227
T	0	227
Transfer Line	22	22
	1240	1540

Assuming the above changes have been made and it is desired to only take on 1240 gallons (for the area intercept mission) a more complicated fuel system will be required. Preliminary discussions indicate the feasibility, but further detail studies would be necessary to determine the extent of the changes required to obtain this feature.

The existing Fletcher Royal Jet tanks have definite limitations when applied to the tactical mission. These tanks are intended for the "ferry" condition only as per model specification ZD-8-005. The requirement does not exist to use these tanks on a tactical basis.

The tanks are inadequate because of the low pressure (10 psi) which is insufficient to provide fuel usage in the climb. In addition, they are limited to subsonic speeds (Mach 0.95).

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The existing external tanks are limited to the following load factors:

Tanks Full (max. takeoff gross weight)
-1 to + 3 g symmetrical pullout up to M = 0.95
0 to + 2.3 g rolling pullout up to M = 0.95

Tanks Empty (Basic mission gross weight plus empty external tanks)
-3 to + 5.0 g symmetrical pullout up to M = 0.95
0 to + 3.9 g rolling pullout up to M = 0.95

An improved subsonic tank has been proposed which would have higher pressure (20 psi) and would be good for Mach 0.95 at sea level. This tank would provide maximum fuel usage in the climb and could be dropped soon after reaching the top of the climb. This would allow the airplane to accelerate to Mach 2 much sooner.

EXTERNAL TANKS (Supersonic)

It is assumed that these tanks will be designed to operate in the supersonic region but that they will be dropped prior to actual combat.

They should be designed to provide maximum fuel usage in the climb and comply with the following design criteria:

DESIGN AND OPERATIONAL CRITERIA FOR EXTERNAL TANKS

The design criteria is applicable to either the subsonic or supersonic tanks and provides a compromise between operational requirements, structural weight, and minimum rework to existing aircraft.

A. Operation of the armament doors and missile launchers should be permitted with the tanks on.

   1. This is primarily for off status or training aircraft.

   2. There appears to be no need for actual firings with tanks on.

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B. Any speed or altitude to the restricted speed of the present F-106A airplane (179° F stagnation temperature). *

C. Normal load factor maneuver (symmetrical) *

(1.) + 4 g to -1 g at any speed to the restricted speed. (Stag. temp. of 179° F).

These limits are compatible with the fire control system requirements.

D. Roll Rates: Limit roll rates to 80°/sec.

(1.) This value is in excess of the fire control requirements (70°/sec.)

(2.) This value was found acceptable for the tactical mission of the airplane during flight evaluation testing.

E. Active Status Aircraft

(1) If external tanks are required for an effective combat mission, then all on-status aircraft will be tank equipped. Exclusive of active hostility periods, tanks must be retained throughout the mission.

F. Off-Status Aircraft

(1) A certain amount of training could be accomplished using aircraft without external tanks; however, the installation of tanks could delay the advancement of these aircraft to active or combat ready status in an emergency.

Following is Convair Report OR-8-43, a complete study of the effectiveness of the various configurations for a typical air defense problem.

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* with supersonic external tanks only.

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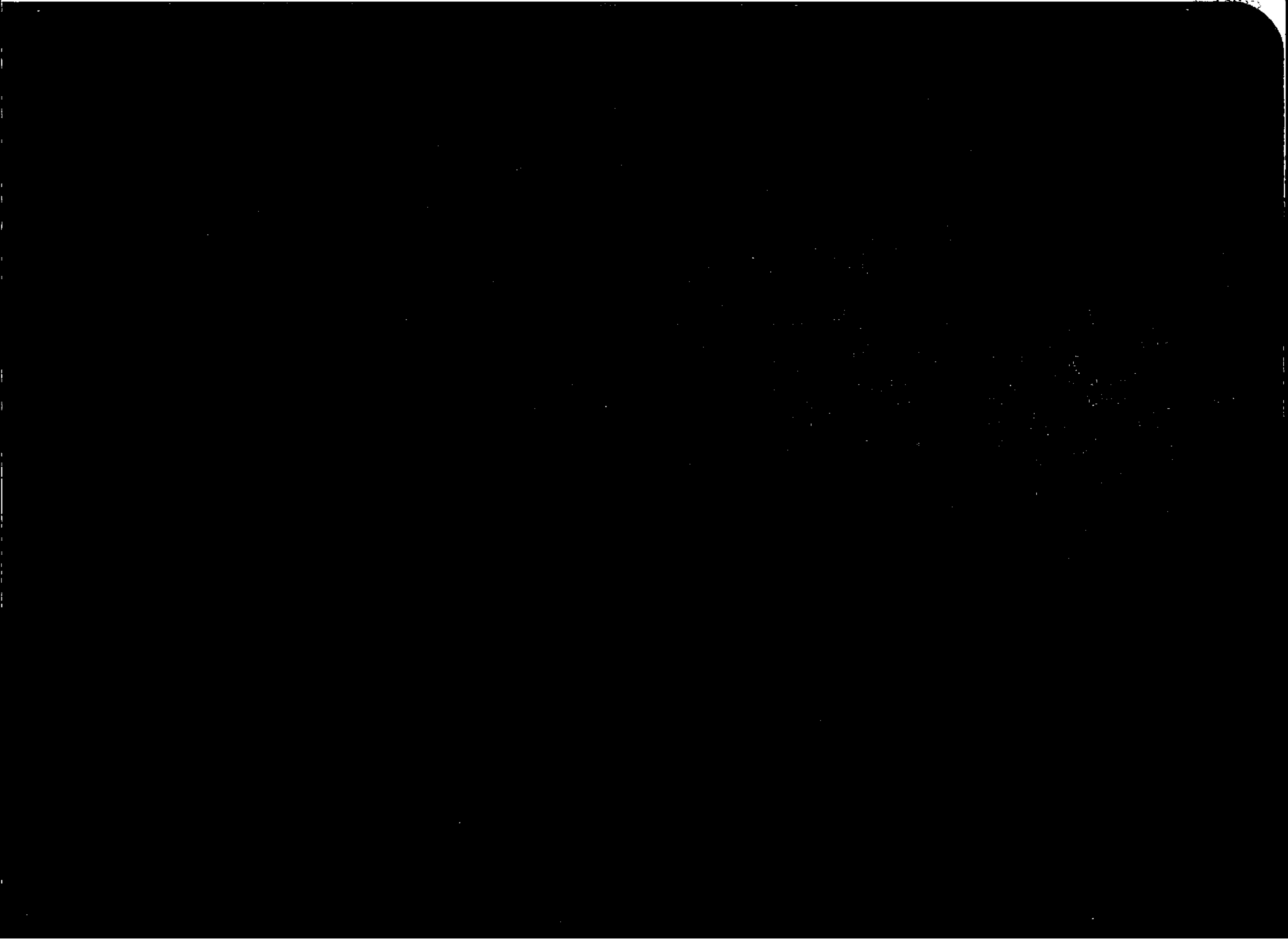
ZP-8-039 *APP. A



CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-039
DATE 30 Mar. 1956
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L. G. Avery ...-1 4-4-67

TITLE

BRIEF DISCUSSION OF THE STATUS OF PILOT ESCAPE SYSTEMS FOR ADVANCED F-102B STUDIES

ZP-8-039

[redacted]

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SUMMARY

The purpose of this report is to review briefly the present status of pilot emergency escape systems which apply to interceptors of the F-102B and proposed advanced F-102B ($M = 2.3$) types. Comparisons of the systems are made in a very general manner.

Appended data is included for reference purposes.

INTRODUCTION

Although notable improvements are being made in ejection seat design with respect to pilot survival, the seat as such is still a questionable means of escape even in the higher subsonic speed range. Figure 1 shows a predicted envelope for safe escape by means of seats and capsules.

It is understood that the USAF plans to make an escape capsule of some type mandatory for aircraft capable of speeds of 600 knots (indicated) or greater. Escape capsules of various types, with special reference to the Advanced F-102B, are discussed in this report.

FACTORS INVOLVED

Some of the major factors involved in the use of any escape system are as follows:

1. Initial deceleration loads imposed on pilot during escape. Human tolerance decreases with altitude for given indicated speed due to increase in duration of peak 'g' (Ref. 1).
2. Effects of aerodynamic heating on pilot during initial exposure to air stream (followed by extremely low temperature shortly afterward).
3. Hypoxia, aero-embolism, explosive decompression, impact pressure injuries and other pressure effects are hazards.

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4. Spinning, tumbling or other gyrations which may produce injuries due to limb flailing and hydrostatic effects.

5. Protection in unfavorable environment after return to surface of earth, (open water, bad terrain, etc.).

6. Protection against injury to pilot due to striking portions of airframe during escape.

Of all the factors involved in the above list, one of the most difficult to overcome with open ejection seats, or protected seat capsules (discussed later) is initial deceleration.

Reference 1 shows that the drag-weight ratio of the usual ejection seat is unfavorable in this regard. Drag can only be reduced to that of the pilot alone and this limit is approached in the Convair "Supersonic Seat". The weight of the seat can be increased and favorably directed rocket thrust may be added. However, the degree of success which can be attained with these and other fixes is not known, but appears to be more limited than is the case with the capsule types.

Protective equipment can surely be more conveniently installed in a capsule than by adding personal equipment, which will tend to further immobilize the pilot, as is now the case with the ejection seat.

ESCAPE CAPSULES - GENERAL REQUIREMENTS

In consideration of the above factors involved in high speed and/or high altitude escape, the following general requirements for a capsule may be stated:

Capsule should:

1. Minimize the possibility of injury to the pilot.
2. Maintain suitable environment throughout the escape cycle.
3. Have potential of providing escape means at low, as well as high altitudes and all speed regimes.

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4. Operate as automatically as possible.

5. Have high degree of stability, and in certain instances, be controllable.

Capsule Types

It is not known how many types of capsule are under study at this time throughout the industry and in military agencies. Apparently Douglas has given consideration to the separating-nose configuration (Ref. 2.).

We have a report on Goodyear's clamshell type capsule (Ref. 5.) and benefit of Stanley Aviation's work for the USAF on a canopy type capsule (Ref. 4.). Out of the information on hand, it is believed that the various types of escape capsules may be classified as follows: (Ref. Figure 2A, 2B & 2C.).

1. Protected seat type (Goodyear Clamshell)

2. Canopy Type (Stanley Aviation)

3. Separable-Nose Type

Some of the apparent advantages and disadvantages of these types as we see them are as follows:

PROTECTED SEAT TYPE I (REF. GOODYEAR CLAMSHELL)

ADVANTAGES	DISADVANTAGES
a. Provides protection against the majority of items listed under "Factors Involved" above.	a. Susceptible to the initial deceleration problem in somewhat the same degree as an open seat.
b. Capsule is relatively small and, therefore, should be less susceptible to combat damage	b. Restricts pilot's vision side and aft.

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ADVANTAGES

DISADVANTAGES

c. Requires that cockpit or canopy width equal to clamshell width continue to point above pilot's head.

d. Pressure sealing the moving parts of the clamshell in closed position.

e. Requires definite pre-ejection procedure for pilot to get in position.

f. Stability is questionable.

g. Airplane weight penalty approaching Types II and III.

h. Does not appreciably improve ability to clear vertical tail when compared with open seats.

CANOPY TYPE II (REF. STANLEY AVIATION)

ADVANTAGES

a. Provides protection against the majority of items listed under "Factors Involved" above.

b. Relatively good aerodynamic shape (stability).

c. Better deceleration characteristics than Type I (drag/weight ratio).

DISADVANTAGES

a. Mechanical complexity involved in this arrangement appears greater than Type III with respect to flight controls, pressure sealed trap doors, design of separable elements, etc.

b. Relatively large area - more susceptible to combat damage than protected-seat type.

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SEPARABLE NOSE TYPE III

ADVANTAGES

a. Provides protection against the majority of items listed under "Factors Involved" above - probably greater than Types I & II.

b. Relatively good aerodynamic shape. Perhaps better supersonic shape than Type II. (Stability and control).

c. Better deceleration characteristics than Types I or II.

d. Minimum of pre-ejection procedure required of pilot

DISADVANTAGES

a. Larger than Types I & II, therefore, more susceptible to combat damage.

b. Size of capsule and proposed scheme (Figure 2C) to separate radome to reduce weight after deceleration increases hazard to people on ground.

c. Low altitudes escape, especially in dive attitudes, may be less effective than Types I or II.

Concluding Remarks:

Much more analytical and experimental work will be required to begin to resolve the escape problem. However, assuming that the capsule is the answer, there still remains the problem of which configuration to use.

It appears that aircraft with the speed-altitude capabilities of the proposed advanced F-102B that an escape system of the canopy type (II) or the separable nose type (III) would serve best.

Although the lines of demarcation are broad, in Figure 1 we have attempted to spot the probable areas of usage for the various escape means on the speed - altitude curves of the F-102B and advanced F-102B.

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"Escape from High Performance A/C"
By John W. Goodrich, Capt., USAF
Plans Office, Directorate of Research
January 9, 1956

2. WESTERN AVIATION - December 1955*

"Escape Systems"
By A. M. Mayo, Douglas Aircraft Company

3. CONVAIR, A Division of General Dynamics Corporation (San Diego)

"Notes on the Design of the Crew Compartment and on the Escape Method of the Nuclear Powered Attack Seaplane" dated February 22, 1956
By George C. Haag, Preliminary Design, Convair

4. STANLEY AVIATION CORPORATION

Document #184 "Preliminary Design Study Report - Seat Capsule; Pilot's Emergency Escape - USAF Contract AF 33(600)-22142 dated Sept. 16, 1953

5. GOODYEAR AIRCRAFT CORPORATION

Ejectable Seat Capsule Program
GER 6843-R-995 dated 15 June 1955

* Included in Appendix 'A'

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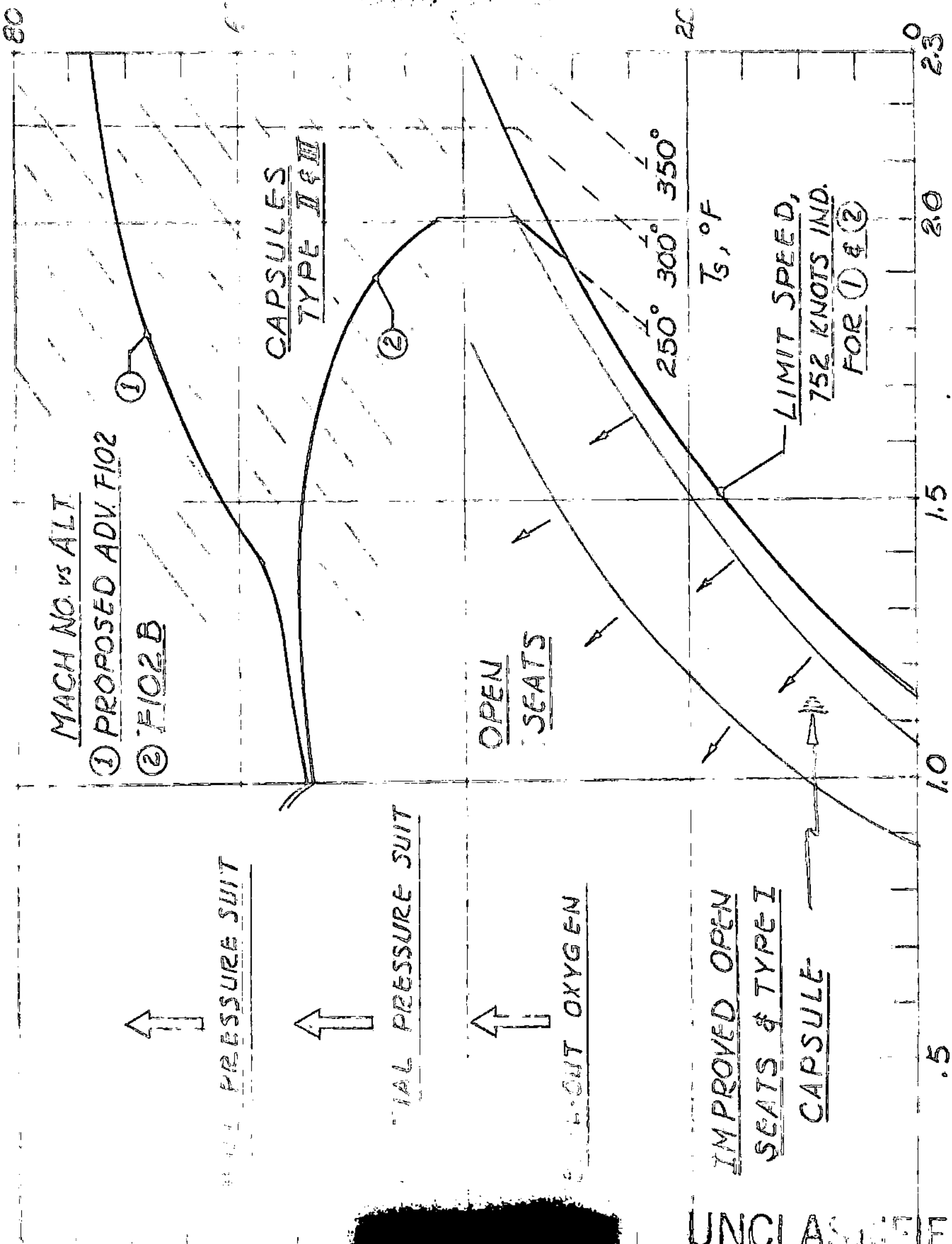
FIG 1



ESCAPE ENVELOPE

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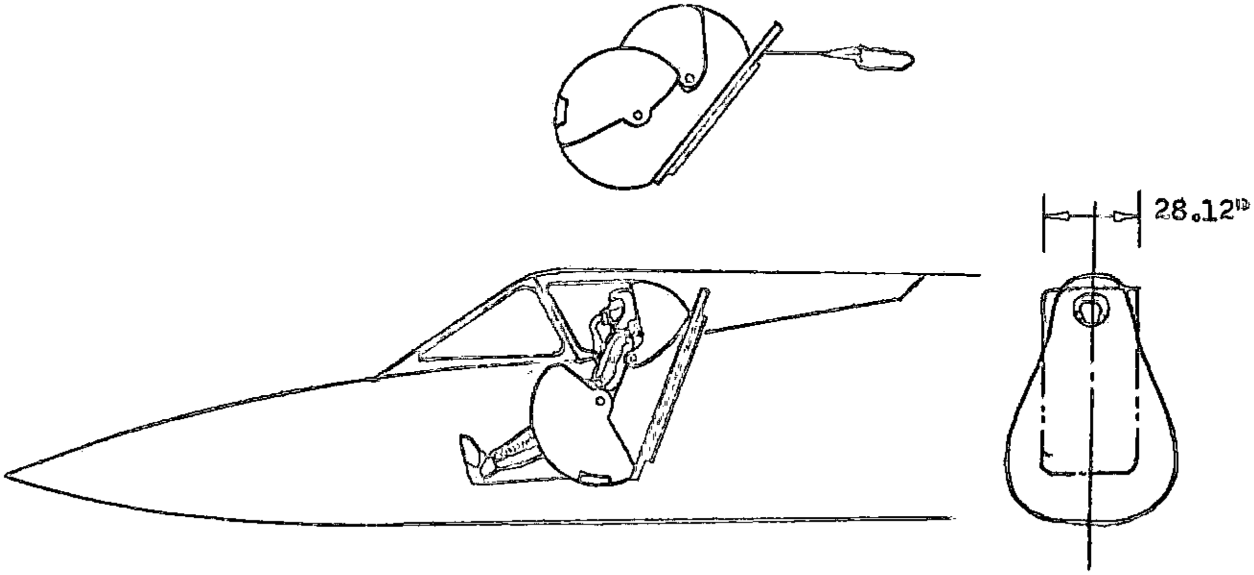
PROTECTED SEAT CAPSULE - TYPE I
(CONVAIR)

FIG. 2A

Weight of ejectable capsule, Lb.	≈ 540*
Main chute Dia., Ft.	–
Drouge Chute Dia., Ft.	3.16
Rate of descent S.L.	–
Ref. area, Sq. Ft.	13 (estimated)



SEQUENCE

- Pilot pulls face curtain which triggers mechanism to retract seat, draw pilot's feet into lower shell, and initiates closing of doors.
- As doors reach closed position door seals inflate, and canopy is jettisoned. Jettisoned canopy arms catapult and allows face curtain to be pulled all the way down.
- Capsule ejects, chutes deploy in sequence.

* Estimated total weight penalty for airplane: approx. 160 lb.

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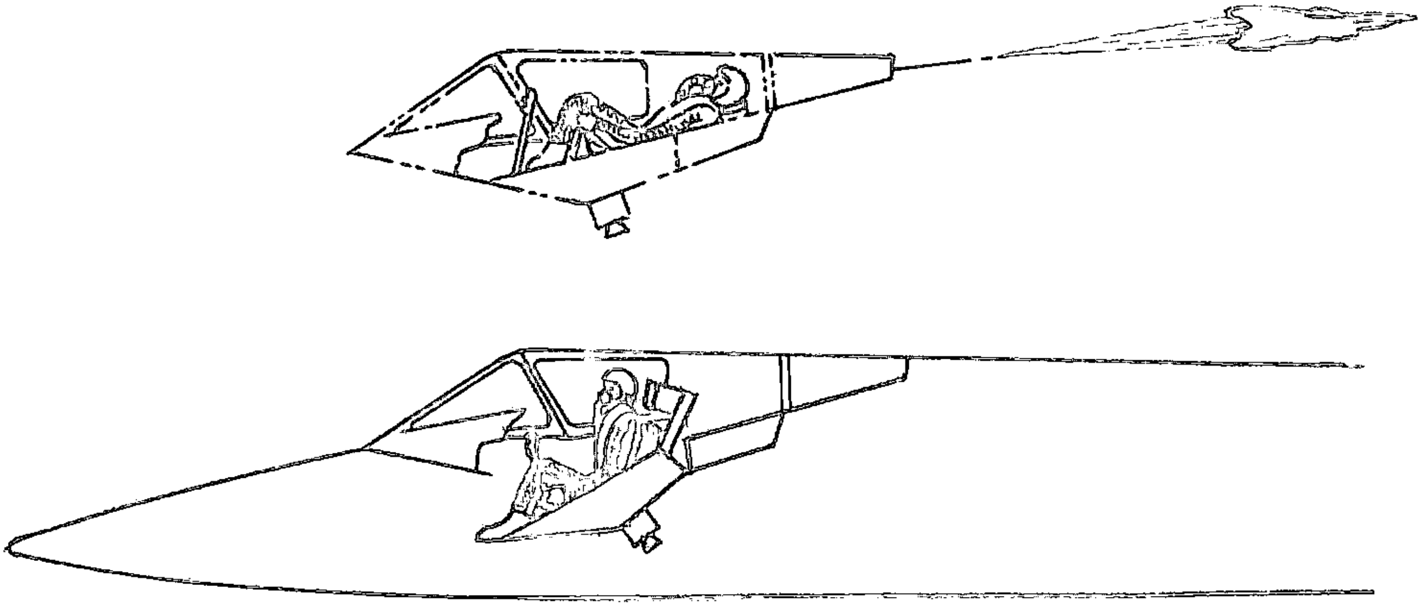
CANOPY CAPSULE - TYPE II

FIG. 2B

(STANLEY AVIATION CORP.)

Wt. of ejectable capsule, Lb. ≈	1,000*
Main chute Dia., Ft.	48
Drouge Chute Dia., Ft.	3.44
Rate of descent S.L. Ft/Sec	24.8
Ref. area, Sq. Ft.	6.4



SEQUENCE

- Pilot draws legs into lower door
- Combination of seat action and closing door places pilot in semi-supine position
- Rocket motor ejects capsule
- Drouge chute and main chute deploy in sequence

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* Estimated total weight penalty for airplane $<$ 200 lbs.

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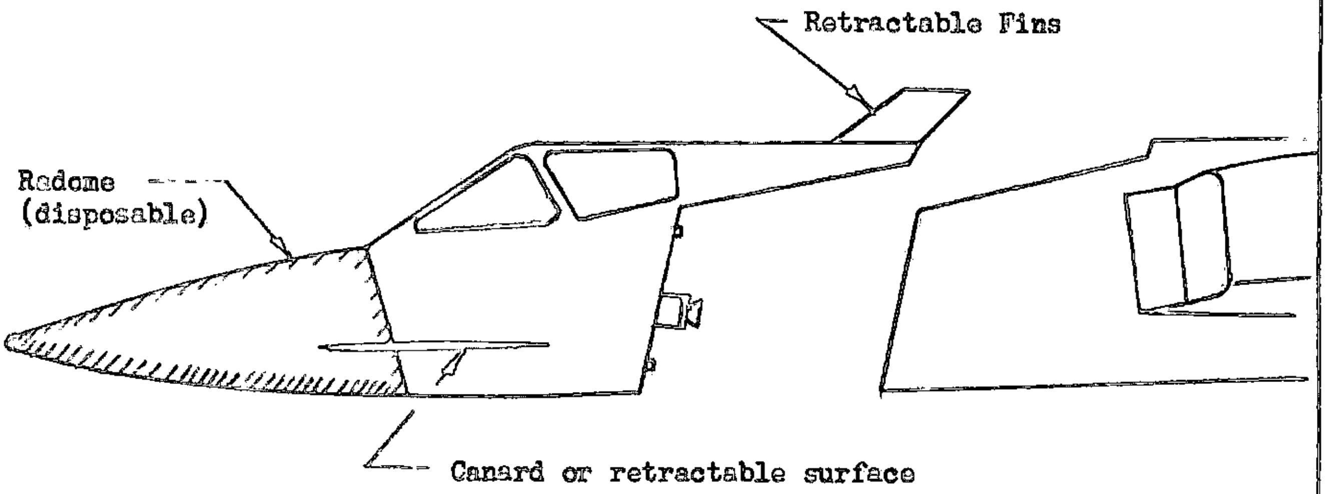
SEPARABLE NOSE CAPSULE - TYPE III

FIG. 20

Initial weight of capsule, Lb.	3,122*
Net weight (initial - disposable), Lb.	2,305
Main Chute Dia., Ft.	72
Rate of descent, S. L., Ft/Sec.	25.4



SEQUENCE

- Pilot actuates panic button
- Rocket motor separates capsule
- Radome and contents ejected after main chute is deployed

*Total weight penalty for airplane undetermined.

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APPENDIX 'A'

1. Escape Systems (Reference 2.)

2. Escape from High Performance Aircraft (Reference 1)

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ESCAPE FROM H.S. A/C

WADC Technical Note 56-7

Escape From High Performance Aircraft

JOHN W. GOODRICH, Captain, USAF

Plans Office
Directorate of Research

9 January 1956

Wright Air Development Center
Air Research and Development Command
United States Air Force
Wright-Patterson Air Force Base, Ohio

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Wright Air Development Center
Air Research and Development Command
United States Air Force
Wright-Patterson Air Force Base, Ohio

WADC-TN-56-7
9 January 1956

Plans Office
Directorate of Research

ESCAPE FROM HIGH PERFORMANCE AIRCRAFT

1. PURPOSE

1.1 An objective analysis of the problem of escape from high performance aircraft is presented in terms of human tolerance to acceleration correlated with pertinent aerodynamic and physical factors. The maximum linear deceleration experienced during escape is computed for the conventional ejection seat system as a function of calibrated airspeed[1], altitude, and Mach number. The decay of deceleration is also computed as a function of altitude for a given initial calibrated airspeed. Correlation of these computed physical forces with human tolerance to acceleration reveals the critical human limitations associated with existing conventional ejection seats for successful escape from high performance aircraft. This study will aid in determining the scope of knowledge required to support escape systems research and development.

2. FACTUAL DATA

2.1 One aspect of the limitations of the ejection seat as a tolerable means of escape from high speed aircraft is the magnitude of linear deceleration caused by the airloads imposed on the ejected occupied seat as it enters the airstream. Human tolerance to acceleration has been fairly well defined as a result of the efforts of Dr. John P. Stapp, Holloman Air Development Center, Doctors Savely, Edelberg, and Henry of the Aero Medical Laboratory, WADC - and others. (References 1, 2, 3, and 4). The human tolerance to linear deceleration in relation to rate of on-set, magnitude, and duration for transverse g (in a stable escape system), as defined by the Biophysics Branch of the Aero Medical Lab, WADC, is presented in Fig. 1. The introduction of positive or negative g forces will lower the human tolerance depicted in Fig. 1. This study does not evaluate the effects of tumbling during escape, or the phenomena of the build-up to the initial peak deceleration (rate of on-set), and is limited to an analysis of a stable escape system.

---

[1] Calibrated airspeed is defined as the indicated airspeed corrected for installation and instrument errors.

2.2 The maximum linear deceleration experienced by the occupied ejection seat as it enters the airstream is determined by the ratio of the drag to the weight of the occupied seat. There are three main variables affecting the magnitude of the g forces at various speeds. These are:

2.2.1 The exposed frontal area of the occupied seat normal to the airstream.

2.2.2 The weight of the occupied seat, and,

2.2.3 The variation of the drag coefficient with Mach Number.

In order to determine the airload imposed on the seat, a drag coefficient for the seat-man configuration must be found. Probably the most reliable drag data available for existing conventional ejection seat systems is that obtained from the MIT Wind Tunnel Report #69, January 1954. Data from the MIT report was used for this study, as shown in Fig. 2, which is a plot of drag coefficient (C_D) as a function of Mach number for a seat angle of attack of 10 degrees. This angle of attack of 10 degrees represents the attitude of the seat just as it clears the rails during ejection and is exposed to the full free stream velocity.

2.3 Using the values of C_D given by Fig. 2, the magnitude of the g force as a function of calibrated airspeed, altitude and Mach number, for a conventional ejection seat having a frontal area normal to the airstream of 6.5 sq.ft. and an occupied seat weight of 325 pounds, is shown in Fig. 3. The most significant fact to consider here is that the maximum linear acceleration is practically constant for a given calibrated airspeed regardless of altitude. The approximation may be made, without introducing too serious an error, that the maximum linear acceleration caused by ejecting into an airstream is constant for any given calibrated airspeed. Referring to Fig. 3, this means that the maximum linear acceleration experienced at M=1 at sea level is about the same magnitude as is experienced at M=2 at 39,000 feet. However, for a given escape system, even though the maximum g forces are equal for constant calibrated airspeeds, two other important factors must be considered in order to determine the tolerable limit of escape as a function of calibrated airspeed at extreme altitudes:

2.3.1 First, for a given ejection seat system at a constant calibrated airspeed, the rate of tumbling is increased with an increase in altitude because of the lower damping forces experienced in the less dense atmosphere. With an increase in altitude, at constant calibrated airspeed, the aerodynamic damping force which tends to reduce tumbling decreases. However, the force which produces the tumbling remains constant. The rate of tumbling for a given escape system at 40,000 feet approaches a rate double that experienced at sea level for the same calibrated airspeed.

2.3.2 Secondly, as altitude increases for any given calibrated airspeed, the kinetic energy of the ejected occupied seat increases because the kinetic energy of the system is a function of the square of the true airspeed. The ratio of the increase in kinetic energy for a given calibrated airspeed at an altitude of 40,000 feet as compared to sea level conditions is approximately three to one. This three-fold increase in kinetic energy must be dissipated as a function of time in the less dense atmosphere at altitude. For this reason, it must be expected that the duration of the acceleration will increase with altitude. For a stable escape system, the relationship expressing g as a function of time may be derived from the basic force equation;

$$m\frac{dv}{dt} = -C_D A \frac{1}{2} \rho v^2 \qquad \text{(Eq. 2.1)}$$

Integrating this equation between proper limits, substituting, and solving for t reveals;

$$t = \frac{2W(v_0 - v)}{C_D A \rho g \, v_0 v} \qquad \text{(Eq. 2.2)}$$

where t is defined as the time required to go from v_0 to v and ρ would represent the density of the air at the altitude of ejection. Analog solutions of the above equation for the conventional ejection seat system used in the sample calculations for Fig. 3 were obtained between an initial and final calibrated airspeed at various altitudes to determine the curves for g versus time at various altitudes, shown in Fig. 4. An analysis of these results revealed that the decay of deceleration at various altitudes closely approximates the relationship

$$t_h \cong \frac{t_{SL}}{\sqrt{\rho/\rho_0}} \qquad \text{(Eq. 2.3)}$$

for a given g force, as shown in Fig. 4. For example, this simply means that it would take approximately twice as long for acceleration to decrease from 35 g to 10 g at 40,000 feet as it would at sea level. In this respect it is important to mention that duration of acceleration is considered just as important as magnitude when defining human tolerance to acceleration. The two factors cited above, tumbling rate and duration of acceleration, may be the limiting criteria for successful escape at extreme altitude rather than the more easily defined 35 g limit of linear deceleration described in Fig. 1.

2.4 Fig. 5 indicates the linear deceleration as a function of calibrated airspeed showing the effect of variation of the weight of the ejected occupied seat and the area normal to the airstream for values of C_D taken from Fig. 2. It can be seen from the equation defining

$$g = \frac{D}{W} = \frac{C_D A}{W}\left(\frac{1}{2}\rho v^2\right) \qquad \text{(Eq. 2.4)}$$

that the maximum linear deceleration is directly proportional to the area normal to the airstream and inversely proportional to the weight of the ejected occupied seat. For the conventional ejection seat in common use today, the weight of the occupied ejection seat will vary from 300 to 360 pounds; and estimates of the area normal to the airstream for various ejection seat systems vary from 5 to 7.5 sq.ft. For any given set of ejection circumstances, the ratio of A/W may be determined, and designated as k so that the equation for g may be written as

$$g = k C_o \left(\tfrac{1}{2}\rho v^2\right) \qquad \text{(Eq. 2.5)}$$

The maximum g force is directly proportional to k, whose values for a conventional ejection seat system vary from about .014 to .025. The curve of Fig. 5 represents the contributing factor of ($C_o \frac{1}{2}\rho v^2$) in describing the relationship of g versus calibrated airspeed. The effect of the variation of k on the magnitude of the g forces is shown by the vertical lines at the left of the graph. To minimize the maximum linear deceleration, it is desirable to have as high an ejection weight as possible confined in the smallest practical frontal area in order to reduce the value of k to its most practical minimum. The important point to consider here is that the limiting 35 g linear deceleration may be expected within the range of calibrated airspeeds from 575 to 725 knots depending on the variables of A and W. For a given escape system, it is desirable to increase the weight of the ejected occupied seat to the maximum permissible value compatible with catapult and structural limitations in order to reduce the maximum g forces. However, the reduction in maximum g obtained by reducing the ratio of k is made at the expense of increased duration of acceleration. Duration of g forces as a function of t is most easily explained by referring to Equation 2.2 which clearly shows that t is inversely proportional to the ratio A/W. Here again, it is apparent that duration rather than magnitude may become the limiting criteria for exceeding the human tolerance curve defined by Fig. 1.

2.5 Because of the number of variables involved, it is not possible to define a limitation of the ejection seat for successful escape in terms of a single value at calibrated airspeed. However, it is important to note that the maximum linear deceleration associated with conventional ejection seat systems increases sharply to significant magnitudes, relative to human tolerance, as the calibrated airspeed is increased above 550 knots. In order to indicate the seriousness of the problem, the rate of increase of the g forces as a function of calibrated airspeed may be determined as shown in Fig. 6. Here again it is assumed that the maximum force is constant for any given calibrated airspeed, and the computed g forces described in Fig. 3 are replotted as a function of speed on log-log paper. The result is a straight line with a slope of 2.47 to 1. From Fig. 6, showing the maximum g forces as the ordinate and the calibrated airspeed as the abscissa, an equation for g may be written as

$$g = K\left(\frac{v}{100}\right)^{2.47} \qquad \text{(Eq. 2.6)}$$

where K is a constant for a given set of escape system conditions. This simply indicates that g increases as the 2.47 power of the velocity. The slope of this line is a function of the product of C_D times velocity squared. Because the nature of the variation of C_D with Mach number, as shown in Fig. 2, will be much the same for most of the conventional ejection seat systems, the slope of this line will always be near the value 2.47 to 1. In the Equation 2.6 the constant, K, is a function of the product of the ratio of A/W and the relative magnitude of the values for C_D. Changing the value of K simply shifts the curve to the right or to the left on this graph. The actual value of K in the Equation 2.6 would vary slightly for different ejection seat systems depending on the relative magnitude of C_D, which in turn depends on the relative aerodynamic cleanness of the ejection seat system. It is postulated that if the relationship of g versus v were plotted on Fig. 6 for most of the existing ejection seat systems, the actual computed values of g would all be in a relatively narrow band presenting a slope very near the value of 2.47 to 1. The important fact to consider here is that the slope of this line is such that a 15% increase in calibrated airspeed results in a corresponding increase in g forces of almost 50%. From this analysis it is reasonable to expect that the maximum linear decelerations experienced during ejections with existing ejection seat systems rapidly approach the limit of human tolerance (as defined in Fig. 1) as the speed of the aircraft at time of ejection is increased above 550 knots indicated airspeed.

2.6 A further indication of the seriousness of the problem may be realized by considering the maximum speed capability of the "Century Series" of fighter aircraft. The straight and level speed capability at maximum power is considerably less than the limiting permissible red-lined diving speed, and would represent a conservative estimate of performance capability. From the brief analytical approach described in this report, it seems quite obvious that the capability of the ejection seat as a means of successful escape is questionable in the "Century Series" fighters, which have a speed capability in excess of 600 knots calibrated airspeed.

2.7 Investigations aimed at solving the problem presented above include such things as the following:

2.7.1 Drogue chutes to slow the aircraft down to an acceptable escape speed.

2.7.2 Thrust augmentation of the ejection seats in the direction of flight to reduce the initial deceleration at the expense of increased ram-air pressures.

2.7.3 Rocket thrust to increase the ejection velocity to insure tail clearance.

2.7.4 Increase ejection seat weight to reduce initial deceleration at the expense of increased duration of g forces.

2.7.5 Stabilization techniques for ejection seats.

A preliminary review of proposed solutions indicates there is no "quick-fix" solution apparent for the existing ejection seat systems. It is expected that optimization of some of the proposals resulting in a judicious compromise of decreased g forces at the expense of increased duration will allow a slight penetration into the present unsafe speed range with the current operational escape systems. The aerodynamic characteristics of a blunt object such as the ejection seat system will dictate a relatively high value for ($C_D\ A/W$). Therefore, this system will not lend itself for serious penetration into the extreme speed range of the "Century Series" fighters because of both magnitude and duration of g forces. However, an aerodynamically clean configuration as presented by an ogive type capsule would reduce the relative magnitude of C_D by a factor of as much as 4 or 5. The value of ($C_D\ A/W$) for the capsule would be such as to reduce the maximum linear deceleration by sufficient magnitude so that the duration of g forces would be insignificant relative to human tolerance. The aerodynamically clean capsule type of escape system could extend into and beyond the extreme speed range capability of the "Century Series" fighters and still not trespass human tolerance either in magnitude or duration of g forces.

3. SUMMARY

3.1 The information presented by this study may be summarized as follows for the conventional ejection seat system:

3.1.1 The maximum linear deceleration is essentially constant for a given calibrated airspeed regardless of altitude.

3.1.2 At constant calibrated airspeed the rate of tumbling increases with altitude and approaches a value proportional to the inverse of the square root of the density ratio.

3.1.3 At constant calibrated airspeed the duration of g forces is approximately proportional to the inverse of the square roo the density ratio.

3.1.4 The maximum linear deceleration forces increase as the 2.47 power of the velocity.

3.1.5 The maximum linear deceleration rapidly approaches the limit of human tolerance as the speed of the aircraft at time of ejection is increased above 550 knots calibrated airspeed.

3.2 The aerodynamic and physical characteristics defined by the parameter ($C_D\, A/W$) are such as to limit the usefulness of the conventional ejection seat system to the lower part of the speed range of the "Century Series" fighter. Only by optimization of these parameters, such as may be obtained by the use of a low drag capsule, can successful escape be expected in the extreme speed range capability of the "Century Series" aircraft and beyond.

Publication Review

This report has been reviewed and is approved.

Frederick Harris
FREDERICK HARRIS
Chief, Plans Office
Directorate of Research

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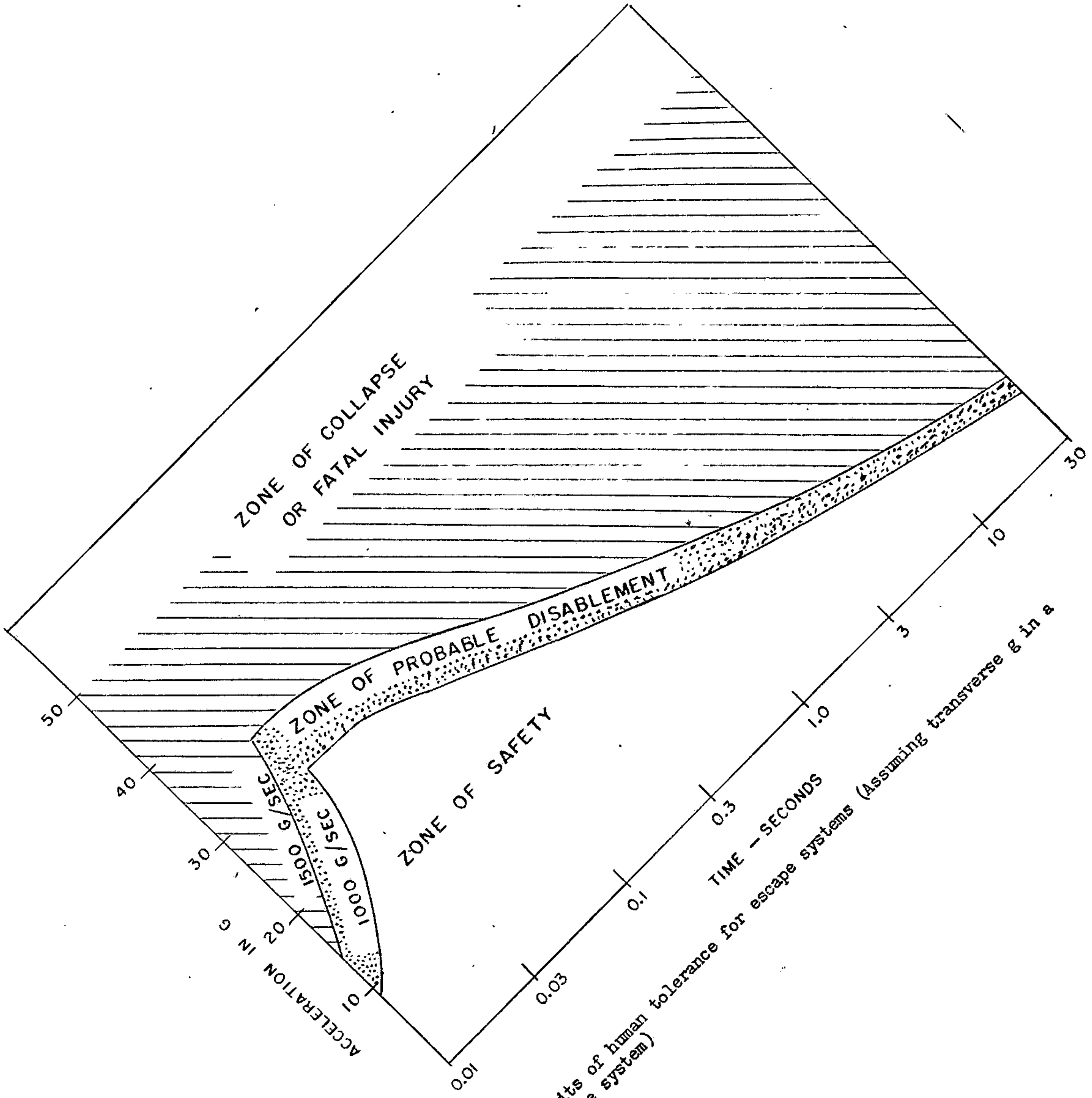


Fig. 1. Limits of human tolerance for escape systems (Assuming transverse g in a stable system)

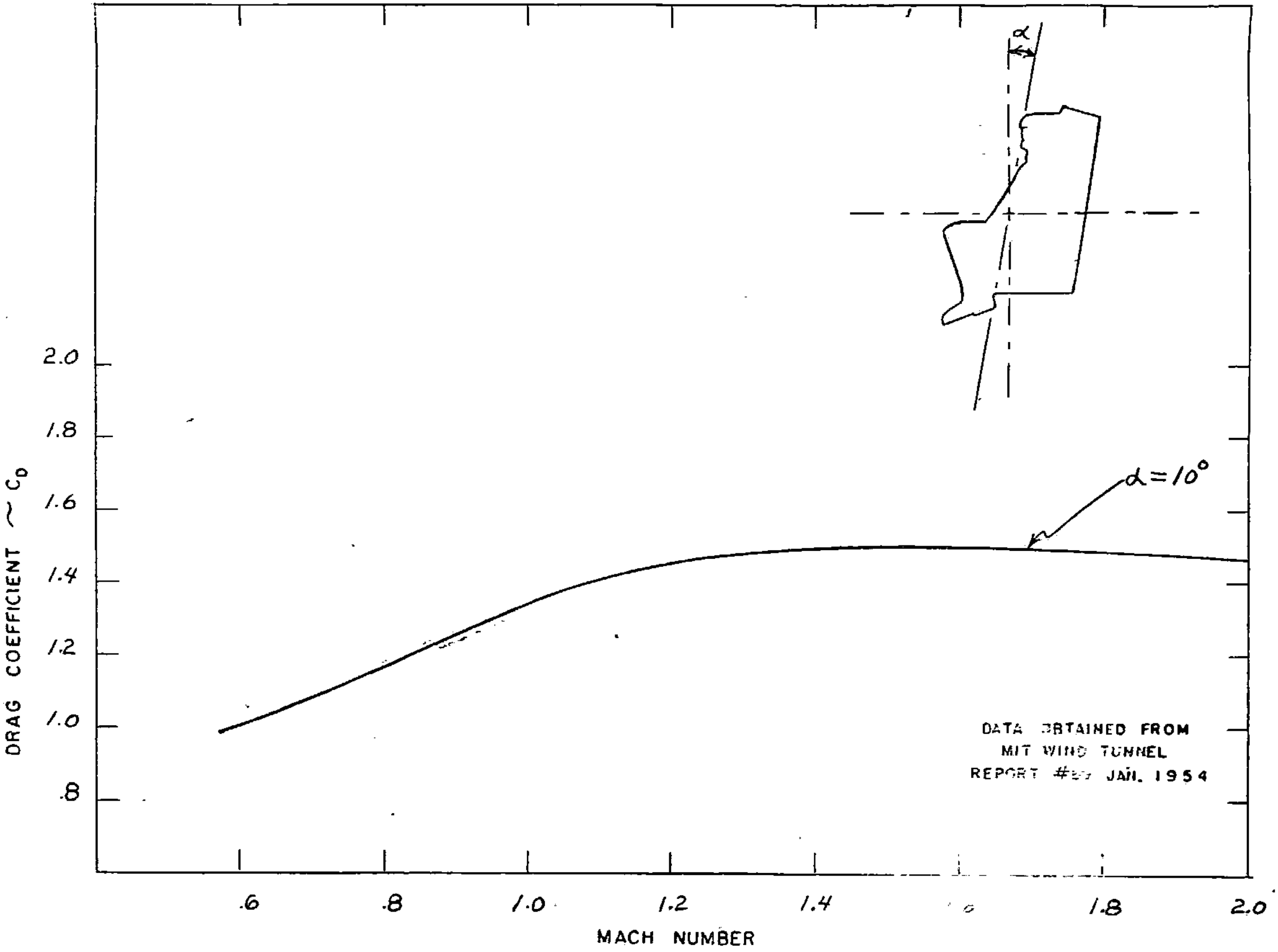


Fig. 2. Drag characteristics for conventional man-seat ejection system

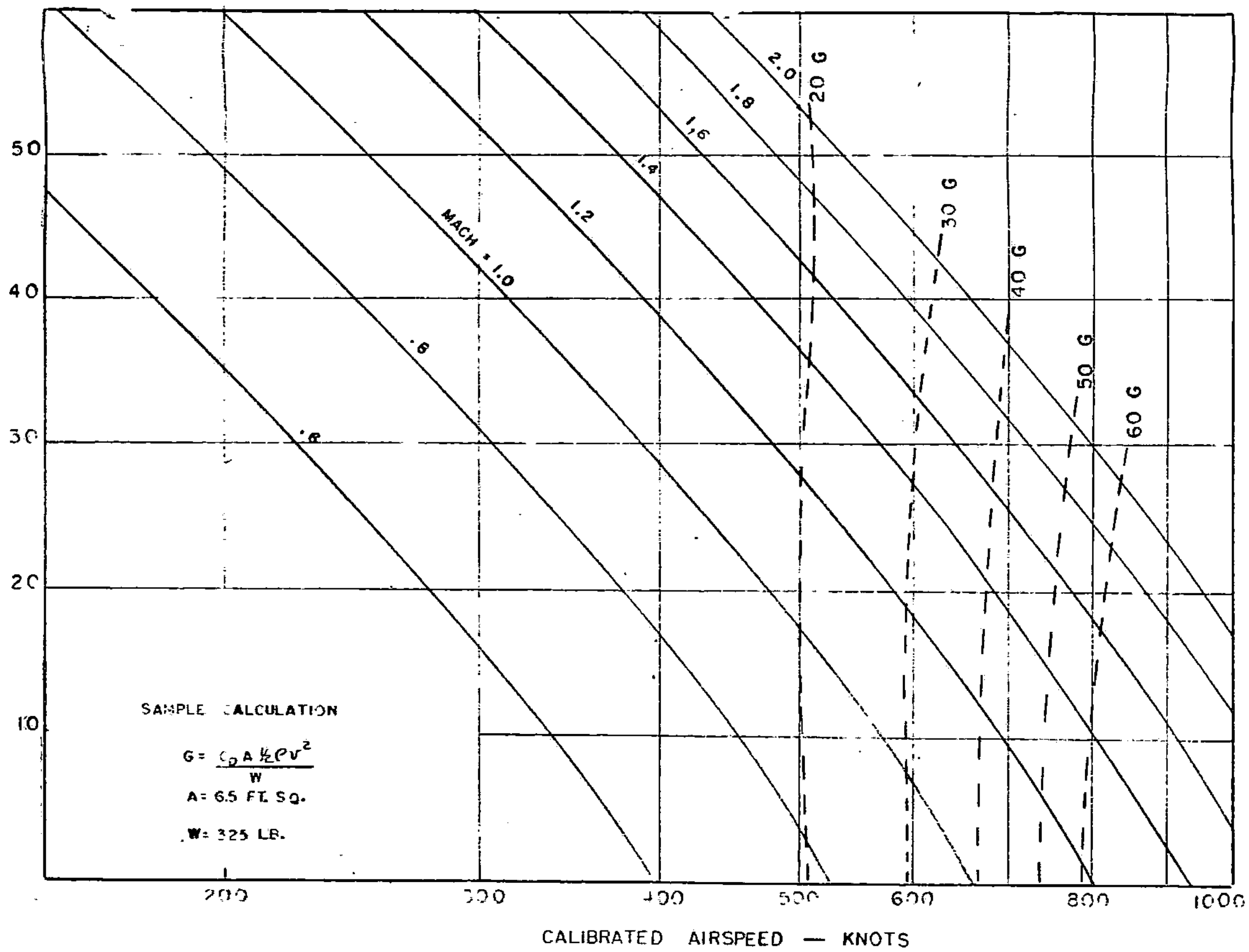


Fig. 3. Magnitude of g force as a function of calibrated airspeed, altitude, and Mach number.

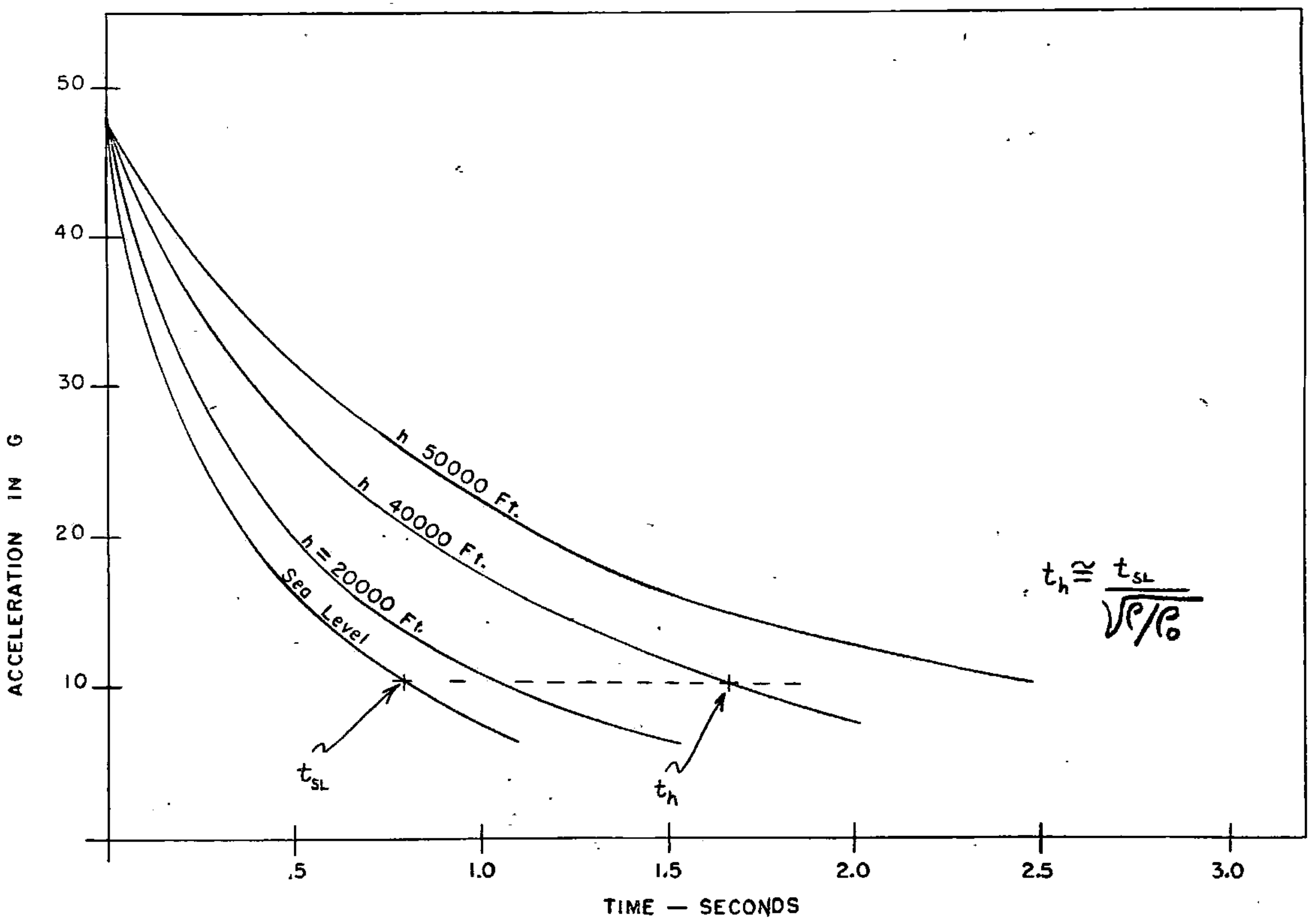


Fig. 4. Decay of deceleration as a function of altitude.

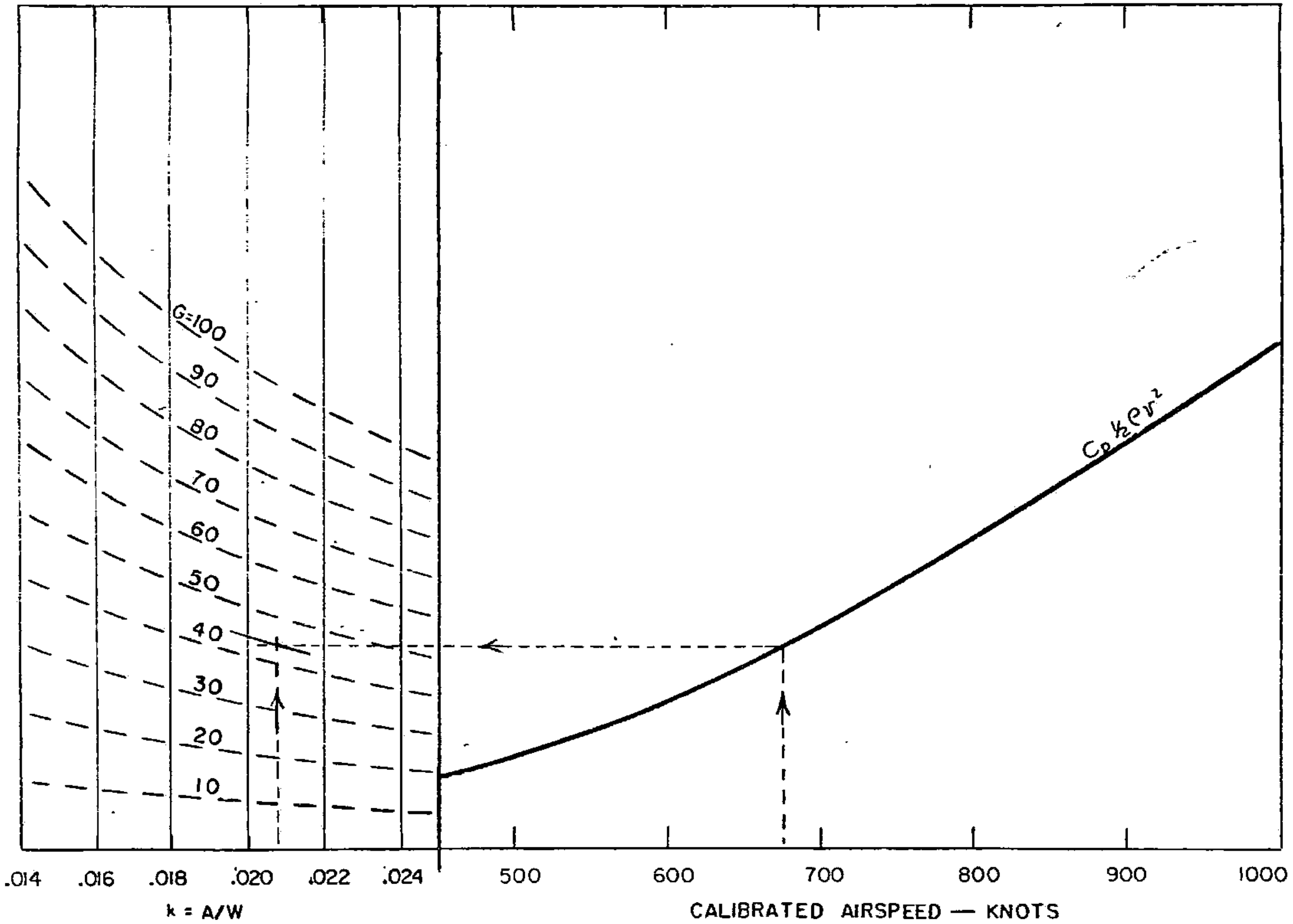


Fig. 5. Linear deceleration as a function of airspeed showing effect of variation of k = A/W.

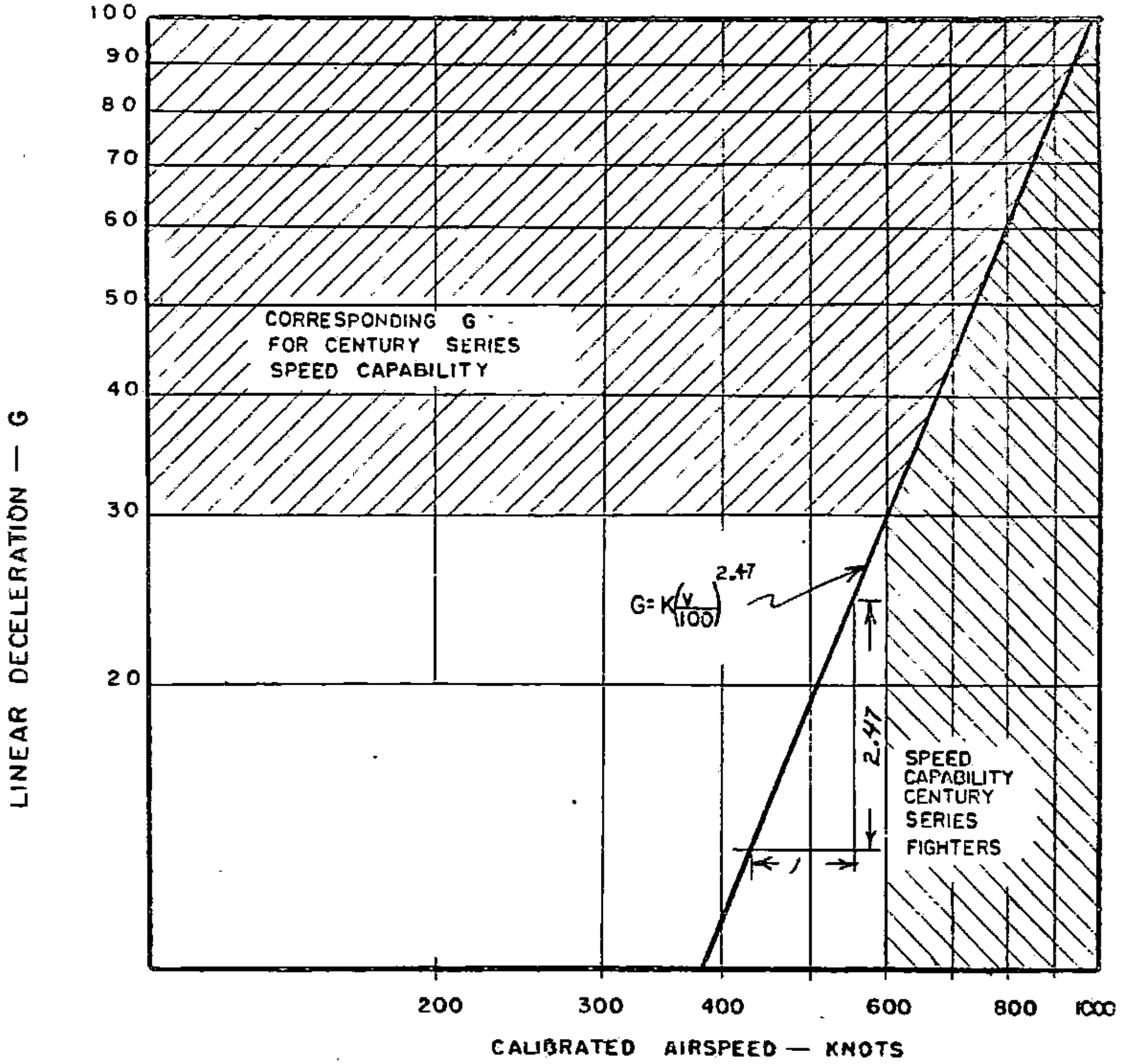
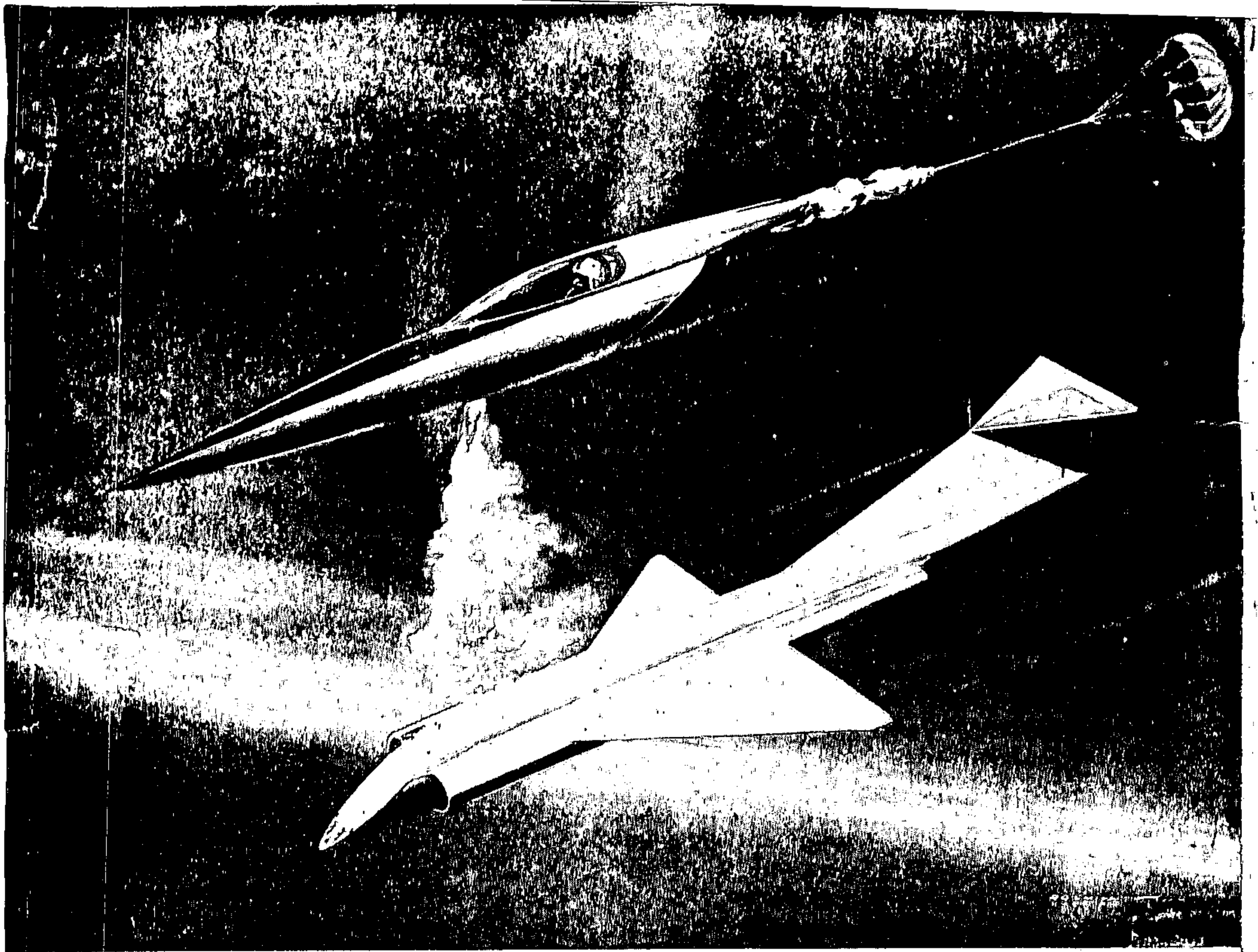


Fig. 6. Linear deceleration as a function of calibrated airspeed.



A configuration for high altitude escape capsule.

ESCAPE SYSTEMS

Because absolute safety is unattainable, moral, morale and economic considerations must be balanced.

By A. M. Mayo

Douglas Aircraft Co.

To be objective, escape system selection criteria should be based on over-all requirements of the man-airplane combination. In addition to the fundamental objectives of aircraft or weapons system being considered, three general considerations appear to be fundamental with respect to escape. These are: (1) Moral Considerations: The basis by which the value of one person's life is compared to another's consistent with American philosophy; (2) Morale Considerations: The basis for estimating the military value of escape equipment with respect to its effect on the will of the individual or group to perform under hazardous conditions; and (3) Economic Considerations:

The over-all cost in man lifetimes of effort to maintain a particular system or device can be objectively estimated by the use of the growth factor concept. This value, balanced by moral and morale considerations, can then be compared on a statistical basis to a probable dividend in lives saved. It can thus be seen that it is not necessary to be completely arbitrary or to rely entirely on expert opinion in making decisions as to what proportion of the effort on a given weapon system should be applied to the emergency escape system.

Examination of statistical evidence indicates that absolute safety is only an approachable objective and is not attained even in the more protected type of everyday living. It should then be recognized that maximum effort in the escape system field should be applied to solving those problems giving the maximum statistical dividends. This course of action is, however, much more easily stated than accomplished because of the fact that the unusual and extreme condition accidents receive a far disproportionate amount of publicity, and thus provide a terrific temptation for the engineer to also apply excessive effort in these directions, some-times to the detriment of total safety. The careful application of available statistical information such as can be obtained from the U. S. Air Force Directorate of Flight Safety and the Naval Aviation Safety Activity can be particularly helpful in arriving at a proper balance.

Environment

In the selection of escape equipment, environmental problems associated first with the actual escape and then with the climatic conditions to which the escapee is projected need consideration. The problem areas may be considered as follows: Pressure and Composition:

In the more common types of escape equipment in which the man leaves the environment of his cockpit, a wide variety of pressure considerations immediately become evident. The first of these has to do with the problems associated with hypoxia, aero-embolism, explosive decompression and such associated changes caused by the loss of cockpit pressure. For escape above 30,000 feet, the problems begin to arise which require the definite use of bail-out oxygen. For escape above 45,000 feet, emergency pressure protection together with explosive decompression protection also becomes increasingly important. In the event of fire or gunfire damage, the added problem of the protection against flames and toxic gases should not be completely overlooked.

Temperature

While the problem of temperature control at low speeds can usually be relatively easily handled by proper clothing to prevent frostbite, at higher speeds the man must be given much more effective protection if he is not to be incinerated in the same manner as that of the meteor coming into the atmosphere.

With flight operations becoming worldwide and single flights spanning both tropical jungles and artic waters, the problem of survival after landing looms as one requiring added emphasis if a highly refined escape system is to save the man in the air only to have him perish from temperature extremes after landing.

The increasing availability of good data with respect to man's tolerances to acceleration and blast forces as a function of time, together with better aerodynamic and physical data, is bringing the problem of force control more nearly into the realm of objective analysis. Figure 1 indicates probable safe escape speeds versus altitudes for a typical light-weight ejection seat or escape chute system. This curve is based on calculations involving acceleration and time figures taken from the works of Lombard and Stapp.

These data are also useful in pointing out areas where it does not appear productive to expend great effort on the basis of known approaches. Such an example is illussrated by Figure 2. In this case certain combinations of altitude and velocities toward the ground can result in so much directed energy that the man's total tolerance to acceleration is exceeded before he can be slowed to a safe speed. Emphasis on safety in these areas should be directed toward preventing the man from getting himself into such a situation as it can be seen that these limitations are entirely independent of the type of equipment used to slow the man down.

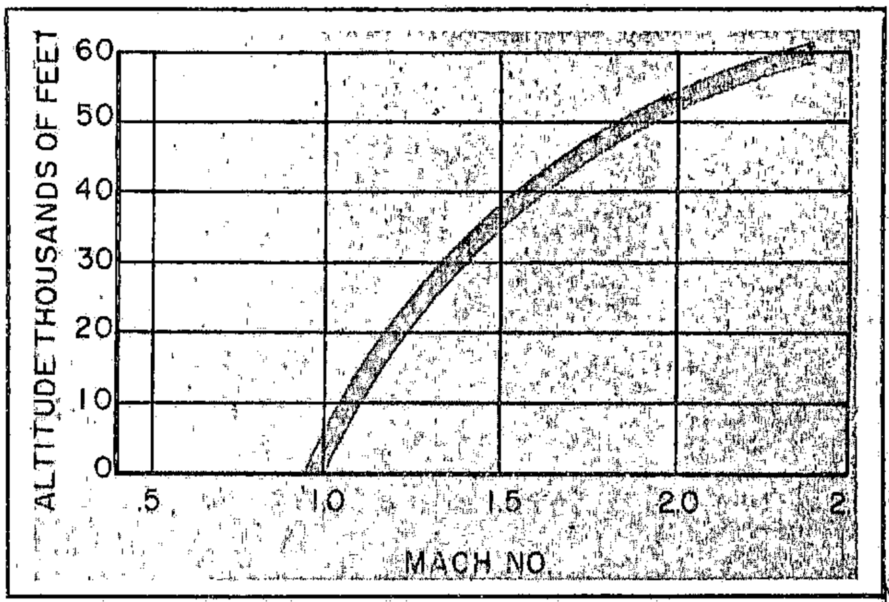


Figure 1.
Maximum speed for safe seat ejection based on probable deceleration tolerance.

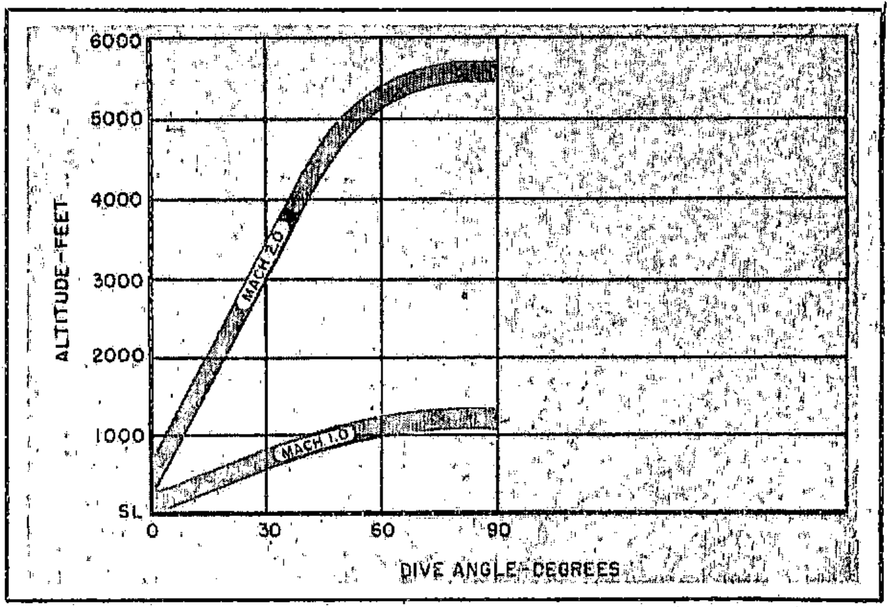


Figure 2.
Minimum escape altitude vs dive angle based on probable maximum deceleration tolerance.

Selection By Requirement

Even though the properly designed escape system must take care of all of the man's environmental and force limitations, it should be emphasized that the over-all objectives of the man-machine combination are still paramount. Large factors of safety on human tolerances can just as thoroughly reduce the over-all efficiency of the man-machine combination by adding unnecessary weight as can poorly designed and over-weight structural components.

Statistical evidence showing the percentage of ground crashes relative to mid-air escapes indicates the need of first order emphasis to that portion of the escape system providing protection against the forces, fire, water or untenable environment

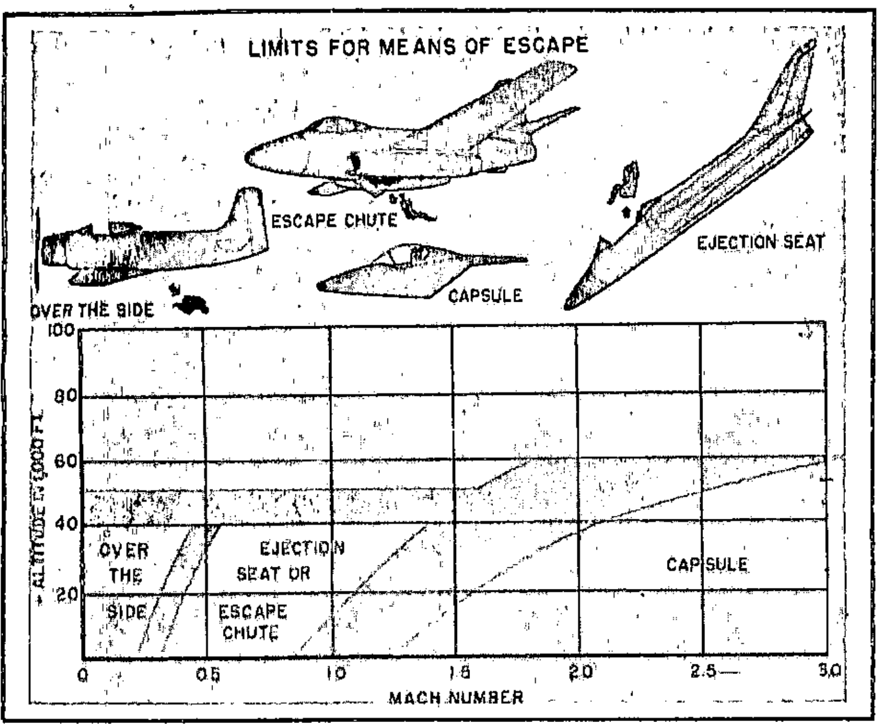


Figure 3.
Data for selection of an escape system.

associated with these ground crashes.

Figure 3 shows probable ranges and areas of overlap of the various general types of escape systems. The broad shaded areas indicate porbable variations in design approach and differences in interpretation of human tolerances data. The environmental problems associated with extreme altitude together with the temperature and force problems of the higher Mach numbers indicate that operational conditions are being approached which are clearly beyond the capabilities of the ejection seat.

It is in the border areas where substantial conflict of opinion as to the best approach is likely to exist. If the drag weight ratio is used as a primary criteria of air drag receleration exposure, it appears evident for the ejection seat that major reductions in these decelerations are not likely from better streamlining. This then leaves the addition of weight as the one remaining important method of substantially reducing seat drag decelerations. Additional weight is also added if stabilizing means are used to reduce tumbling.

The difficulty of utilizing equipment already required in the aircraft to take care of any large percentage of the weight needed to reduce G forces in the ejection seat becomes immediately apparent when space limitations are considered. In the case of the cockpit capsule, however, much greater flexibility exists both as to streamlining and the amount of useful equipment capable of being included in the escape section. While the weight of a cockpit capsule system increases directly as the total ejected mass, present studies indicate that its over-all weight penalty may be less than that of an ejection seat system, with satisfactory drag-weight characteristics, for sea level speeds that in some cases may approach values as low as Mach 1. The additional environmental and water survival capabilities inherent in the cockpit capsule could well swing the pendulum in its favor in some borderline cases.

One configuration of cockpit capsule which could meet escape requirements over substantial speed range at a relatively low weight penalty is illustrated.

In order to prevent excessive performance or economic penalty, the selection of the escape system should be based on an objective study of the over-all requirements of the man-aircraft into which it is to be incorporated. (Figure 3). Full utilization should be made of all available aeromedical, engineering, and statistical data available. The designer should be extremely wary of any system adding substantial increments of complexity, dead weight, or size increments to the over-all aircraft system.

BRITISH REGRET JET CANCELLATION

"I think this . . . is a national decision we shall regret for many years," George Edwards, Managing Director of Vickers-Armstrongs (Aircraft) Ltd., said commenting on the Ministry of Supply's announcement that orders for the V.1000 transport have been cancelled.

With a passenger capacity of more than 100 and capable of flying the Atlantic non-stop in both directions, a civil variant of the turbojet V.1000 was to be Britain's most outstanding candidate for the Blue Riband route in 1960. "With 100 passengers," he went on "the Americans in our view are unlikely to have engines capable of the non-stop westbound Atlantic crossing before that date.

"The decision to cancel the V.1000 means we have abandoned to the Americans, without a struggle, this highly important market. How important this market is can be seen from the fact that the orders so far placed by a few American operators for big jets total more than £200 millions. The total size of the market could be of the order of £1000 millions."

The Ministry explained that "during development, the weight of the V.1000 had grown substantially and had been outstripping the engine capacity. It had taken longer than was expected to solve the problems of R.A.F.

However, Mr. Edwards states that the increase in structure weight had in fact been fully matched by increases in engine power, and that the V.1000 could have been in all ways competitive with the two American jet airliners (D.C.-8 and 707), "and in some ways better."

The engine concerned is the Rolls-Royce Conway by-pass turbojet which combines power with great economy. It is, at present, the only large by-pass engine that has been made public, and is the second most powerful type-tested turbojet yet announced.

MILLER—ARS AWARD

Fred S. Miller, for sixteen years a key figure in the American rocket industry, is this year's recipient of the American Rocket Society's C. N. Hickman Award. The prize, a medal and citation, is presented annually for outstanding work in the field of solid propellants. Miller, Assistant Works Manager of the Aerojet-General Corporation Solid Rocket Plant at Sacramento, Calif., originally suggested the use of asphalt as a rocket fuel. This culminated in development of the first JATO (jet-assisted takeoff) units placed in large-scale production in the United States.

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CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-520
DATE March 28, 1958
MODEL Adv. F-106

TITLE

ADVANCED F-106 (u)
SUMMARY REPORT
OF
DESIGN STUDIES

SECURITY NOTICE
This document contains information affecting the national defense of the United States [illegible] Laws, Title 18 U.S.C., Sections 793 and 794, the [illegible] or the revelation of its contents in any manner to an unauthorized person is prohibited by law.

PREPARED BY F. L. Tripp

GROUP Preliminary Design

GR. 4
DOWNGRADED AT 3 YEAR INTERVALS; DECLASSIFIED AFTER 12 YEARS.
DOD DIR 5200.10

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
Chief, Pre-Design

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INTRODUCTION

Under contract AF 33(600)-30169, Supplemental Agreement 14, Convair has undertaken study of Design Features pertinent to an Advanced F-106 Interceptor. This study contract became effective 31 January 1957, with a contractual completion date of July 1958.

One requirement of the contract is a final summary report. The contractor has chosen to fulfill this requirement with a series of four reports, dealing with each of the major aspects of the study:

		Report
A.	Auxiliary Control Surface and Bi-Convex Wing	ZA-8-506
B.	Pre-Compressor Cooling Propulsion System and Related Studies	T-106Adv.-32
C.	Pilots Escape Capsule	ZP-8-519
D.	Preliminary Design Studies of Application of Features to a Tactical Configuration	(This Report)

These reports summarize and coorelate material contained in many other reports, which are not contractually required, but have been submitted for information (Listed in Appendix B). An outline of this report structure is shown by Figure I.

This report will discuss application of each study feature to a tactical aircraft and briefly describe the basis of synthesis of the proposed F-106-30, which has resulted, in part, from this study.

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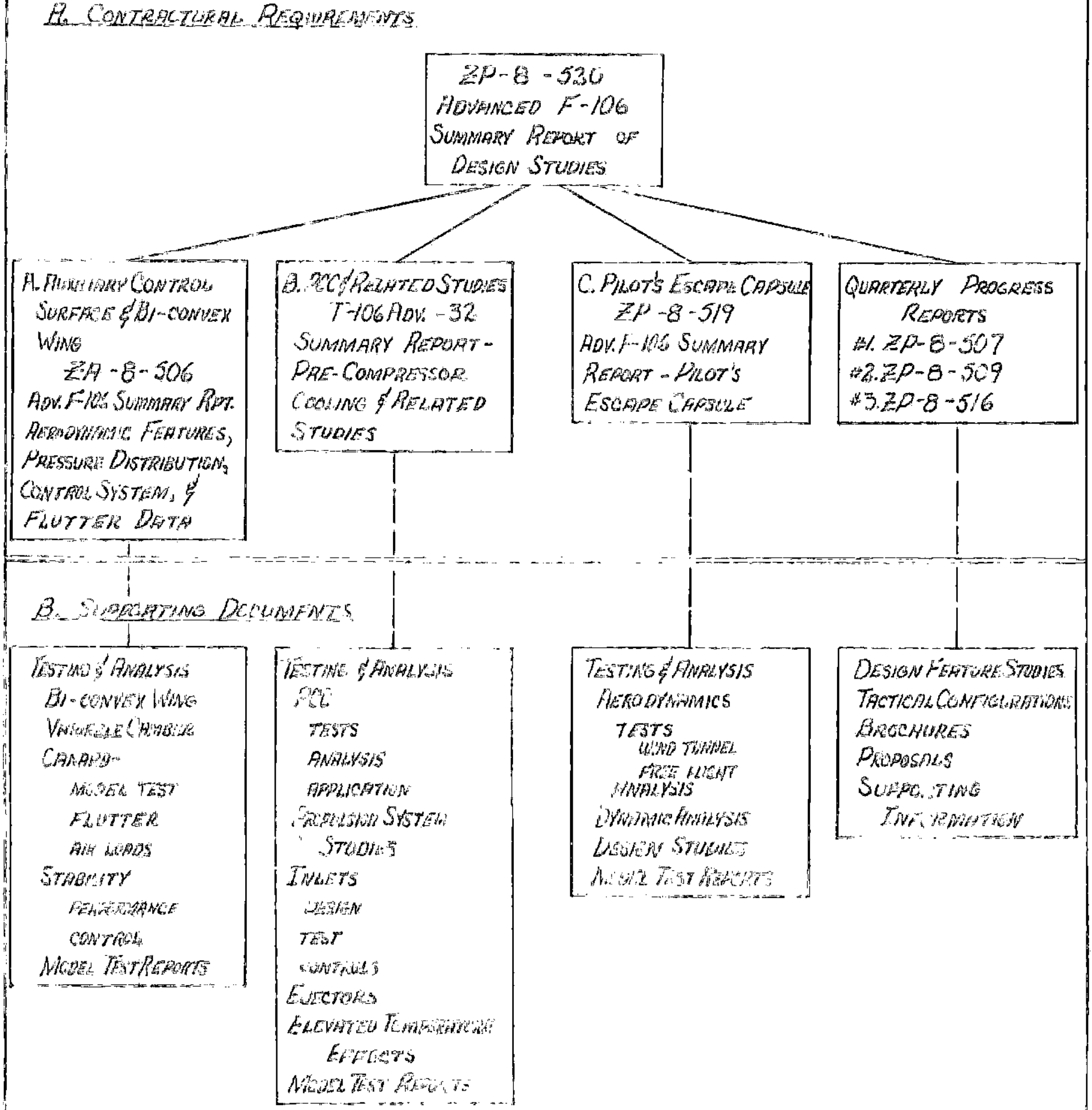
FIGURE I

ADVANCED F-106

REPORT STRUCTURE



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SUMMARY

Preliminary Design studies of each study feature have been made. Each can feasibly be incorporated in a tactical interceptor aircraft. Hardware development programs of varying time and cost would be required to achieve this end.

Development of a tactical interceptor is based upon many factors, such as:

1. The threat in the operational life.
2. Engine, Fire Control and Armament availability.
3. Funding available for defense.
4. Engineering and production state-of-the-art.

As a result, Convair has developed an F-106 type interceptor that utilizes those components of the study that will provide greatly improved performance with a minimum development program. The features included in this airplane, designated the F-106-30, are:

1. Auxiliary longitudinal control (canard surface).
2. Bi-Convex airfoil.
3. Variable camber wing.
4. Ventral (plus) vertical tail.
5. Internal Compression Inlets.
6. By-pass air ejector.
7. "Industry Crew Escape System".

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DISCUSSION

I. Initial Study Configuration

A. Advanced F-106

The configuration of the Advanced F-106 at the initiation of the study is described in Convair Report ZP-57-08000, February 1957, and shown by the three-view and inboard, Convair Drawings SD-56-08075 and SD-56-08076 respectively.

Design features of this configuration included canard control surfaces equipped with flaps to increase low speed lift capabilities, elevons acting as flaperons, variable camber wing leading edge, bi-convex wing, crew escape capsule, internal compression inlets, pre-compressor cooling and variable ejector. The fire control system designated was the Advanced MA-1 with a 40" radar antenna.

The power plant for this aircraft was the JT4B-22 with a sea level rated thrust of 26,500#. Normal take-off gross weight of 44,555# included 10,645# of fuel and 2,870# of water for thrust augmentation by means of pre-compressor cooling.

With this arrangement, the performance envelope (Figure 1) at M = 2.5 extends from 42,500 feet to 75,000 feet with afterburner power. At M = .95 and afterburner power, the 1g ceiling is 53,800 feet. Climb performance from brake release to 40,000 feet and M = .95 is 3.8 minutes, to 50,000 feet at M = .95 is 8.5 minutes, and 60,000 feet at M = 2.3 is 8.3 minutes. Take-off over 50 foot obstacle requires 4330 feet and landing over a 50 foot obstacle requires 4160 feet.

A major portion of the high altitude performance accrues from the effects of pre-compressor cooling. The gains from this augmentation consist of approximately 20,000 feet of altitude at M = 2.5 reducing to a zero gain at M = 1.2. (See Figure 21.1)

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Also contributing to the performance are: internal compression inlets which yield a higher pressure recovery at high speeds and lower cowl and bypass drag than external inlets at the same design speed; an ejector, tailored to provide near optimum propulsive efficiency at supersonic speeds.

Other aerodynamic features include the variable leading edge, with which optimum wing camber can be approached for any flight condition of speed and altitude, thereby, decreasing the drag compared to a wing compromised to minimize the drag at supersonic speeds only and the bi-convex airfoil which reduces the zero lift drag in comparison with conventional airfoil sections at high supersonic speeds.

Auxiliary control surfaces, in this application, a canard, add to the maneuverability, reduce the trim drag at supersonic speeds and improve take-off performance.

These features of the Advanced F-106 were more fully investigated under studies designated as:

A. Auxiliary Control Surface and Bi-convex Wing.

B. Pre-Compressor Cooling and Related Studies.

In addition, the requirements for safe escape from an airplane at high speed and altitude as well as escape during landing or take-off as a result of malfunction was considered in another portion of the study:

C. Pilot Escape Capsule.

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B. Advanced F-106 - Extended Fuselage

Analytical studies, underway in the early phases of this study, were made to determine the effects on airplane drag, high altitude maneuverability, and take-off performance, of variations in geometry and longitudinal stability of canard arrangements of the Advanced F-106 airplane. The parameters studied are canard arm, canard size, longitudinal stability, Mach number and flight lift coefficient.

The configuration investigated included the Advanced F-106 geometry and two modifications with the fuselage length increased by 40" and by 80". This resulted in a nose length increase of 25" and 50" respectively. In terms of canard geometry, the canard arms become 1.26 and 1.35 $\bar{c}$ compared to 1.17 $\bar{c}$ for the unmodified configuration. Canard volume (canard area x canard arm) was held constant, hence, no change in take-off or maneuverability characteristics would be expected. However, an increase of 685 pounds is incurred for additional fuselage structure for the 50" extended nose.

Zero lift drag effects of canard geometry are shown in Figure 2. In the subsonic case, an increase in canard arm leads to an increase of drag through fuselage skin friction while a reduction of canard volume for a given canard arm leads to a decrease in drag through canard skin friction.

In the supersonic case, drag reductions are obtained as in the subsonic case as canard area decreases, and increases are obtained in fuselage skin friction as the canard arm is extended. However, at canard arms up to 1.4 $\bar{c}$, skin friction increases are more than offset by the decrease in fuselage wave drag obtained through increased fineness ratios.

The combined effects of the drag and weight changes on the 50" extended nose compared to the Advanced F-106 are:

a. 3-1/2% reduction in subsonic specific range.

b. 200 foot decrease in Mach 2.0 ceiling.

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c) 1/2% increase in acceleration rate at 35,000 ft.

While these effects are small and indicate that the nose length change would probably not be warranted on the basis of performance, certain other considerations lead to the desirability of an increase in fuselage length.

One of these factors in the Advanced F-106 was the effect of the canard on the inlet performance. This configuration allowed the super-velocity field above the deflected canard surface to impinge on the inlet face at high Mach numbers. A fifty-inch increase in the canard-inlet separation assures that the trailing shock will lie above the inlet at all Mach numbers and that the wake shed by the canard surface will pass below the inlet.[1] An increase in fuselage length will also provide increased fuel capacity and permit an improved electronics gear distribution.

Because of these factors, the Advanced F-106 configuration was modified to provide a 50" nose extension, or overall length increase of 80". Studies of this extended nose version are shown in drawing CV-SD-57-08028.

Subsequent testing and evaluation was based on this 50" extended nose version of the Advanced F-106.

[1] Further considerations of the canard-inlet geometry are discussed in Section II. B.

[redacted]

II. Studies as Related to Application in a Tactical Airplane

During the course of the studies on the design features (Variable camber; bi-convex wing; longitudinal control surfaces, etc.), these items were continuously integrated into the preliminary design of a tactical airplane. This was necessary to provide an insight into the overall effects of the inclusion of the design feature and to insure that these features did not yield adverse effects when combined into a unit design.

In this section the performance effects and the installation problems are presented for the application of these design features into a tactical vehicle. The selection of the design features for the final recommended tactical aircraft will be carried out in a latter section (Tactical Version Resulting from These Studies).

A. Variable Camber; Bi-Convex Wing

The small low aspect ratio wings of high performance airplanes have high drag due to lift. These wings can be provided with conical camber to provide minimum drag due to lift in a region near the design lift coefficient. However, the camber increases drag at lower than design lift coefficients. Also, the camber benefit of drag due to lift disappears as the velocity component normal to the leading edge becomes sonic.

With variable camber, the camber can be matched with flight conditions over a wide range of speeds and altitudes and can eliminate camber minimum drag penalties at high Mach numbers. Sharpening of the leading edges by the use of a biconvex section reduces wing wave drag at speeds where the leading edge is supersonic.

Figure 3 shows the drag due to lift for the Advanced F-106 Airplane for 4 different wing cambers at Mach numbers of 0.8, 0.9, 1.0, 1.5, 2.0, and 2.5. The wings shown are at (Mach = 1.0 design) C_L design of: NACA Section, $C_L = 0.1$; NACA Section, $C_L = 0.3$; biconvex section, $C_L = 0.1$; and biconvex section, $C_L = 0.3$. The drag due to lift data shown are all referenced to the biconvex section,

[redacted]

C_{L_D} (C_L design) = 0.1 untrimmed drag at zero lift[1] so the plots give a true comparison of the different wings.

The drag polars are marked to show the region in which the airplane will be operated during acceleration, at ceiling, and during subsonic cruise. The region between acceleration and ceiling will be involved during climb and supersonic cruise. Subsonic loiter will occur at a C_L slightly greater than 0.2 at Mach numbers of 0.8 and lower. The lower portions of the curves will affect low altitude acceleration and climb performance at subsonic speeds.

The benefits of variable camber are apparent on comparison of the biconvex $C_{L_D} = 0.1$ and 0.3 drag polars. Subsonically, the $C_{L_D} = 0.3$ camber has better acceleration and ceiling performance due to the reduced drag compared to the $C_{L_D} = 0.1$ camber. At Mach 1.0, there is still a ceiling benefit for the $C_{L_D} = 0.3$ camber, but the acceleration lift coefficients have decreased into the region where the high zero lift drag of the $C_{L_D} = 0.3$ camber causes a reduction in acceleration performance.

In the supersonic regime, the $C_{L_D} = 0.1$ camber wing has performance at least equal to the $C_{L_D} = 0.3$ camber wing and at low values of lift coefficient it has a substantial reduction in drag relative to the $C_{L_D} = 0.3$ camber wing. As the Mach number is increased, the $C_{L_D} = 0.3$ camber wing suffers a further increase in drag due to lift with respect to the $C_{L_D} = 0.1$ camber wing and also retains its zero lift drag penalty.

Since the airplane must perform its mission in several different flight regions, any fixed camber would seriously compromise the performance in some of these regions. Approximate increments of performance between the $C_{L_D} = 0.1$ and 0.3 camber wings are shown in Table Ia. The performance increments shown are for a biconvex

---

[1] Drag effects of section and camber are referenced to the biconvex $C_{L_D} = 0.1$ wing as follows:

$$C_D = C_{D_i} + C_{D_{Trim}} + (C_{D_o} - C_{D_{o_{Biconvex}\ C_{L_D} = 0.1}})$$

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wing with a fixed $C_{L_D} = 0.3$ camber relative to a $C_{L_D} = 0.1$ camber. With a variable camber, both the subsonic and supersonic factors show positive gains.

The effects of the biconvex section may be seen from a similar analysis as that for the variable camber. For the $C_{L_D} = 0.1$ camber wings, the zero lift drag of the biconvex section is about the same as that of the NACA section at subsonic speeds, but at supersonic speeds the zero lift drag is significantly less for the biconvex section. The biconvex section has higher drag due to lift than the NACA section at subsonic speeds, and the same drag due to lift as the NACA section at supersonic speeds. Thus, the biconvex wing has superior supersonic performance benefits while the NACA section has a slight advantage at high C_L's at subsonic speeds.

Approximate increments of performance between the biconvex section and the NACA section, both with variable camber, are shown in Table Ib. The biconvex section is superior to the NACA section except for some small decreases in performance in the low speed-high altitude region.

The mechanization of the variable camber wing introduces essentially the same problems as those associated with movable control surfaces. The differences lie primarily in the fact that the required rate of motion is considerably less and that the juncture of the movable portion of the wing with the fixed portion be adequately sealed and smoothly faired.

A layout of the structure for the variable camber leading edge is shown by Convair Drawing SD-56-08064. The variable camber section occupies the leading edge outboard 15% of the semi-span. The construction consists of honeycomb filled aluminum skins with a spar extending along the hinge line and local supporting ribs. Methods are shown for providing a sealed joint and stabilized fairing at the hinge line as well as well as for the interconnections and seals necessary to accommodate wing deflections by a series of panels along the leading edge.

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The variable camber leading edge is positioned by an irreversible screw jack, hydraulic drive system as shown in Convair Drawing SD-56-08065. The primary screw drive shaft, driven by a hydraulic motor, drives the unit drive gear boxes providing a positive measure of deflection to the variable leading edge.

The servo positioning system, shown with a manual override in the event of system malfunction, would be the optimum method of programming the deflection of the variable camber leading edge. With this system, near optimum wing camber can be obtained at all conditions of speed and altitude.

However, a system of lesser complexity can be used and still retain a very high percentage of the gains of variable camber. A minimum of two positions for $C_{L_D} = 0.1$ and 0.3 will provide a substantial benefit with a third position further contributing in the high subsonic and low supersonic regimes.

Initial estimates of the variable camber mechanism indicated a growth in weight of wing structure (including actuating mechanism) of 500 lbs. Later more refined estimates yield a value of 250 lbs.

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B. Canard; Canard/Inlet Relationship

In addition to those performance effects (zero lift drag and drag due to lift) affected by the variable camber-biconvex wing, consideration must also be given to minimizing trim drag, providing sufficient control power, and retaining reasonable landing and take-off distances. Studies on these problems have included investigations of aft horizontal tails, canard control surfaces and unstable configurations having artificial stability devices and trimmed with either a canard or elevons.

Trim drag, the drag resulting from the deflection of surfaces to balance the aircraft longitudinal moments, is adversely affected by the transition from subsonic to supersonic flight by the aft shift in aerodynamic center (Figure 4a) and by decreasing aerodynamic surface effectiveness as supersonic speeds increase (Figure 4b). In addition, high performance aircraft wings are relatively small to reduce wave drag thus resulting in high wing loadings. High wing loadings and high altitude operation both increase airplane lift coefficients and trim moments required.

The combination of large trim moments and low surface effectiveness results in high trim drag. This trim drag can be reduced by a reduction of load on the trimming surface, by decreasing the stability of the airplane, or by increasing the moment arm of the trimming surface.

As has already been noted, elevator effectiveness drops off rapidly with increasing supersonic speeds. It is therefore necessary to provide an additional trim capability.

Landing and take-off speeds and distances, a function of wing loading and maximum lift coefficient, are limited by trimming moment available and by maximum tail clearance angles. Further, ground effect causes an aft shift of the aerodynamic center for the F-106 type airplane which increases the trimming moment required for take-off.

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Take-off characteristics can be improved by increasing trimming power through the use of a canard or an aft horizontal tail. Maximum lift coefficients can be increased by the use of wing elevons as flaps but also require trimming forces to balance the pitching moment created by the flaps.

1. Aft Horizontal Tail

Preliminary tests of horizontal tails behind a delta wing indicated the presence of adverse transonic pitching moment behavior for tails mounted above the wing chord plane. Further, the amount of fuselage and structure extending aft of the F-106 wing does not provide a sufficient tail moment arm. The problem of providing the necessary ground clearance angles at take-off and landing with a reasonable aft tail arm precluded detailed investigations of the aft horizontal tail trimming method.

However, the aft tail does provide good take-off characteristics and also provides good maneuverability and trim drag at high speeds and altitudes. At high subsonic speeds, the aft tail provides better maneuverability than the canard or elevon configurations.

2. Elevons

The elevons, as longitudinal control surfaces, provide a configuration that has superior acceleration characteristics compared to both canard and aft tail configurations. The additional wave drag and skin friction drag of a separate control surface is not present with the elevon control surface.

However, to realize this gain with the elevon system, it is necessary to minimize the 1g trim drag with speed. As noted previously, the aerodynamic center shifts its location from subsonic to supersonic speeds. This effect is approximately a 12% MAC value. Unless this moment (about the c.g.) is correspondingly balanced out by aircraft CG shift by means of a fuel transfer system, the resulting trim drag becomes excessive. For the same CG - A.C. relationship, (12% range) a canard or aft tail can provide this trim more effectively than the elevon.

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In a F-106 type single engine configuration, it is impractical to provide a C.G. shift approaching this magnitude because of the transfer volume requirements. Other factors associated with the elevon are poor supersonic high altitude trim drag and maneuverability because of the decrease in elevon effectiveness (Figure 4b). The take-off characteristics of the elevon system are inferior to both the aft tail and canard configurations. The disadvantage of the elevon is the large negative lift developed along with the trim moment.

3. Canard

The addition of a canard surface to a F-106 type airplane increases the usable lift both at low speeds, where the angle of attack is limited by visibility, sink rate, or ground clearance, and at high speeds where the maximum lift coefficient is determined by maximum trimming moment.

Planform selection of the canard is based on the canard trimming power, drag characteristics, and flutter stability characteristics. Maximum pitching moment coefficients attainable for a series of canard planforms are shown in Figure 5. The 60° swept canard provides the best average pitching coefficient.

Flutter tests have indicated that the canard is stable at speeds and dynamic pressures in excess of 20% beyond airplane speeds with the swept planform being more stable than the 45° delta surface.

Aerodynamically, the effect of an 8.8% canard, set at a fixed attitude, on the longitudinal stability is shown in Figure 4a in a plot of the aerodynamic center as a function of Mach number. As the canard size is increased, the aircraft center of gravity must be moved forward with respect to the wing in order to maintain a minimum stability margin at the critical flight condition. For the F-106 type airplane, this critical condition occurs during take-off, approach and landing.

As the center of gravity is moved forward with respect to the wing, the following trends develop in the general arrangement of the aircraft:

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a. The wing trailing edge moves back on the fuselage resulting in a less desirable wing-fuselage intersection.

b. The slopes of the aft portion of the sonic area distribution become excessive aft of a certain wing position.

c. For a constant touch-down angle and main landing gear-c.g. relationship the main landing gear must be attached nearer the wing leading edge where space and structural rigidity are critical.

A configuration having a canard surface of moderate size (about 9% of the wing area) can be developed without incurring serious penalties due to these effects. A surface of this size is not capable of performing the entire longitudinal control task, but in combination with the elevons, it produces desirable aerodynamic characteristics.

Trim drag at high lift coefficients and at high Mach numbers is significantly reduced when the longitudinal balancing moment is provided by a combination of the canard with elevons rather than by the elevons alone. The rate at which the drag increases due to deflection of the control surfaces is shown in Figure 6a. It appears possible to link the elevon and canard surfaces mechanically in such a way as to produce a minimum of trim drag at the more important flight conditions.

Minimum total trim drag occurs at the pair of points where the sum of $(\triangle C_m)_c$ and $(\triangle C_m)_e$ equals the total moment required to trim and the slopes of $\triangle C_D$ versus $\triangle C_m$ are equal for both surfaces. The canard and elevon deflections necessary to obtain the minimum drag values are shown in Figure 6b for various lift coefficients at $M = 2.35$. This curve represents the desirable gearing between the canard and elevons since it allows minimum trim drag to be achieved both at high Mach numbers and at cruise.

In addition, it allows a smooth variation to full deflection, assuring that maximum trimming power is always available.

The effect of the canard on the maximum lift coefficient is shown in Figure 7. Canard size as related to maximum lift coefficient is shown in Figure 7a. The effect of the canard

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with Mach number is shown in Figure 7b. This increase of trimming power available leads to better maneuvering characteristics at altitude and to improved landing and take-off characteristics.

The effect of the canard on maneuverability at altitude is shown in Figure 8. At high Mach numbers the 3g maneuverability ceiling is increased approximately 15,000 feet by the canard. This represents an increase of more than 25% in 3g altitude over the F-106A. The effects of the canard on the speed altitude curve of a tactical airplane are shown in Figure 9.

In summary, the aerodynamic effects of the canard provide the following:

a. Combined use of the elevons and a canard surface significantly improves take-off and ceiling performance and high altitude maneuverability, compared to the corresponding tailless airplane.

b. A fixed relationship between the canard and elevon deflections can be determined which yields near-minimum drag for the more important flight conditions and also allows full deflection of both surfaces for maneuvering.

In addition, indications are that a canard surface has a favorable effect on directional stability at the critical flight condition.

A major problem associated with the use of a canard surface on a tactical airplane is the fuselage-canard-inlet geometrical relationships. Wind tunnel tests to determine the proper location of the canard relative to the inlet duct indicated that large losses can result if the inlet is located in the conical flow field of the canard. These losses in terms of an increment of total pressure recovery are shown in Figure 10a.

The same tests showed that the canard wake entering the inlet also caused large losses. As a result, a triangular region was defined, between the canard trailing-edge shock and the

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canard wake, in which the canard effect on the inlet should be small. The 50" nose extension of the Advanced F-106 (I.B.) provided this revised arrangement.

Pressure losses at the inlet will still occur, but should be a function principally of the airplane angle-of-attack and the canard deflection. Pressure losses were measured in wind tunnel tests. At low combined angles of attack and canard deflections (< 7°) the canard has only a small effect. However, under maneuvering conditions and at high altitudes, both the angle of attack and canard deflections are increased. The effect of the canard on the inlet pressure recovery is shown in Figure 10b.

Canard losses could be decreased by locating the inlet in the triangular region below the wake and aft of the expansion from the trailing edge of the canard lower surface. However, such an arrangement in a tactical airplane becomes difficult in regards to the location of equipment in the airplane and attempting to retain a minimum drag configuration.

Because the losses associated with large canard deflections are high when the canard is located in front of the inlet, and to eliminate adverse canopy and fuselage effects on inlet performance, it becomes desirable to extend the inlet forward on the fuselage with the canard located aft of the inlet.

This arrangement adds to the skin friction drag of the airplane and also gives a weight penalty in increased duct length and in a more complicated canard support and control mechanism.

A typical structural arrangement for a delta type canard is shown in Convair Drawing SD-56-08089. The support structure shown is for a configuration where the canard is located directly off the fuselage where a carry-through structure is applicable. For the canard mounted aft of the inlet, the canard is mounted on the inlet requiring separate support through the frame. The total weight of the canard surfaces, support, and associated controls is estimated at 250 lbs. for the carry-through type.

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4. Unstable Airplane

The analytical parameter studies, discussed in Section I.B, also investigated the effect of longitudinal stability on aircraft performance. The relationships between high altitude maneuverability, stability margin and canard volume are shown in Figure 11. Compared to the effects of the canard size and location, the stability margin has a large effect on the high altitude performance.

An extension of this work has indicated that an optimum stability margin exists on the basis of high altitude performance. For the tailless airplane, this point is at approximately a -15% stability margin while for the canard, the point is at a -25% stability margin. However, the practical degree of low speed instability is limited by the control system capability. This amount of instability is less than optimum and for the present state-of-the-art, the tailless arrangement is limited to a -5% margin at low speed and the canard arrangement to - 10%.

A performance comparison is shown in Figure 12 for the tailless configuration with stability margins of 0.0 and -.05 and for the canard configuration with stability margins of 0.0 and -.10. The general results indicate that the canard version has better climb and altitude performance but has inferior acceleration characteristics. Also, take-off performance is improved by decreased stability margin and is further improved by a canard.

In order to obtain the performance advantages of static instability, it is necessary to provide the pilot with automatic stability augmentation devices which will enable him to obtain satisfactory and reliable handling characteristics since the pilot is incapable of properly controlling a statically unstable aircraft. The system, therefore, must be extremely reliable.

The device selected to provide this artificial static stability is a mechanical angle of attack computer. This "q-bob" system derives its intelligence from a bob weight and a "q_c" sensitive spring. In addition, the development of an electric force stick configuration has been approached to provide the broad versatility of electrical systems in obtaining the desired dynamic

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response, feel forces, speed stability and automatic tie-in. To attain a maximum in system reliability, a manual or back-up mode of operation is designed into the system such that in the event of the electric force stick system failure the aircraft is still flyable with the manual system.

Good dynamic response is obtained through the operation of a pitch damping function in the flaperons and acceptable feel forces are made available from a mechanical feel system similar to that of the F-106 A/B employing a q-pot and a cam controlled centering arrangement. Likewise in the event of a q-bob failure, the electric force stick system alone is capable of providing static stability to the airframe. By making the electric force stick the normal mode of operation and having the q-bob operative at all times, the desirable redundancy is provided which substantially increases the aircraft reliability. A functional schematic of the control system is presented in Figure 13.

For longitudinal control in the electric force stick mode, the pilot applies a load to the stick force sensor which is summed with acceleration and a function of pitching rate to form an error signal. The error signal commands both the canard and flaperons. The canard is actuated in this mode by engagement of the electric force stick actuator clutch and an automatic trim follow-up system removes the resultant forces developed by the emergency mechanical feel system. Sufficient symmetrically flaperon action is available through the electrical authority of the dual input flaperon valves. The motion reflected to the stick is that which is directed into the canard. On analog simulation, a proper division of the error signal input between the electric stick actuator and flaperon valves provides a smooth and desirable stick response. The pilot also can manually override the electric force stick clutch to effectively cancel any electrical system failure.

Lateral control, in the electric force stick mode, is obtained by asymmetrical action of the flaperons as in the F-106A/B. Feel and centering forces are provided by a cam controlled spring.

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Automatic longitudinal control is accomplished by generating the desired input electrically and imposing it upon the electric force stick actuator and flaperon valves in place of the force sensor output. Automatic lateral control is obtained by engagement of the full authority aileron actuator clutch and the limited asymmetrical electrical authority of the flaperon valves.

The manual mode operation is available upon disengagement of both the electric force stick actuator and full authority aileron actuator clutches. The automatic trim follow-up function at the canard becomes a manual trimming device operated from the control stick.

The canard together with the flaperons produces an extremely versatile control system which can feature many desirable effects such as smooth stick motion reflected from the surfaces and fine dynamic responses.

The landing problem for the unstable canard introduced additional problems in the flight sensing equipment which did not provide adequate control for this condition. It appears necessary to provide pitch attitude stabilization during ground roll in order to allow a normal landing procedure.

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C. Pre-Compressor Cooling and Related Studies

1. Pre-Compressor Cooling

Thrust augmentation by means of pre-compressor cooling was proposed on the Advanced F-106 in order to extend the performance envelope beyond the capabilities of the dry engine. A four phase pre-compressor cooling (PCC) test program was designed to investigate the effects of PCC on the performance of a P&W J-75 Turbojet Engine. The test program was conducted at the Arnold Engineering Development Center, Tullahoma, Tennessee, from July 30, 1957 through February 28, 1958.

The program consisted of the following phases:

Phase	Configuration	Test Purpose
I	Straight cylindrical duct.	Investigate a series of spray bar configurations to determine the optimum arrangement for use in Phase II.
II	Straight cylindrical duct with P&W, J-57-P-11 engine attached.	Investigate effects of PCC on J-57 turbojet engine performance. Uncover possible problem areas of later phases.
III	Bifurcated aircraft type duct.	Investigate two spray bar configurations to determine the optimum one for use in Phase IV.
IV	Bifurcated aircraft type duct with P&W, YJ-75-P-3 Engine attached.	Investigate effects of PCC on J-75 turbojet engine performance using a realistic inlet duct configuration.

During the planning period of the test program, the availability of the J-75 engine was questionable. Hence, the use of the J-57 during the Phase II testing. The preliminary information

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and experience obtained from Phase II contributed substantially to the success of Phase IV. A discussion of the phase IV results follows:

A P&W YJ-75-P-3 Engine, Serial P-610002, was operated for a total of one hundred and five hours of which forty-six and one-half hours were at maximum afterburning power. Of this forty-six and one-half hours, forty hours were run with water on. No adverse effects on the engine or engine controls were observed as a result of operating the engine with PCC.

At maximum RPM, the P-3 engine is compressor face temperature limited at 700° R ($M = 2.0$). Successful continuous operation up to initial inlet air total temperatures of 880° R ($M = 2.5$) was accomplished during the test program. These operating conditions are shown as open circles on the envelope for PCC performance documentation, Figure 14. At each open circle condition, the water-air ratio was varied from the minimum required to prevent the compressor inlet total temperature from exceeding 700° R up to the theoretical saturation value. Higher Mach number operation was accomplished for a short duration when the initial inlet air temperature was increased to 1100° R ($M = 3.0$) by maintaining the minimum water-air ratio to prevent compressor face temperature from exceeding 700° R. These operating conditions are shown as solid circles on Figure 14.

The net thrust, airflow, and specific liquid consumption at altitudes of 45,000 feet and 75,000 feet are shown in Figures 15 through 20. The solid lines represent the crossplots of the actual data obtained. The dashed lines represent the maximum theoretical values possible based on 100% evaporative efficiency (defined as the ratio of actual temperature drop to the maximum theoretically attainable temperature drop).

The net thrust and engine airflow performance augmentation resulting from PCC at saturation water-air ratios is given in Table I.I. All values are derived from Figures 15, 16, 18, and 19. Column Three gives the thrust and airflow attained without PCC (subscript-dry). Column Four gives the value actually attained with

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theoretical saturation water-air ratio for the free stream total pressure and temperature representing the flight conditions of Column One and Two (subscript-sat.). Column Five gives the value based on theoretical saturation water-air ratio with 100% evaporative efficiency and represents the highest theoretical value attainable with PCC (subscript - at sat. & 100% evaporative efficiency).

Column Six gives the percent of actual augmentation obtained (Column 6 = (Col. 4 - Col.3)/Col. 3) and is applicable only for flight conditions where the engine can operate in the dry condition. For the JT-4B-22 engine, which could operate without PCC within the envelope represented by open circles on Figure 14, the percentage of actual augmentation could be obtained for all conditions therein. Column Seven gives the percent of maximum attainable thrust ratio that was actually obtained (Col. 4/Col. 5).

The percent of maximum attainable thrust augmentation is found by dividing the difference between dry and actual saturation values by the difference between dry and theoretical saturation value based on 100% evaporative efficiency. As in the other cases, this percent value is only applicable for the flight conditions where the engine can operate without PCC. The percent of maximum theoretical augmentation for a series of points of Table I.I is as follows:

Speed	Altitude	% Theo. Aug.
M = 1.80	45,000'	67
1.994 (max. dry)	45,000'	76
1.80	75,000'	64
1.980 (max. dry)	75,000'	93

The trend indicated by these values is that with higher Mach numbers at higher altitudes, the higher the percent of theoretical maximum augmentation attainable. This data corroborates calculations made prior to the testing.

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With the JT-4B-22 engine which is M = 2.5 limited (dry), 80% of theoretical thrust augmentation was applied for use with the Advanced F-106 study calculations. The above trend indicates that this is justified.

The two curves of specific liquid consumption versus Mach number, Figures 17 and 20, show that the actual and theoretical SLC are in close accord, and that for some conditions at 75,000 feet, measured values of SLC are less than theoretical. This can be attributed to some thrust augmentation resulting from interstage cooling of the air due to raw water entering the compressor and evaporating in the first few stages of the low pressure rotor. At a constant water-air ratio with changing total pressure and temperature (increasing Mach number at constant altitude), the interstage cooling effect varies, giving the actual SLC curve a different rate of change than the theoretical where interstage cooling is not applicable.

Figure 21 shows the amount of water required for a constant Mach number and altitude flight condition for either actual saturation or minimum water-air ratio for a 5, 10 or 20 minute water on flight period. By following the dashed arrow line, at 45,000 feet and M = 1.90, 1430 lbs. of water would be required for a five minute flight maintaining continuous saturation water-air ratio. By interpolating, an approximate weight of water can be determined for a changing flight condition. For example, starting at 45,000 feet and M = 1.90, and accelerating to 65,000 feet and M = 2.1, requires approximately 2700 lbs. of water (a 10 minute period maintaining continuous saturation water-air ratio).

The water spray system consisted of two manifolds of twelve spray bars each, for each leg of the bifurcated duct, or a total of 48 spray bars. The spray bars were small tubes projecting radially into the duct with holes drilled perpendicular to the airflow. The water pressure was varied from 30 to 300 psi.

The effects of PCC (with 80% evaporative efficiency) on the Advanced F-106 are shown in Figure 21.1 in terms of thrust and drag variation with altitude. In addition to the ceiling increments attainable with PCC, there is a significant effect on climb and acceleration characteristics.

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Both climb and acceleration performance are a function of the excess thrust. For example, at M = 2.0 and 5,000 feet below the appropriate ceiling, the dry JT4B-22 engine has an excess thrust of 1,000# and with PCC, has an excess of 1,700#. This additional thrust also improves maneuvering characteristics in permitting higher g loadings without deceleration.

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2. Propulsion System Studies

The propulsion system studies were conducted in three phases: (a) Comparison of the J-75 (JT4B-22) engine without PCC and with PCC optimized for large thrust increases, (b) Same as (a) except that the installation was optimized for water-off operation thus considering PCC as a supplemental method of increasing thrust, (c) Study of the J-58 (JT11-5A) engine installation.

a. Initial performance of the JT4B-22 was evaluated considering maximum PCC at M = 2.5 which required an inlet capture area of 14.1 square feet. At lower Mach numbers, the bypass and additive drag of this configuration was prohibitive.

Additional evaluation to determine the best inlet size which gave high PCC augmentation and yet did not affect the low speed performance severely, resulted in an inlet sized for the air requirement of full PCC at M = 2.3. This resulted in a capture area of 11.7 square feet.

Installed engine performance is shown in Figure 22 with the total liquid flows in Figure 23. At M = 2.3, the thrust is increased by approximately 65% over the dry engine for an increase in specific liquid consumption of approximately 70% over the dry specific fuel consumption (Reference 63 of Appendix B).

The results of the pre-compressor cooling tests, discussed in the previous sections, have verified the 80% evaporative efficiency used in obtaining the above data at high Mach numbers and altitudes.

b. The JT4B-22 was also considered with and without PCC for the inlet sized for water-off operation (Reference 90 of Appendix B). The inlet for this configuration was sized for the M = 1.6 dry air requirements resulting in a capture area of 9.7 sq. feet. For normal operation, bypass would be required at higher Mach numbers if the inlet mass flow ratio is to be held to values giving minimum drag. However, additional thrust can be obtained by inducting the bypass air into the engine by using PCC.

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Installed engine performance is shown in Figure 24 with the total liquid flows in Figure 25. At M = 2.3, and 65,000 feet the thrust is increased by approximately 35% with a corresponding increase in specific liquid consumption of 35%.

c. The J-58 was included in the evaluation when the schedule of engine production became compatible with that of the fire control system and airframe. The installed performance for the J-58 is given in Figure 26 and the fuel flow in Figure 27.

In general, the J-58 provides the following advantages relative to the JT4B-22; (1) higher supersonic thrust; (2) lower afterburning specific fuel consumption; and (3) a lighter engine. The main disadvantage of the J-58 is in higher subsonic specific fuel consumptions.

With this powerplant, large thrust increases are obtained over the JT4B-22 optimized with or without water. At M = 2.5 and 65,000 feet the JT4B-22 optimized with maximum PCC gives but 84% of the J-58 thrust and at a specific liquid consumption which is 205% the specific fuel consumption of the J-58. For the configuration optimized to water-off conditions (b), 70% of the thrust of the J-58 is obtained with 135% of the J-58 specific fuel consumption at this condition.

In Figure 28, the Fuel Specific Impulse is given as a function of Mach number for the engine installations considered. The J-58 is superior to any of the JT4B-22 installations. The slight advantage that exists with the JT4B-22 engine with PCC (Optimized on dry air requirements) over the dry engine is the result of using the bypass air that is normally dumped overboard by the dry engine.

3. Internal Compression Inlet

Tests with internal compression inlets have indicated that the cowl drag of this configuration is less than that for the external compression type inlet. The comparative drag reduction through the use of internal compression inlets is shown in Figure 28.1a as a percent of equivalent thrust gain versus Mach number.

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These tests have also shown that an improved pressure recovery can be obtained over the external compression inlet at design Mach numbers greater than 2.0. The pressure recovery for a series of inlets is shown in Figure 28.1b.

In addition to these gains, the internal compression inlet more nearly approximates the engine airflow requirements than does the external compression inlet. (Figure 28.2) The result is a decrease in drag due to a lower amount of ejected bypass air.

The inlet performance of the Advanced F-106 was estimated from experimental data obtained during tests of a single nacelle internal compression inlet at Pt.Magu. The test data was corrected to account for the pressure loss incurred by the proximity of the fuselage to the inlet since the actual performance depends on the local flow conditions. This environmental field influencing the inlet airflow is a function of the fuselage shape and the position of the inlet relative to the fuselage. To verify the predicted performance, a 1/12 scale model incorporating a forward fuselage, canopy, and twin side-inlets (Figure 29) was tested in the Langley Aeronautical Laboratory 4' x 4' Unitary Plan Wind Tunnel.

Experimental results have indicated that fuselage forebody creates an overvelocity field at the face of the inlet. In addition, a velocity gradient, due to fuselage cross flow and canopy influence, exists across the inlet face. The net result is a loss in inlet pressure recovery.

Addition of internal spike support struts caused excessive changes in internal diffusion rates; and a bleed system, not optimized for the inlet throat location, resulted in lower pressure recoveries than previously attained. A loss in inlet pressure recovery also can be attributed to the low Reynolds number of the test. Figure 30 gives the effect of Reynolds number on the inlet pressure recovery at the design Mach number of 2.50. Noted on the figure is the Reynolds number which would result for a free stream $M = 2.5$ at 60,000 feet. In Figure 31, the predicted inlet pressure recovery is shown with the reduction due to the above losses and the resulting experimental values.

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The effect of the canard on the inlet performance is discussed in Section II.B.3.

The inlet installation gave no indication of twin duct instability and proved to be relatively insensitive to changes in angle of attack between 0 and 3 degrees and in angle of yaw between ± 2°. At off design Mach numbers, the inlet performance was good (a recovery of .87 at M = 2.0 was attained), but still below the predicted values.

The results of the tests lead to the following recommendations:

a. Revise the fuselage forebody to result in a neutral or compression field at the inlet stations.

b. Locate the inlet forward of the influence of the canopy.

c. Locate the canard aft of the inlet station (Section II.B.3).

The variable geometry internal compression inlet, designed to operate over a wide range of flight conditions, requires a precise automatic control system to optimize inlet performance. The variable geometry features of the proposed inlet include a translating spike for varying the internal contraction, and a bypass system for positioning the terminal shock (Figure 32). The proposed control system will position the spike as a function of flight Mach number, and the bypass as a function of terminal shock position. Normal control operation is achieved when the terminal shock is downstream of the inlet throat. The desired control action is to maintain the terminal shock as close to the inlet throat as possible with an adequate response rate to prevent terminal shock regurgitation. This action is accomplished by forming a ratio between the spike surface static pressure in the vicinity of the terminal shock, and the total pressure further downstream. The ratio is then supplied to an analyzer, the output of which is used to position the bypass door. As the bypass door is opened or closed, the terminal shock moves downstream or upstream in response to pressure changes downstream of the shock.

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The sequence of events in the control system operation is as follows: As the terminal shock structure passes over the static pressure sensor a pressure ratio change signal results. The analyzer examines the change signal and moves the bypass so as to return the terminal shock to its original position. In the event of terminal shock expulsion, the bypass is opened to maximum position and the spike is extended at maximum rate. When the inlet has restarted, the normal control sequence is resumed.

To determine the feasibility of this method of inlet control, an analog computer study of the system was made. To provide an understanding of the scope of the analysis, a control system block diagram is included in Figure 33. The diagram indicates which interactions within the control system were considered, and enumerates the control parameters investigated. To provide as realistic analysis as possible, results of wind tunnel tests were included in evaluating system parameters. It was necessary, however, to derive analytic expressions for some parameters such as diffuser and terminal shock dynamics (Reference 85 of Appendix B). The most severe transient disturbances to the control system were assumed to be a step input to the throttle position.

From the analytical studies of internal compression inlet controls, the following conclusions are made:

a. The control system is feasible.

b. The optimum control system requires the bypass to be close to the terminal shock.

c. Satisfactory control operation depends on the pressure sensor location, saturation limits of the pressure sensor, control set points, loop gain, dynamic response of the various components (mechanism, diffuser, valve, etc.) and non-linear gain compensating devices.

d. A wind tunnel test program is necessary to check the validity of the parameters used in the analog study, and to evaluate the inlet control system under actual operating conditions.

The proposed unit for the inlet controls test program is shown by Convair Drawing WT-57-82515.

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4. Ejector

Effective thrust augmentation of a convergent nozzle is provided by an ejector. For the Advanced F-106, a series of divergent ejectors were tested statically and in wind tunnel installations.

Preliminary net thrust ratios, defined as the ratio of the convergent nozzle plus divergent ejector net thrust of the propulsion system to the convergent nozzle net thrust of the propulsion system, for the Advanced F-106 at 35,000 feet for max. JT4B-22 A/B power are shown in Figure 34.

The predicted curve was estimated from various ejector data and it was assumed that the performance could be attained through a development program. The ideal convergent-divergent nozzle is included as a reference.

The estimated net ejector thrust ratios for the static test compares favorably with the predicted values. The wind tunnel results had not been completely analyzed at publication time. The data were recorded for configurations that were not on design at each Mach number. Therefore, the results must be analyzed to obtain correlation with the static test results.

The design studies on ejectors were initially directed to obtaining a completely variable ejector in diameter at both the throat and exit by an overlapping leaf arrangement. Thus an ideal throat and exit diameter could be maintained for any Mach number from take-off to maximum speed. This design required two individual actuating systems plus a complex seal between overlapping leaves. It is anticipated that such a mechanical device would require excessive maintenance in service. Further, the weight of this ejector has been estimated at 1000 lb. This ejector design is shown in Convair Drawing SD-57-08119.

To eliminate the undesirable problems of this design, a different design philosophy was evolved, that of providing an effective variation by means of bleed doors. The ejector throat and exit diameters are designed for M = 2.5 optimum values. Bleed doors,

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located at the throat of the ejector, allow ambient air to enter the ejector, thus preventing over expansion of the exhaust gases. Although the ideal expansion ratio is not attained for all Mach numbers, the net thrust ratio approaches the predicted performance.

The operating mechanism of this type ejector is greatly reduced and the weight is estimated at 400 lbs. This ejector design is shown in Convair Drawing SD-57-08161.

5. Related Studies

The effect of higher performance and increased temperature on various aircraft sub systems was considered in this study.

a. Air Conditioning System

An air cycle system is proposed for the Advanced F-106. Although an air cycle system has limitations, the forced air requirements of the fire control system render a vapor cycle system unfeasible.

The air cycle system uses engine bleed air as a primary fluid for air conditioning and pressurization, and engine inlet diffuser air as a sink for heat rejection. The basic components of the system consist of an air-to-air heat exchanger, a water boiler, a turbo-compressor unit, necessary valves, ducting and controls. For an engine with PCC, a mechanical water separator would be required in addition to the above components.

A 5.5 psi cabin pressure is recommended rather than an 8,000 ft. equivalent pressure as a substantial increase in cooling capacity with reduction in structural weight may be achieved by a lower cabin pressure schedule. The cabin pressurization schedule controls the maximum available pressure ratio across the refrigeration turbine at a given operating condition. A higher pressure ratio yields a greater temperature drop obtainable through the turbine.

To maintain maximum airplane performance by keeping bleed airflow to a minimum, requires insulation of some compartments. In addition, a net reduction in airplane weight may be achieved by insulation.

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b. Radome and Canopy

The radome of the Advanced F-106 will be subjected to temperatures of 430° F at the base of the nose boom and 380° ± 10° F at the surface of the radome cone. Two laminates, Vibrin 135 and 136, show promise on the basis of preliminary tests. The material used will depend to a large degree on the transmission characteristics that are established. Developments indicate that no fundamental problems should be encountered in establishing a satisfactory radome configuration.

Effects of temperature on the windshield have been considered by applying the most severe flight profile of aerodynamic heating to the windshield by a digital program. The resulting transient temperatures on the windshield surfaces are shown in Figure 35A. The thicknesses of the recommended panel components and the resulting temperature gradients are shown in Figure 35B. The outside ply of glass will be subjected to a severe thermal shock, thus requiring a low coefficient of expansion glass. The high internal temperatures require a high temperature silicone compound for the interlayer material.

A development program will be necessary to determine the compatibility of the silicone interlayer and an electrically conductive coating (for anti-icing and anti-fogging). Optics and adhesion in the laminate are also in need of improvement.

The high boundary layer temperatures preclude the use of plastics for the canopy. Because of the limited development of high temperature plastics, the canopy construction must be similar to that of the windshield.

c. Engine Oil Cooling

Fuel is used as the main heat sink in providing engine oil cooling. Supplementary cooling may be required at high altitude, low Mach number conditions and during low power deceleration from high speeds. At low Mach numbers, this supplementary cooling can be accomplished by air-oil heat exchangers. At the high speed conditions, a water boiler type unit is sufficient.

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Since the low speed supplementary cooling will be required more frequently, a combination cooling system using fuel, ram air, and water was considered. Figure 36 shows a schematic circuit of an engine oil cooling system that includes these 3 coolants. The water boiler can be pressurized to provide temperature control.

d. Aerodynamic Heating

The high temperature of the stagnation and boundary layer air over the aircraft surfaces limits the design flexibility. Several restrictions are imposed on materials selection, compartment cooling and the operating life of the airframe and equipment.

General exposure temperatures for a $M = 2.5$ design are 375° F or less for the major portion of the airframe thus permitting the use of aluminum alloy construction. Leading edges of engine inlet ducts and wings as well as the area around the engine turbine will be exposed to temperatures between 375° to 425° F and will require a higher temperature material as will the internal walls of the inlet duct.

Internal structure, except in the area of the engine, will be at a lower temperature than the exterior surfaces. However, stiffener flanges adjacent to and in contact with the skins, will have temperatures approaching those of the skins. In Figure 37, the temperatures through a typical section are shown as they approach equilibrium temperatures during a high rate of acceleration to V_{max} from a cold soak condition. The temperature gradients giving rise to thermal stresses are also shown. If the web in this example were partially submerged in fuel, then the thermal gradients would be appreciably higher.

Engine compartment temperatures can be reduced by a radiation shield. Figure 38 indicates such a shield must be used to maintain the temperature within the range of aluminum or titanium capabilities.

Equipment compartment temperature control can be obtained by a combination of insulation and internal cooling. Cooling and insulation requirements may be estimated from the

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materials used, external temperature conditions and the internal environment that must be maintained. In non-conditioned compartments, the ambient air temperature will reach an equilibrium temperature between stagnation and boundary layer temperatures. Many of the components that are normally located in non-conditioned areas may have to be relocated unless they are developed to withstand higher operating temperatures.

Fuel temperatures as a result of aerodynamic heating will not exceed 200° F. Figure 39 gives the temperatures of the fuel in two wing tanks. This example assumes four separate uninsulated fuel tanks in each wing, a fin fuel tank and 7000 lbs. of fuel in a fuselage tank with the fuel sequencing as used for C.G. control. Since all of the fuel passes through wing tank #3, the temperature history of this tank is of most interest.

For airplanes with a design $M = 3.0$ or higher, some type of fuel tank insulation will be required. A general evaluation has indicated that a stainless steel honeycomb skin may be a sufficient barrier for 5 to 10 minutes of flight at $M = 3.0$.

Explosion hazards of JP fuel vapor will result at temperatures above 450° F. When fuel cells become heated to this temperature, about $M = 2.7$, some inert gas must be used for tank pressurization. The $M = 2.5$ airplane will require no special precautions against fuel explosion hazards.

e. Structural Heating Criteria

The structural effect of the aerodynamic heating as speeds are increased has led to the selection of 2014 aluminum alloy rather than 7075 because of its superior high temperature strength. However, it will be necessary to obtain additional information on the at temperature properties of this material prior to its use beyond temperatures of 300°. To do this, Convair is engaged in a test program to expand the ANC-5 data shown in Figure 39.1.

A series of cross-plots from interpolated data of Figure 39.1 has resulted in a relative structural weight variation with Mach number and exposure time as shown in Figure 39.2.

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The data has been normalized with respect to a M = 2.3 design, inasmuch as the at temperature strength for this point has a constant value up to 100 hours (Figure 39.1, Point A). Thermal stress effects have also been included.

The optimum points of these curves lie in the vicinity of M = 2.5 with respect to the structural weight increase. A 20 hour exposure, for example, with an increase of speed from M = 2.3 to 2.5, requires a 26% structural increase in weight while an increase from M = 2.3 to 2.6 requires a 66% increase in weight over the M = 2.3 design.

On the other hand, the weight increase as a function of exposure time is relatively linear, For the F-106-30 (Section IV), providing for 100 dash missions,requires a 35 hour exposure design.

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D. Escape Capsule

With the increasing operational speeds and altitudes of modern weapons systems, the requirements of a safe escape system for the pilot become more stringent. On the escape capsule studies, the following requirements have been considered:

1. The occupant must be completely enclosed when separated from the aircraft. The capsule must provide environmental protection against dynamic pressure, low ambient pressure, and lack of oxygen.

2. The capsule must be an aerodynamically stable free flying body of low drag configuration in order to avoid excessive acceleration on the occupant after separation from the aircraft.

3. The complete capsule must be recovered while still occupied by the pilot.

4. The capsule shall serve as a survival shelter after landing, on land or water.

Four types of capsules were considered as having possible application to the Advanced F-106. These are:

1. The Nose Capsule which consists of all the aircraft structure forward of the aft end of the pilot's compartment. Because of the large radar antenna and amount of electronics required in the nose of the Advanced F-106 and the large fuselage cross section, the resulting weight of the capsule would be approximately 5000 lbs. and have a cross-sectional area of about 20 sq. ft. The size of the stabilizing devices, recovery system, and ejection power unit would be large and thus impose a severe weight penalty on the airplane.

2. The Enclosed Seat Capsule (Figure 40a) is essentially an ejection seat in which the pilot is enclosed. This is the minimum unit to meet the escape capsule requirements. It is the minimum weight capsule and will have the smallest recovery package and power

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unit, and further, will have the least number of disconnects at ejection. The stability and loading problems are similar to those of the ejection seat, therefore, some type of pre-ejection positioning to near trim attitude will be required. The enclosed seat capsule, as it is installed within the cockpit enclosure, will require an increase in the canopy size to provide ejection clearance. Stowage of the moving parts of the capsule enclosure will cause some interference with normal flight operations while emergency actuation and sealing of these closures prior to ejection presents additional design complexity.

3. The Canopy Capsule (Figure 40b) consists of the aircraft canopy and adjacent structure. Prior to ejection the pilot is positioned in a semi-supine attitude and a clamshell door is closed to seal the bottom of the canopy. This unit utilizes existing structure as the major portion of the canopy while still leaving behind most of the cockpit equipment. However, it requires a complex mechanical system to position the occupant and close the clamshell. In addition, this re-positioning must be provided with the positioning space requirement and will result in a larger canopy.

4. The Cockpit Capsule (Figure 40c) features integration of the normal flying operation and the escape function by permanently enclosing the complete aircraft cockpit. Although the ejected weight (1500 lb.) is greater than for the canopy type (1100 lb.) or the enclosed seat (600 lb.); it does not represent a large aircraft weight penalty as a major portion of the ejected weight consists of existing aircraft structure and equipment in the cockpit area. Further, it does not require pre-positioning devices prior to ejection. This capsule, since it allows more pilot mobility, offers superior survival shelter after landing than the enclosed seat or canopy types. The canopy size may be reduced from the F-106 canopy since its design will not be based on clearances required for an ejection seat.

The nose capsule, because of the severe weight penalties, and the enclosed seat capsule, because of the mechanical complexity required to insure sealing along with its installation requirements, were not investigated in detail. Aerodynamic testing, analytical

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studies and design evaluations were directed towards the canopy type and the cockpit type capsules. Basic single-place configurations for these types are shown in Convair Drawings SD-57-08134 and SD-57-08174. A two-place configuration is shown in Convair Drawing SD-57-08136.

The investigations were concerned with the sequential steps required in the operation of the escape system. These steps are:

1. Switch to emergency pressurization. The air in the cabin will be trapped at ejection and will be sufficient to prevent de-compression and anoxia, however, because of probable leakage an emergency pressurization system is provided to maintain a cabin altitude of 12,000 feet during descent.

2. Deploy stabilizing devices. Free flight stabilization of the capsule may be accomplished by means of fins attached directly to the parachute compartment or attached to long booms extending aft from this compartment.

3. Fire rocket. The rocket ejection unit will provide sufficient thrust to insure satisfactory clearance of the aircraft under all escape conditions, and ejection to a safe recovery height under low altitude escape conditions.

4. Disconnect capsule from airplane.

5. Disconnect controls, electrical leads, etc.

6. Deploy drogue parachute. Immediately after separation from the aircraft a small parabrake will be deployed which will slow the capsule to safe speed for deployment of the main parachute.

7. Deploy main parachute. A combination speed and altitude sensing device will control the main parachute and will deploy it only after the capsule has slowed to safe deployment speed and descended to a pre-set altitude.

8. Inflate auxiliary flotation devices (if required).

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9. Collapse main parachute after landing.

The design studies have determined the parameters required to mechanize the ejection sequence and the effects of the capsule on the airplane. Since both the canopy and cockpit capsules are partially internal and partially external to the aircraft, they must be installed in a large cutout in the aircraft fuselage. This is the same type of cut out, although larger, and in the same vicinity as the normal cockpit cut out in most airplanes. Conventional structural design can be followed in reinforcing this cut out with large sill beams. Where necessary, bulkheads, stiffeners, etc., may be added beneath, alongside, and in front or behind the capsule to provide necessary fuselage structural integrity or to isolate conditioned compartments. The capsule will be attached to the aircraft by quick release fittings.

The aerodynamic studies on the capsule included the selection of feasible configurations, wind tunnel tests to determine aerodynamic characteristics, analog computer studies to ascertain the ejection characteristics, slingshot tests of improved configurations, determination of airloads on a typical capsule and a brief study of a two-man capsule configuration.

As the study progressed, wind tunnel tests showed that the configurations selected were lateral-directionally unstable and that a severe trim change occured with Mach number.

Slingshot tests on a series of free flight model configurations (Figure 41) were conducted. The results indicated that it is difficult to achieve satisfactory dynamic characteristics for the capsule with stabilizing fins attached to the body. Attaching stabilizing fins to booms projecting aft and away from the capsule enabled much better flight characteristics to be attained, due to increased damping and a higher fin effectiveness in free stream air away from the capsule. The poor directional stability characteristics can be corrected with a boom-fin combination. In subsequent low speed wind tunnel tests of the boom-fin stabilized slingshot developed capsule, the same results were attained.

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The pitching moment characteristics of the capsules studied were such that the designs would not provide a satisfactory escape at all the design flight conditions for any feasible combination of rocket thrust, rocket burning time and tail incidence setting as determined by analogue computer studies. The results for the two-man capsule, using estimated aerodynamic characteristics, show that theoretically it could be utilized successfully with the correct combination of parameters. However, the margin of safety would be small.

The difficulty lies in the basic aerodynamic characteristics common to all the capsule types examined. In all cases a large change in pitching moment at zero lift occured with Mach number, especially with stabilizing fins attached. This characteristic caused a large variation in trim lift coefficient with Mach number for a given tail incidence setting. The result is that to attain the minimum lift coefficient required for successful escape at $M = 1.15$ at sea level, a negative tail incidence must be used, while to limit the lift coefficient at $M = 2.5$, a positive tail incidence is required.

To correct this problem and obtain a capsule shape which will give satisfactory ejection characteristics at all flight conditions, the capsule must be designed to give a more symmetrical shape, eliminating the flat underside of the nose.

In general, the escape capsule studies have resulted in the following basic conclusions:

1. A capsule of the completely enclosed cockpit type is practical for application to an Advanced F-106 type aircraft. Although a major portion of this study was directed towards both the canopy type and cockpit type escape capsules, the cockpit type was selected since the added mechanical complexity of the required positioning and closure mechanism of the canopy capsule was not warranted on the basis of a slight superiority aerodynamically to the cockpit type.

2. Development of a capsule for integration into this type of airplane will require substantial development engineering prior to incorporation into a production aircraft. Additional aerodynamic development work is required to further verify the configuration recommendations developed during the course of this study.

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3. When the capsule free flight problems are solved to provide a shape that is completely satisfactory over the entire required performance area, the task of designing it into the aircraft should involve no problems which cannot be solved by known design techniques.

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III. Fire Control, Armament and Weapons Systems Factors

Concurrent with the studies on aircraft design features discussed in Section II, studies were directed towards the controlling factors of the fire control system and the operational aspects of the Advanced F-106.

A. Operational Aspects

The operational concepts resulting from these studies were applied to the F-106-30, a description of which follows in Section IV of this report. The following summary is derived from one of the supporting documents of the F-106-30. (Reference 15)

The F-106-30 is designed for zone of interior defense against manned bomber threats in the 1962-1965 era. It represents a logical extension of the F-106A and the F-106C (F-106A with 40" radar antenna) concepts, including an improved fire control system, armament, engine and aerodynamic capabilities.

It possesses the ability to destroy bombers at speeds of M = 2.0 and greater and also to destroy air-to-surface cruise type missiles traveling at speeds of M = 2.5. Further, it can provide target kill from sea level to 100,000 feet in altitude. For the time period considered, it represents the maximum defense against a bomber raid for a given defense dollar. This is a result of the relatively small difference in penetration distance effected by a change in Mach number of 0.5 while the same 0.5 change in Mach number alternately requires a more expensive steel airframe and its development costs in relation to the wide technology available on the lower speed aluminum airframe.

B. Fire Control and Armament Factors

The objectives of the study on fire control and armament systems were:

1. Study the detailed characteristics of proposed systems to determine that system or combination of systems most suited to

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Convair requirements for an interceptor weapon system to be operational in the 1962 time period.

2. Conduct independent studies to determine ways of improving the counter-countermeasure and low altitude attack capabilities of the selected system.

3. Make recommendations and suggest further improvements where required.

In order to provide a common basis for comparison of proposed fire control systems and associated armament, the following criteria were assumed:

1. The interceptor will be required to attack air breathing threats having a velocity range from M = 0.8 to M = 3.0 and altitude varying from sea level to 70,000 feet.

2. Forward hemisphere attack capability will be required against these targets.

3. A minimum radar detection range of 40 nautical miles would be required with a closing rate of M = 2.0 on a 10 square meter target.

4. Maximum immunity to all types of countermeasures including decoys, chaff, range gate stealers, confusion repeaters, inverse conical scan reapeaters, and automatic barrage and multi-spot noise jammers is desired.

5. Maximum all weather performance at low altitude is desired.

Of the proposed fire control system considered, there are two systems which are superior because of their counter-counter measure techniques, which when coupled with the decision making capability of an ECM trained operator result in virtually no degradation in armament launch capability due to ECM directed against the interceptor radar. Pertinent characteristics of these two

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systems, both proposed by the Hughes Aircraft Company, are given in Table IV. The two systems are the Advanced MA-1 and the Pulse Doppler.

The RCA Aries Fire Control System was also considered but was not considered in detail since it employed two-way conical scan angle sensing and a magnetron of insufficient tuning rate to operate successfully against an automatic spot jammer. It also lacks a digital computer, which limits system flexibility in utilizing various armaments.

As the two systems considered have many features in common (Table IV), the determination of the most effective system can be made by considering the following areas of performance.

1. Performance under close control. The operational capability of the interceptor weapon system when operating in a non-jamming environment and at relatively high altitudes (above 10,000 feet) where ground clutter returns do not affect system operation.

2. Performance at low altitude. The operational capability of the interceptor in a non-jamming environment at altitudes where ground clutter returns can affect system operation, and where vectoring information may be poor due to insufficient low altitude coverage of the ground radars.

3. Performance against ECM. The operational capability of the interceptor weapon system when operating in an ECM environment, where vectoring information is certain to be poor and may only consist of an angular direction from a ground radar site.

The results of these studies lead to the recommendation of the Pulse Doppler Fire Control System and its associated armament, the GAR-9, missile for the following reasons:

1. The significantly greater detection range of the radar allows effective employment of the weapon system with a minimum amount of information from the ground environment.

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2. Very low altitude performance, if required of the interceptor weapon system, can only be obtained using pulse doppler velocity discrimination techniques for clutter rejection.

3. The system retains a large portion of its capability in an ECM environment due to the CCM characteristics of the pulse doppler radar and missile.

Supporting studies conducted by Convair indicate the inclusion of the following provisions into the pulse doppler system would result in improved counter-countermeasures capability.

1. Inclusion of an alternate unambiguous range "chirp" mode (pulse compression technique) of operation with pulse-to-pulse frequency hopping would insure forcing of barrage jamming as well as allow use of a Type A presentation (range versus signal amplitude).

2. Steps should be taken to initiate training of radar operators specifically in anti-jamming techniques and tactics. A well trained operator may mean the difference between a bomber kill and abort or decoy kill in many jamming situations.

3. Inclusion of infrared augmentation provide advantages in supplementing the radar in ECM, clutter or malfunction conditions. Further, the infrared equipment can provide a passive attack capability through maneuver ranging techniques.

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IV. Tactical Version Resulting from These Studies

The weapons system proposed by this Advanced F-106 Study Contract, AF 33(600)-30169, Supplementary Agreement 14, is the Convair designated F-106-30. In conjunction with these studies, the proposed weapon system is the evolution of 10 years of continuous improvement and development of fire control, armament, engines, and airframe by the F-102/F-106 prime contractors.

A. Airframe Description

The configurations that have been developed are shown in Convair Drawings SD-58-06001, SD-58-06002, SD-58-06012 and SD-58-06013. These two configurations vary only in the canard-inlet relationship. The factors associated with this relationship were discussed in Section II.B. The effect of the canard in front of the inlet does not become significant until a Mach number of 2.0 is attained, except during maneuvering conditions when a greater pressure loss occurs due to increased canard deflections. On the other hand, the version with the inlet duct moved forward and the canard mounted on the side of the inlet incurs a weight penalty of 700 lb. and adds to the skin friction drag of the airplane.

Prior to a selection between the two configurations, additional wind tunnel testing is required to verify the estimated effects on the actual canard-inlet fuselage geometry on the canard forward configuration and further investigations on the effects of added' drag and changes in pitching moment on the inlet forward version. These results will then be developed into a performance comparison between the versions to provide a basis for the selection.

The design features of the study that are incorporated into the F-106-30 include a canard, internal compression inlets, variable camber, and bi-convex wing. In addition, a ventral fin is used in conjunction with a vertical fin to yield more efficiently high supersonic directional stability.

The Fire Control System is the pulse doppler system under development by Hughes Aircraft Company as discussed in Section III of this report. Two GAR-9 weapons are carried as the armament loading. Components of the Fire Control System are packaged as in the F-102/106 airplanes.

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The airplane is configured around a pilot and radar operator/navigator to most effectively utilize the counter-countermeasure capabilities of the fire control system. Space provisions are available for adding flight controls in the rear cockpit for trainer versions.

The crew escape system recommended for this application is the Industry Crew Escape System Committee "B" type seat. This system is currently under development with speed-altitude specifications that are equal to or greater than the design flight envelope of the F-106-30 except for flight Mach numbers in excess of 2.11. It is believed that the system can be qualified to the M = 2.5 flight speed of the F-106-30.

The type "B" supersonic ejection seat is recommended for the F-106-30 in place of the escape capsule for the following reasons:

1. The "B" seat will provide safe escape within the complete flight envelope of the airplane. The maximum q for this airplane is the same as for the F-106A for which the "B" seat is being designed and the effect of increased Mach number and altitude on the "B" seat performance is expected to be negligible.

2. The only advantages to be obtained from a capsule type escape system in the aircraft would be in the elimination of the necessity for wearing partial pressure suits and in providing a survival vehicle and shelter after escape and before rescue. The weight penalties associated with both the "B" seat and the escape capsule are shown in Table II. The additional weight increment of the capsule is 240 lbs. over the "B" seat.

3. The "B" seat will be in production in late 1958 and therefore will not affect the delivery schedule of the F-106-30. A capsule, on the other hand, will require approximately three years of development to provide a capsule for a prototype airplane.

4. The capsule study program did not solve all of the associated problems but did lead to corrective recommendations on the primary configurations tested. As such, additional development work should be undertaken before scheduling a capsule into an aircraft.

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5. Such capsule development and testing would increase the cost of the F-106-30.

The Pratt and Whitney JT-11 (J-58) engine is proposed for the F-106-30 weapons system. This engine is the best available during the proposed time period of the F-106-30.

Convair proposes to utilize this engine without the added thrust augmentation of pre-compressor cooling which has been shown possible in tests with the J-75. Reasons for this are as follows: The J-58 engine develops sufficient thrust to attain the design Mach number of the aircraft without augmentation; steady-state altitude performance 12,000 feet in excess of that required for missile launch to its limit performance altitude of 100,000 feet is obtained; zoom altitude performance provides an additional 15,000 feet prior to afterburner blow-out.

The addition of pre-compressor cooling to this J-58 propulsion system would require:

1. Larger inlet capture area. This handicaps the aircraft in two ways. The increased capture area reflects itself in wider fuselage lines which cannot be completely faired out prior to the maximum cross section of the aircraft. Hence, fuselage fineness ratio is decreased and wave drag is increased. Also, when water is not being used, larger quantities of air must be by-passed, resulting in less efficient performance under this condition.

2. Water Control System. Details of the control system required have not been ascertained completely at this time. Some additional features would be necessary in the inlet and engine controls. Water pumps, lines, and spaybars of approximately 200 lbs. are required.

3. Water and water tank weight and volume. Space and weight would be required depending upon the water volume desired. Since water requirements depend on mission requirements, the desired water tankage can vary considerably. A problem exists in providing a sufficient supply for a meaningful capability without penalizing other missions unduly.

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4. The concept of the F-106-30 is to produce the minimum vehicle required, in cost and size. Therefore, considering the above factors and the performance obtained by means of the basic engine alone, P.C.C. has not been included in the F-106-30.

Relative to the JT4B-22 engine with P.C.C., the J-58 engine provides a higher thrust at a lower total installed weight (tankage and controls) and with lower specific liquid consumption at high power settings. A comparison of the installed performance of these two engines is given in Section II.C.2.

The water required for P.C.C. with the JT4B-22 may be converted into fuel with the J-58 to provide better range performance. In the F-106-30, a 15,000 lb. fuel capacity is provided through the use of fuselage, tail and wing integral fuel tanks. An aft fuel transfer system, similar to the system in the F-106A, is used to reduce the stability margin for supersonic flight.

The weight of the F-106-30 has been estimated on the basis of a structural design gross weight of 35% of internal fuel remaining, with a 7.0g symmetrical maneuver load factor to M = 1.73 and 5.33g at higher Mach numbers. The structural weight provides for a 35 hour life at M = 2.5 (above 35,000 feet). Table III gives a weight summary for the F-106-30 for both the canard forward and inlet forward (canard on inlet) versions.

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B. Performance

The performance presented in this section is based on the installed thrust of the J-58 engine with the canard located in front of the inlet. The airplane configuration is such as to provide a minimum stability margin of 3% at the critical low speed condition. With the fuel transfer system, the average C.G. for subsonic flight is 19% $\bar{c}$ and for supersonic flight is 23% $\bar{c}$.

The speed-altitude performance (Figure 42) compares closely with that of the Advanced F-106 (Figure 1). Acceleration times and time to climb have been significantly reduced (Figure 43) from those discussed in Section I.A (8.2 minutes to M = 2.3 at 60,000 feet) primarily through the higher thrust ratings of the J-58. The constant time climb performance that is attained between Mach 1.5 and 2.5 is due to improved climb performance at higher speeds which compensates for the increased acceleration time to higher speeds. The supersonic climb from low altitude is an optimized path with a climb schedule of:

1. Take-off and climb to arrive at M = .95 at 10,000 ft.

2. Accelerate and climb concurrently to arrive at the final Mach number along the V limit line (Figure 42).

3. Climb to 50,000 feet.

This excellent climb performance is one of the benefits gained in the F-106-30 when designing to the minimum required aircraft philosophy.

Two typical missions are shown in Figure 44. The dash mission is illustrated in two parts.

1. A 500 nautical mile mission for use over the continental U.S. where a landing field within 100 nautical miles of combat is selected.

2. A normal return to home base mission with a radius of 345 nautical miles for maximum dash operation in coastal areas.

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In Figure 45a, the intermediate missions between the dash mission and the maximum radius mission are illustrated by a plot of time versus distance. Alternate mission capabilities of the loiter type are shown in Figure 45b as a function of the loiter time available versus lateral distance from the loiter point to the combat point. All missions shown provide landing with Sage reserves of 15 minutes loiter at 45,000 feet and 10 minutes loiter at sea level.

Take-off and landing distances are shown in Figure 46. Although the increase in gross weight of the F-106-30 at take-off is about 14,000 lbs. compared to the F-106A, the take-off distance over a 50 foot obstacle has only increased about 500 feet due to the added trimming power of the canard control surface and increased take-off thrust of the J-58.

The performance of the F-106-30 could be increased by utilizing an unstable configuration. Primary gains (as discussed in Section II.B.4) would be realized in ceiling capabilities, maneuvering at high altitudes, and conditions related to the maximum trimming power available such as at landing and take-off. These gains, although desirable, have not been included into the F-106-30, since the development time required to produce the complex and extremely reliable control system necessary in an unstable configuration would seriously delay the proposed schedule for the F-106-30 as well as add to the unit cost.

Even without the altitude increments available with an unstable configuration, the F-106-30 has a performance envelope that will fully utilize the altitude capability of the armament, the GAR-9 missile, that is carried. The missile can be utilized to its design limit of 100,000 feet upon launching from a snap-up attack at 60,000 feet or more.

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TABLE Ia

Approximate Performance Effects of Variable Camber

M	Phase		Increment of Performance (Bi-convex Wing) $(C_{L_D} = 0.3) - (C_{L_D} = 0.1)$ (+ indicates advantage of $C_{L_D} = 0.3$ camber)
.8	35,000' Acceleration	Kts/Min	+21.3
.9	35,000' Acceleration		+11.8
	Ceiling	Ft.	+ 1800
	Cruise	% Change	+ 11%
1.0	35,000' Acceleration		- 4.9
	Ceiling		+ 2000
1.5	35,000' Acceleration		- 32.7
	Ceiling		- 700
2.0,2.5	————	————	Data is not available, but the effects will be minus.

TABLE Ib

Approximate Performance Effects of Biconvex Section (with variable camber wing)

M	Phase		Increment of Performance (Biconvex Section - NACA Section) (+ indicates advantage of biconvex section)
.8	35,000' Acceleration	Kts/Min	-2.3
.9	35,000' Acceleration		0
	Ceiling	Ft.	- 600
	Cruise	% Change	0
1.0	35,000' Acceleration		- 1.4
	Ceiling		- 400
1.5	35,000' Acceleration		+ 4.7
	Ceiling		0
2.0	35,000' Acceleration		+ 30.6
	Ceiling		+ 600
2.5	35,000' Acceleration		+ 60.9
	Ceiling		+ 1000

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TABLE I.I

Net Thrust and Air-Flow Performance Augmentation Resulting From PCC at Saturation Values of Water-Air Ratios for P&W YJ-75-P-3, P-610002 at Max. AB, PT1 = PTo

NET THRUST

(1)	(2)	(3)	(4)	(5)	(6)	(7)
h	M_o	(Fn)Dry	(Fn)Sat.	(Fn) At Sat. & 100% Evaporative Eff.	%Fn Augmentation	%Fn sat / Fn_{sat}-100
45000	1.80	12250	14700	15900	20.0	92.4
	1.995 (Max.Dry)	14350	18200	19400	26.8	93.8
	2.20	- - -	22100	23700	- - -	93.2
75000	1.80	2670	3200	3500	19.8	91.4
	1.98 (Max.Dry)	2900	4250	4350	46.5	97.7
	2.20	- - -	5650	6000	- - -	94.1
	2.50	- - -	7850	9150	- - -	85.8

ENGINE AIR-FLOW

h	m_o	(Wa)Dry	(Wa)Sat.	(Wa) At Sat. & 100% Evaporative Eff.	%Wa Augmentation	%Wa sat / Wa_{sat}-100
45000	1.80	160	180	192	12.5	93.7
	1.995 (Max.Dry)	190	232	247	22.1	93.9
	2.20	- - -	300	318	- - -	94.3
75000	1.80	35.5	44	46	23.9	95.7
	1.98 (Max.Dry)	43	54	58	25.5	93.1
	2.20	- - -	73.5	80	- - -	91.8
	2.50	- - -	112	122	- - -	91.8

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TABLE II

ADVANCED F-106

Weight Comparison-Escape System

The following represents the estimated increment in weight due to escape system installation in the airplane.

Item	Max. Design q=1920 psf (752 Kn IAS)		Max. Design q=2600 psf (880 Kn IAS)	
	"B" Seat	Capsule	"B" Seat	Capsule
Seat	350	45	460	45
Structure		115		165
Ejection Unit		90		90
Recovery System		130		135
Stabilizing Devices		100		135
Survival Gear		70		70
Emer. Breathing Equip.		20		20
Pilot Equipment		15		15
Disconnects		75		90
Canopy Jettison System	20	- -	25	- -
Miscellaneous	--	40	--	40
Total	370	700	485	805
Less Glass Weight	0	- 90	0	-110
Total Increment	370	610	485	695

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TABLE III

F-106-30 WEIGHT SUMMARY

	Canard Forward	Inlet Forward
Structure and Controls	12,029	12,224
Landing Gear	1,574	1,639
Propulsion Group	9,400	9,840
Equipment		
Instr.,Hyd.,Pneu.,Elec., and Aux. Gear	1,606	1,606
Electronics (Pulse Doppler)	3,875	3,875
Armament Provisions	210	210
Air Conditioning & Anti-Ice	726	726
Furnishings and Equipment	940	940
Weight Empty	30,360	31,060
Crew (2)	560	560
Oil and Unusable Fuel	290	290
Usable Fuel	15,000	15,000
Armament (2) GAR-9	1,690	1,690
Useful Load	17,540	17,540
GROSS WEIGHT	47,900	48,600

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TABLE IV

FIRE CONTROL SYSTEM CHARACTERISTICS

	Advanced MA-1	Pulse Doppler
Radar Type	Pulsed MOPA	Pulse Doppler
Transmitter Frequency	9000 ± 500 MC	9000 ± 500 MC
Average Power Output	700 Watts	5000 Watts
Antenna Size	40 Inches	40 Inches
Antenna Gain	37 db	37 db
Azimuth Scan Pattern	± 70°, ± 40°	± 70°, ± 40°
Elevation Scan Pattern	8°	8°
Scan Frame Time	8 sec., 5 sec.	8 sec., 5 sec.
Receiver Dynamic Range	110 db	110 db
Receiver Noise Figure	8 db	7 db
Detection Range (4)	46 Nautical Miles	100 Nautical Miles
Clutter Rejection	30 db	70 db
Navigation	Inertial	Inertial
Computer Type	Digital	Digital
Search Display	Range-Azimuth	Velocity-Azimuth
CCM Features	Frequency Hopping (1) Jittered PRF Silent Lobing Edge Tracking Jam Angle Track Maneuver Ranging	Frequency Hopping (2) Programmed PRF (3) Silent Lobing Velocity Tracking Jam Angle Track Maneuver Ranging

(1) Puke to pulse frequency hopping.

(2) Frequency hopping at intervals corresponding to 1/B where B is the doppler filter bandwidth.

(3) The PRF is programmed in 2 fixed patterns for range determination after lock-on.

(4) Detection Range on the nose of a B-47 type target with a M = 3.0 closing rate.

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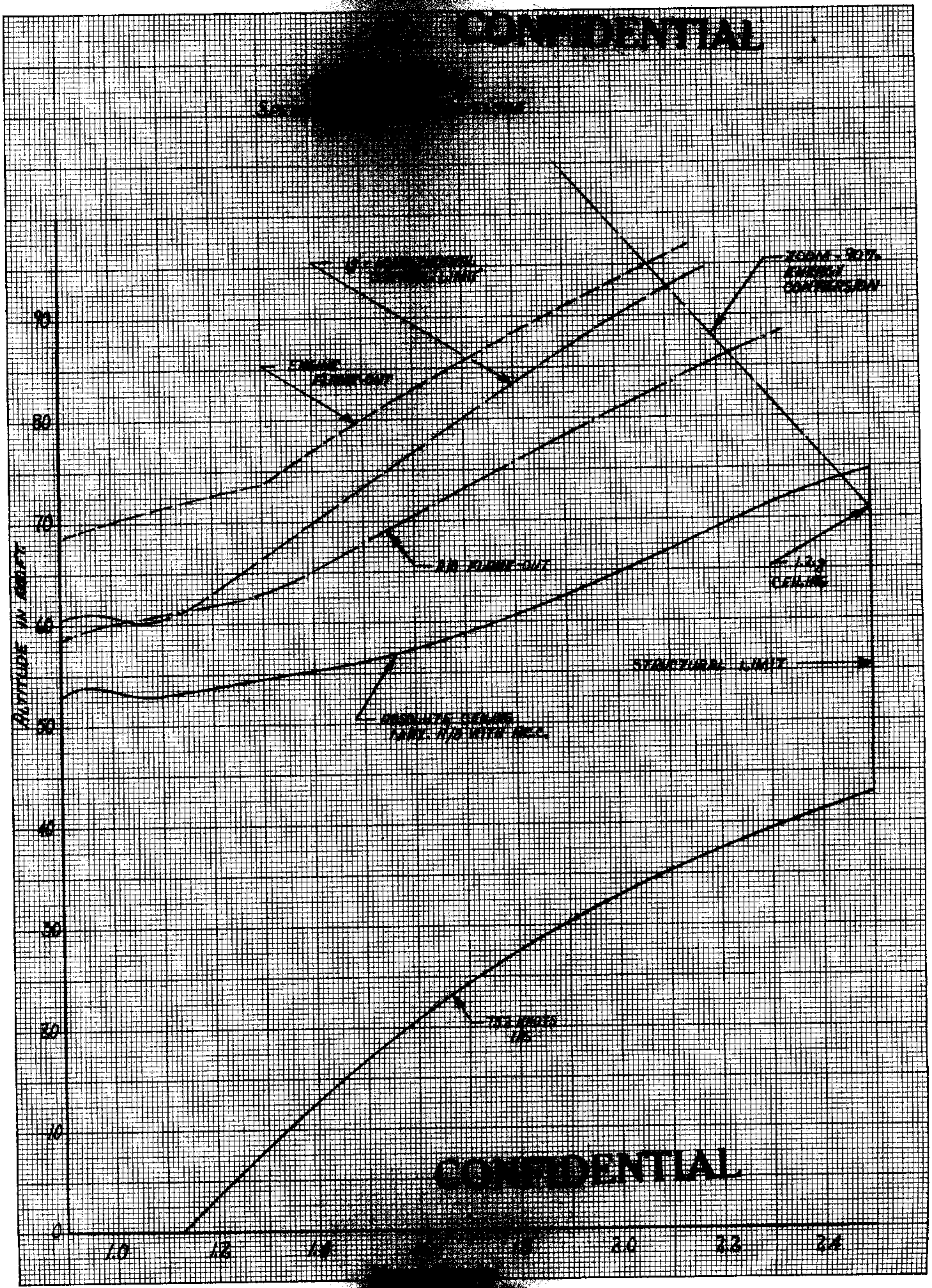

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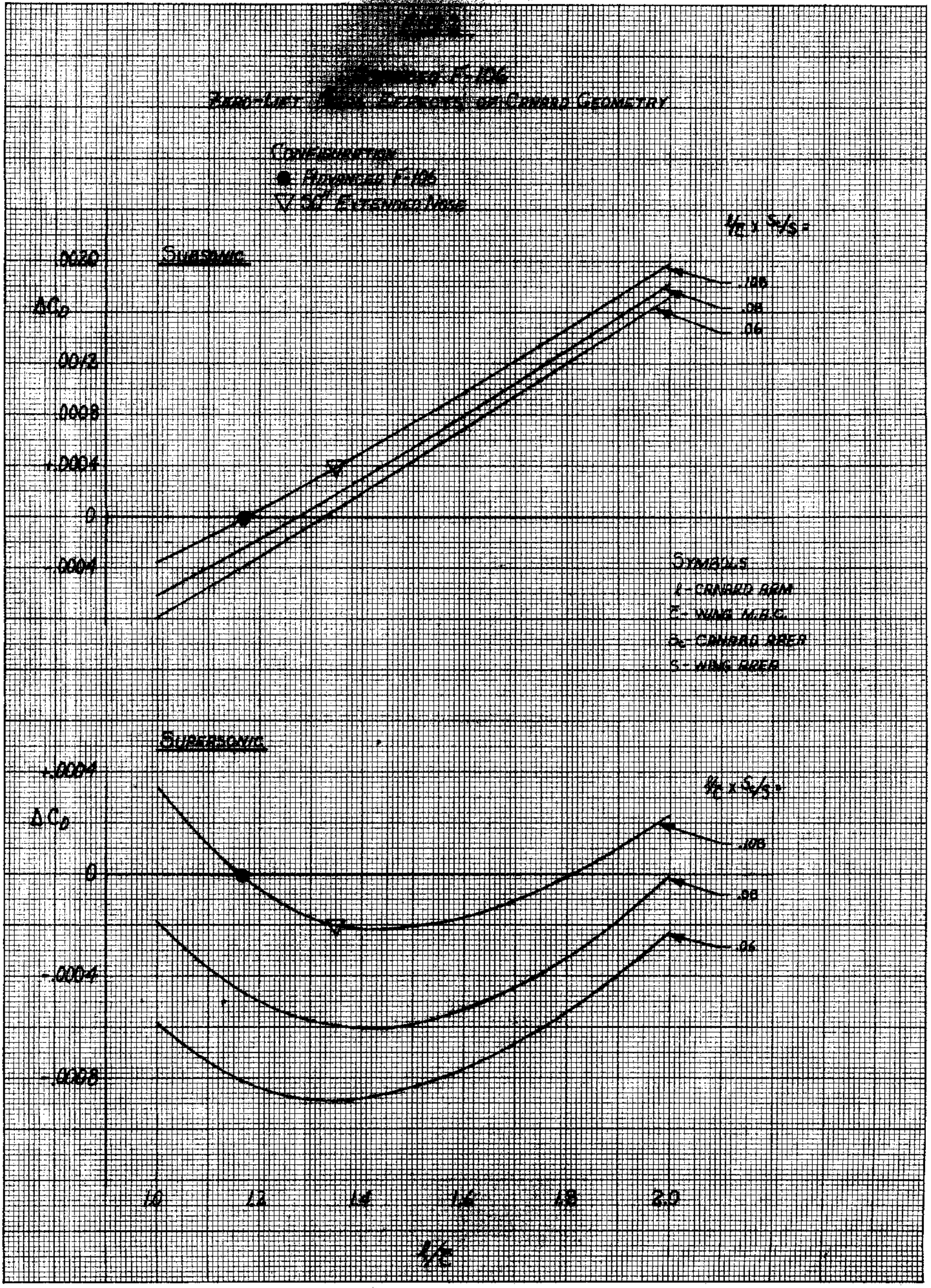
K+E 10 X 10 TO THE CM. 359-14 KEUFFEL & ESSER CO. MADE IN U.S.A.



K+E 10 X 10 TO THE CM. 359-14
KEUFFEL & ESSER CO. MADE IN U.S.A.

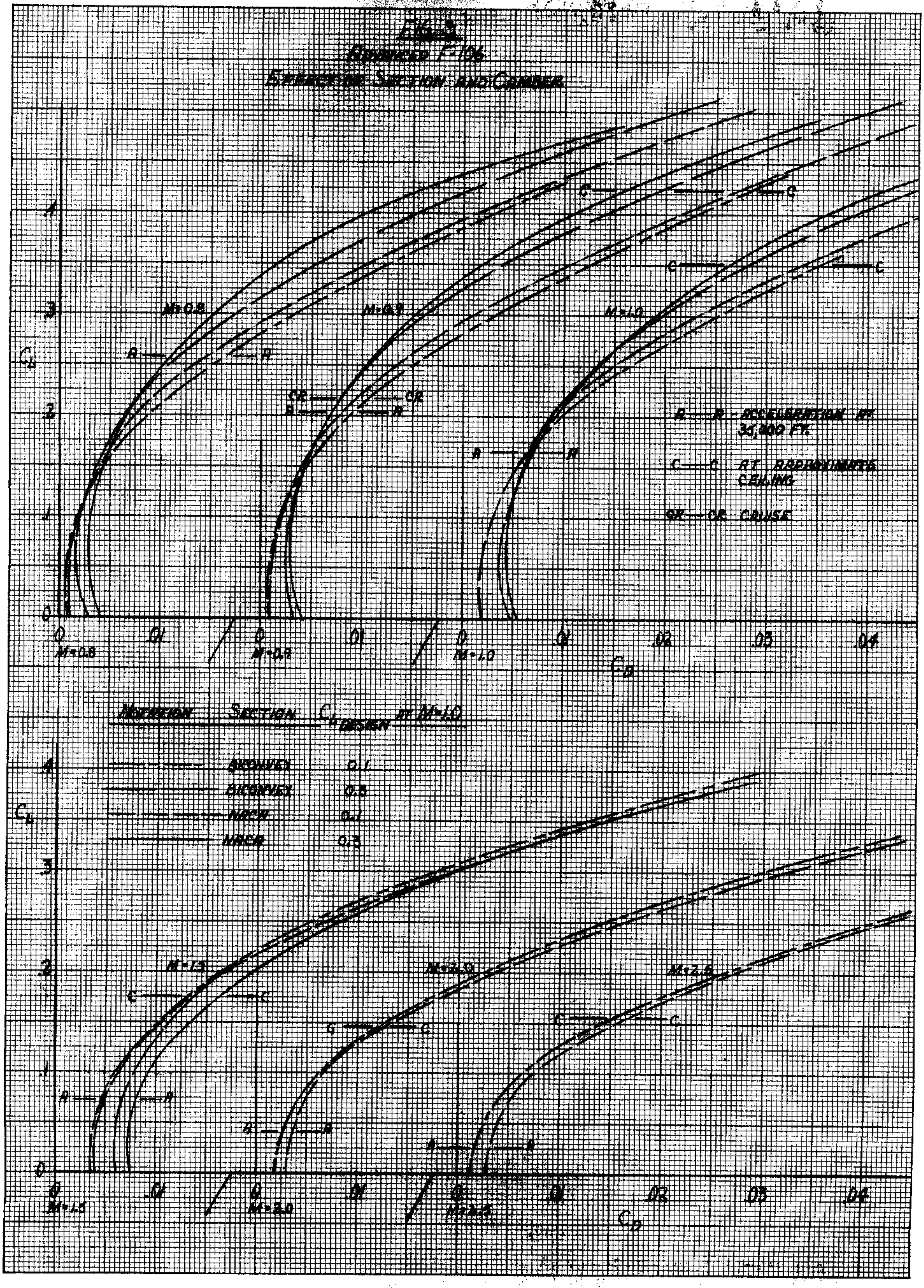


Fig. 3
Reduced F-106
Effect of Section and Camber

K✠E 10 X 10 TO THE CM. 359-14
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ADVANCED F-106

EFFECT OF MACH NO. ON LONGITUDINAL STABILITY
(WITH & WITHOUT 8.8% CANARD)

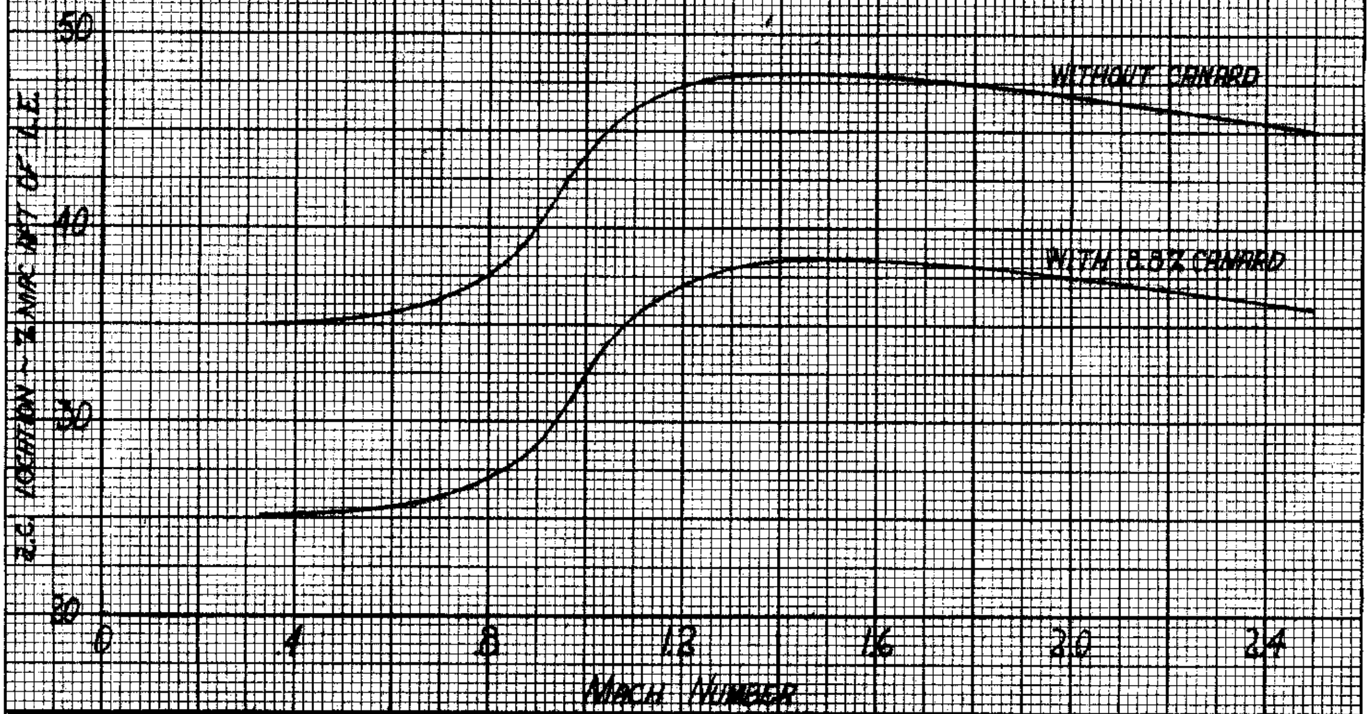
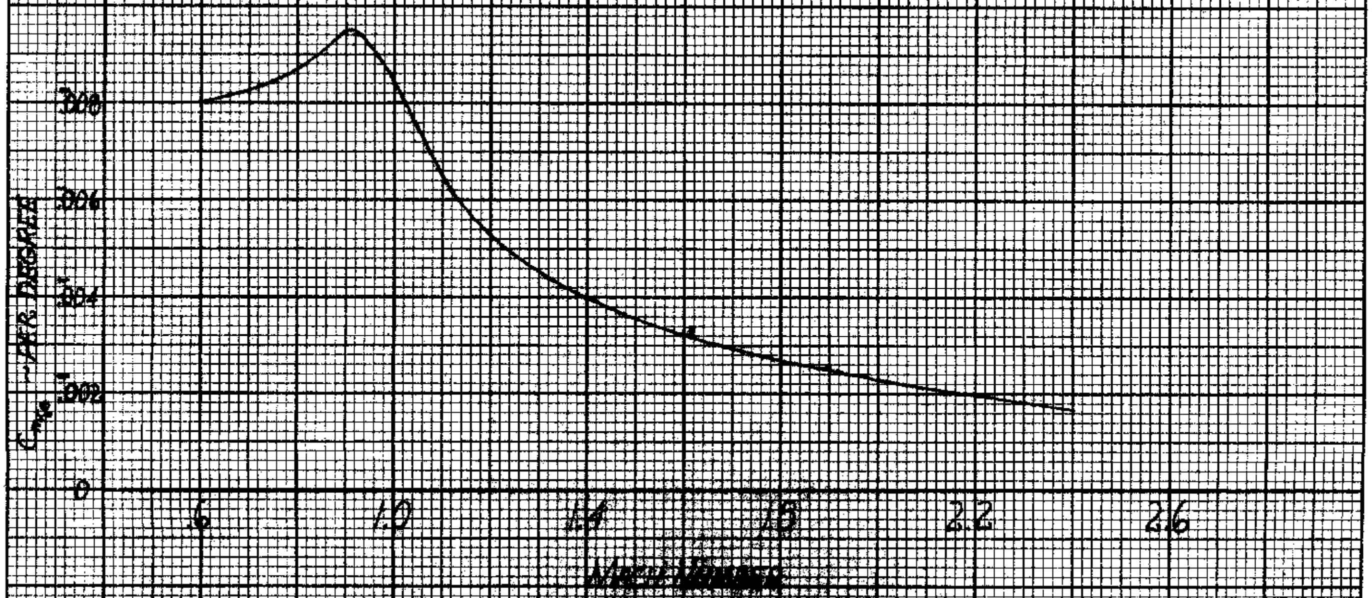


FIG. 4a

EFFECT OF MACH NO. ON ELEVATOR EFFECTIVENESS
(EXTRAPOLATED F-106B DATA; α=0)



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K+E 10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO. MADE IN U.S.A.

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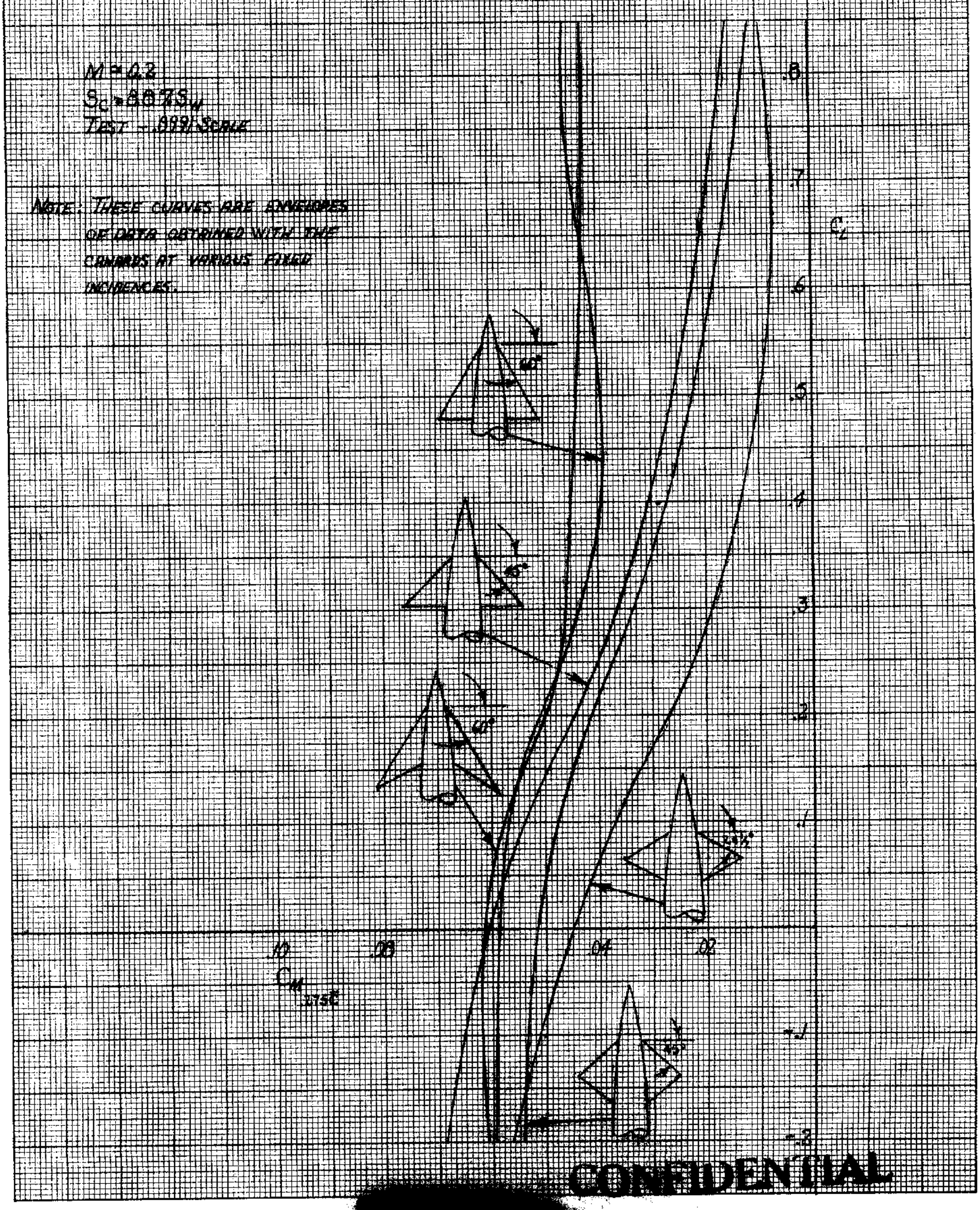
ADVANCED F-106

MAXIMUM PITCHING MOMENT COEFFICIENTS ATTAINABLE FOR CANARDS OF VARIOUS PLANFORMS



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K+E 10 X 10 TO THE CM. 359-14 KEUFFEL & ESSER CO. MADE IN U.S.A.

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ADVANCED F-106

COMPARISON OF ΔC_M VS ΔC_D FOR DEFLECTIONS OF CANARD ONLY & ELEVONS WITHOUT CANARD

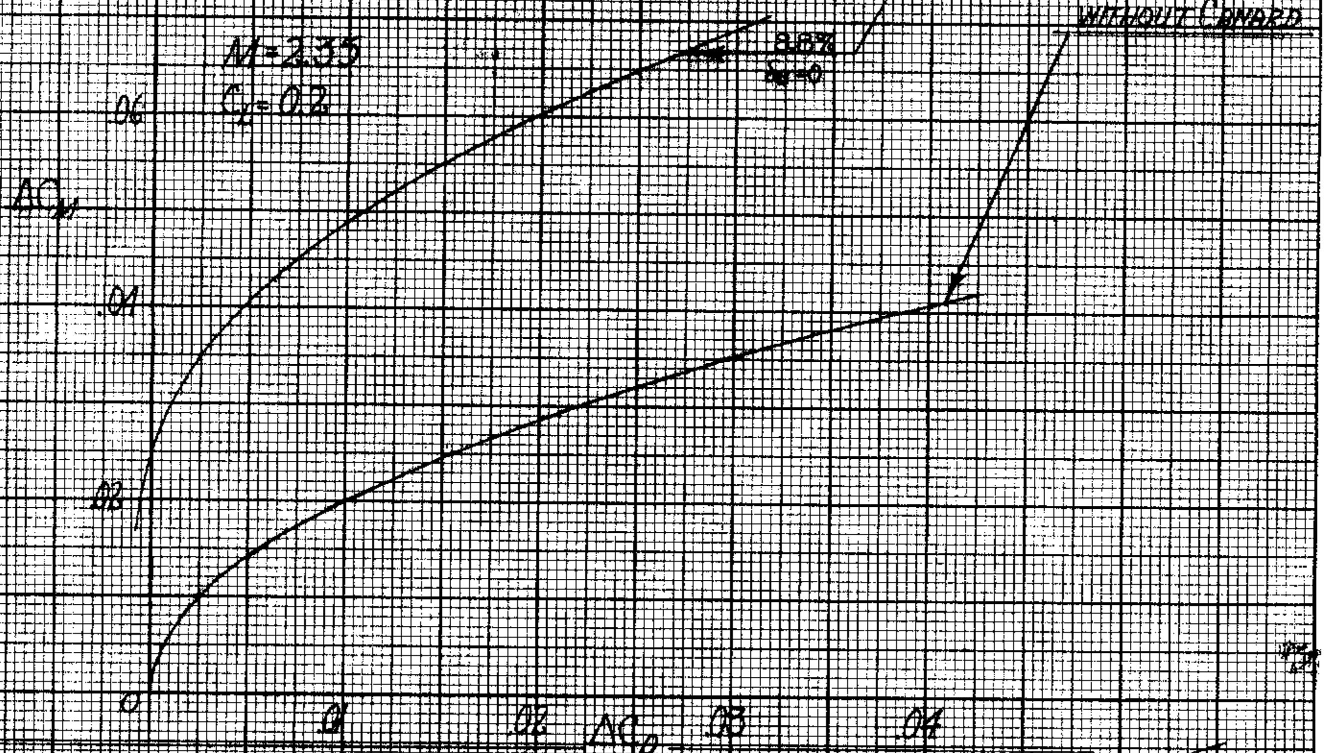
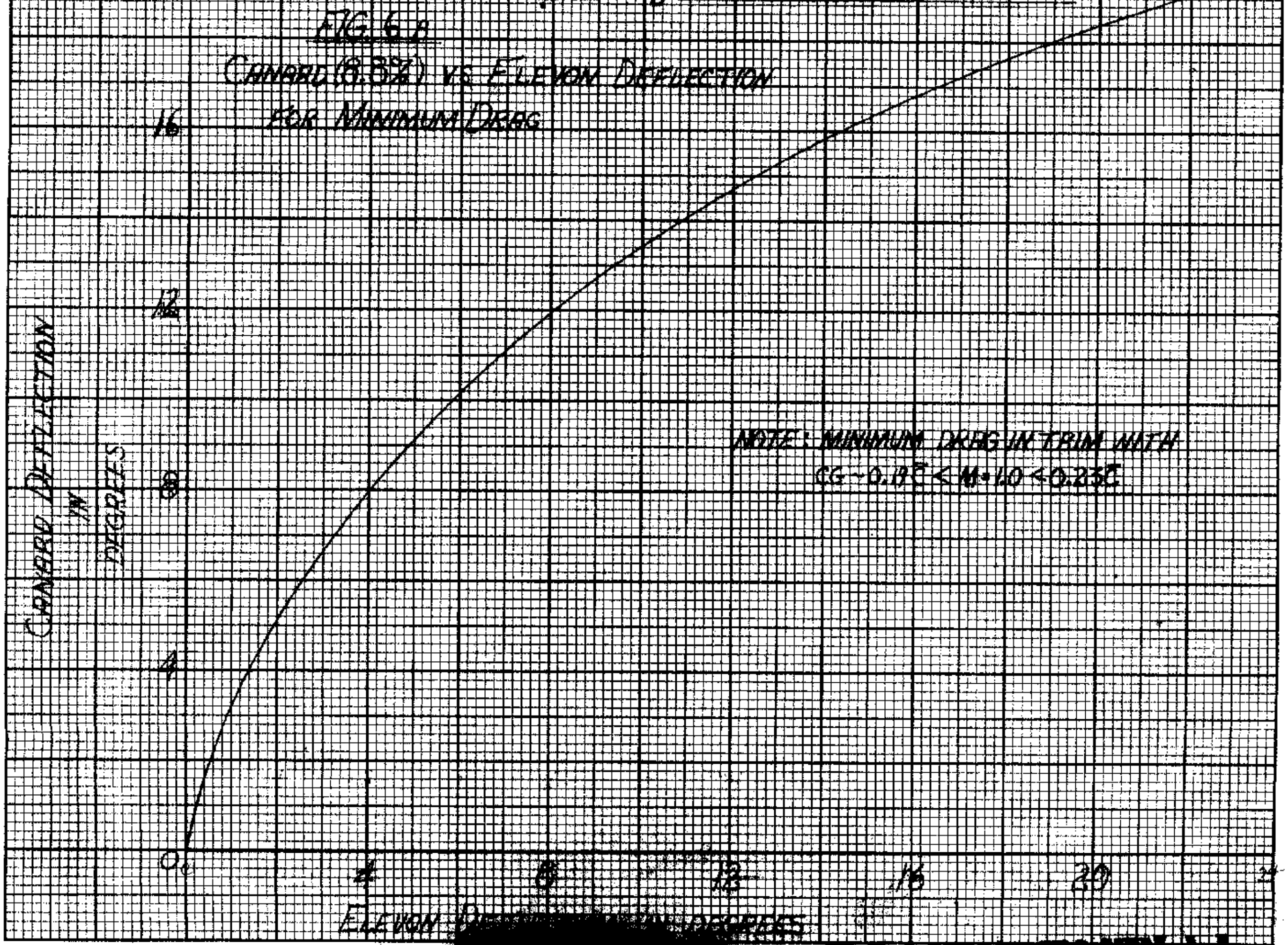


FIG. 6 B

CANARD (8.8%) VS ELEVON DEFLECTION FOR MINIMUM DRAG



NOTE: MINIMUM DRAG IN TRIM WITH
CG ~ 0.18 $\bar{C}$ < M=1.0 < 0.23 $\bar{C}$

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K+E 10 X 10 TO THE ½ INCH KEUFFEL & ESSER CO. 359-11 MADE IN U.S.A.

ADVANCED F-106

EFFECT OF CANARD SIZE ON MAX. LIFT COEFFICIENT

ANGLE OF ATTACK = 16°
STABILITY MARGIN = 0.03c̄
MACH NUMBER = 0.33

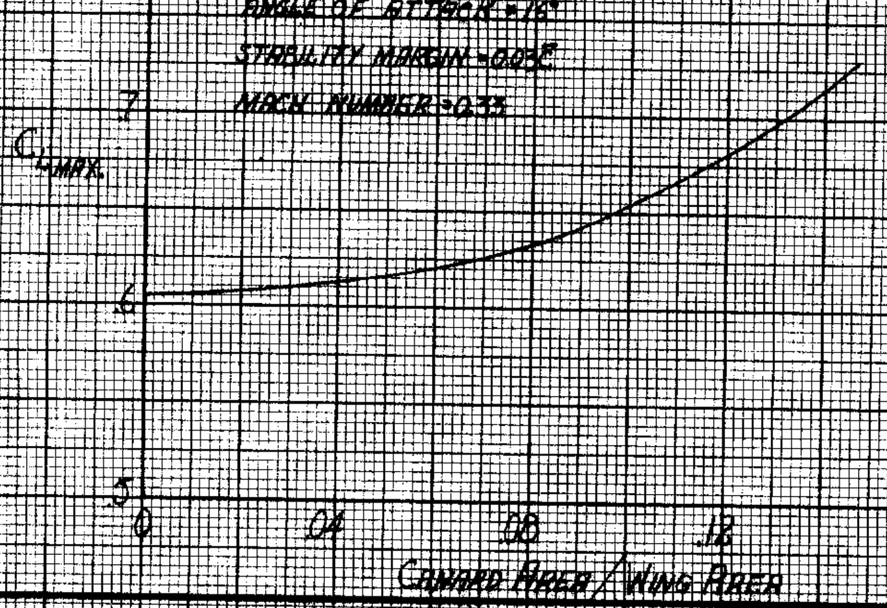
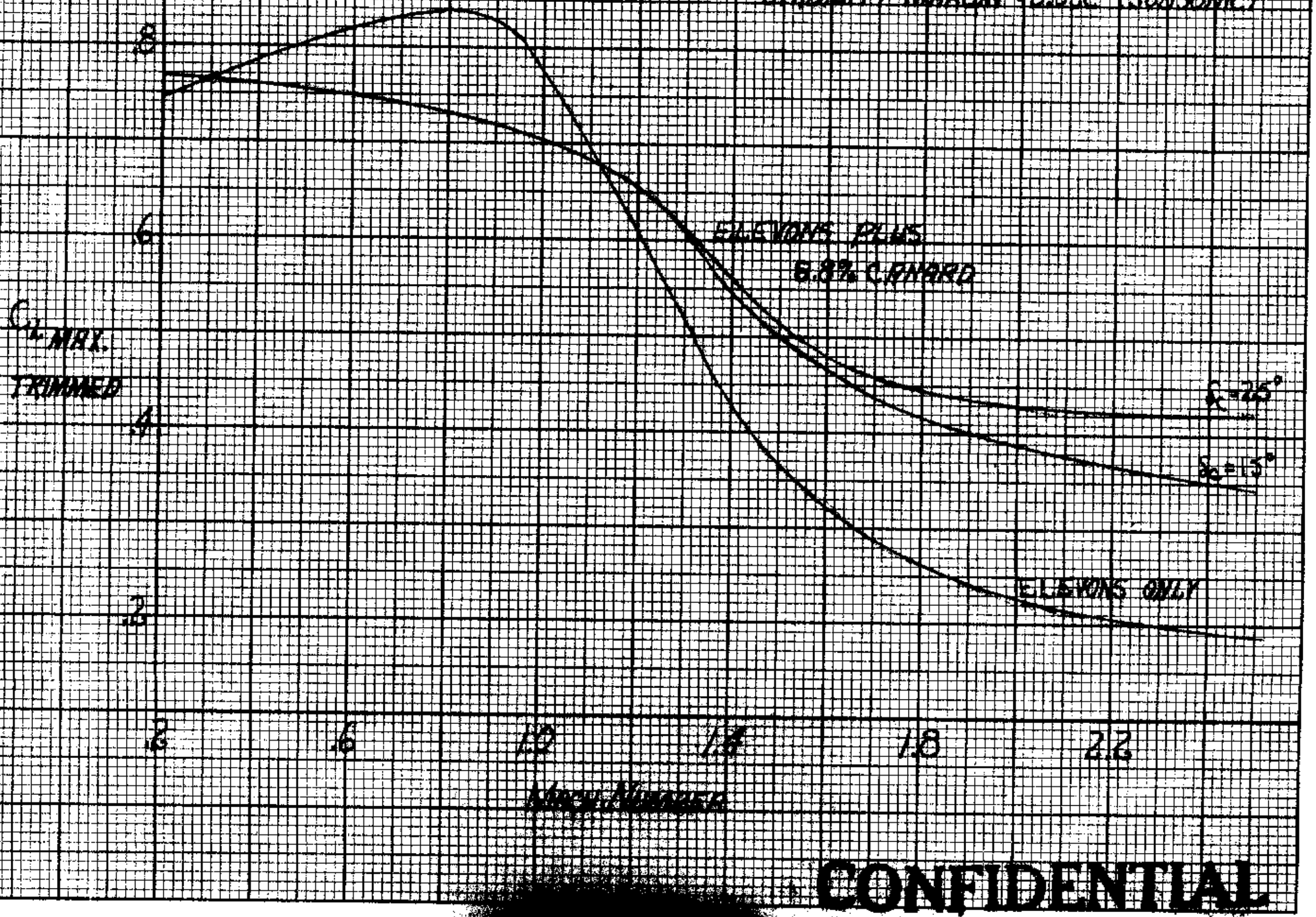


FIG. 7A

EFFECT OF CANARD ON MAXIMUM TRIMMED LIFT COEFFICIENT

STABILITY MARGIN = 0.03c̄ (SUBSONIC)



K+E 10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO. MADE IN U.S.A.

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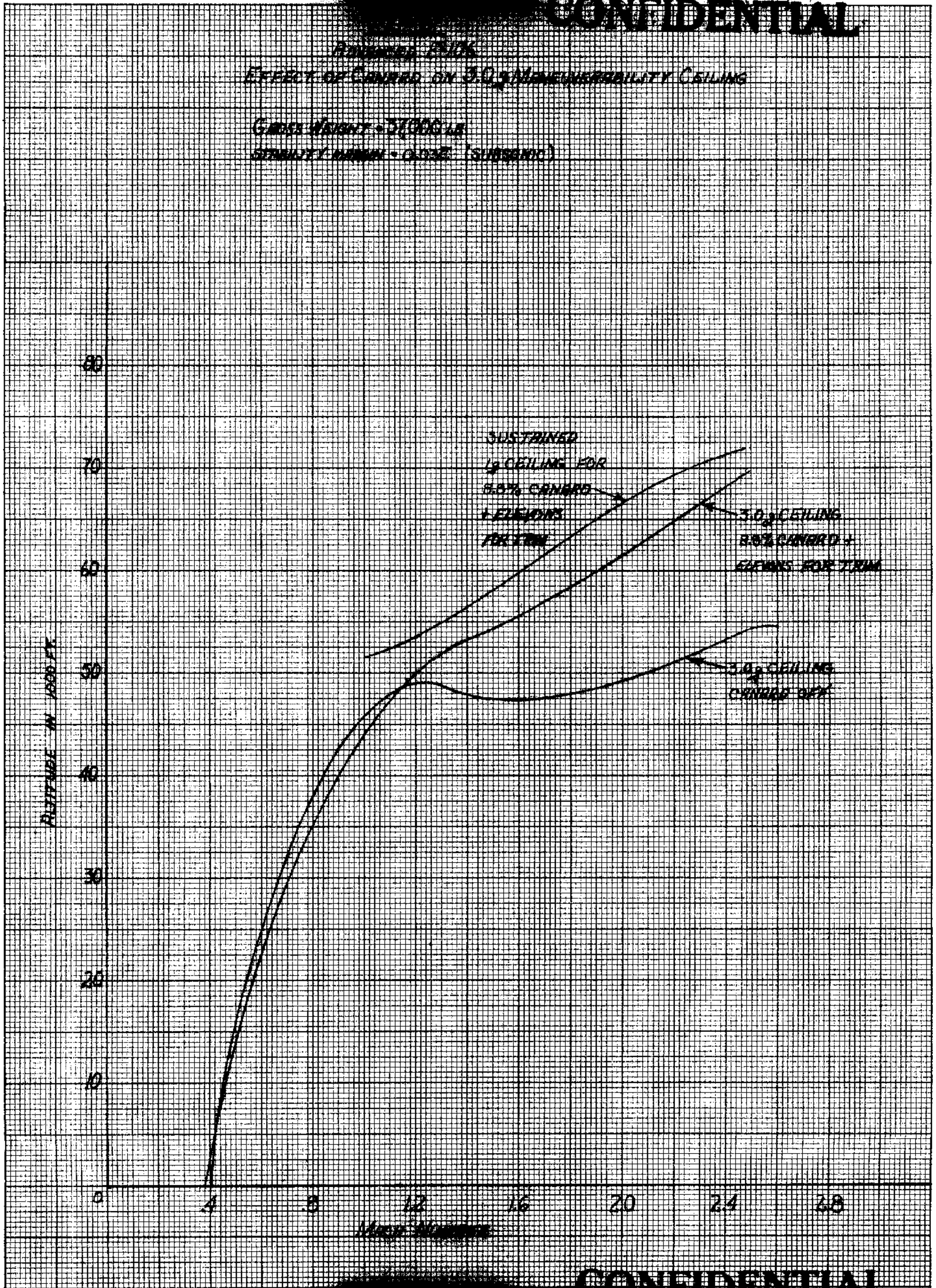

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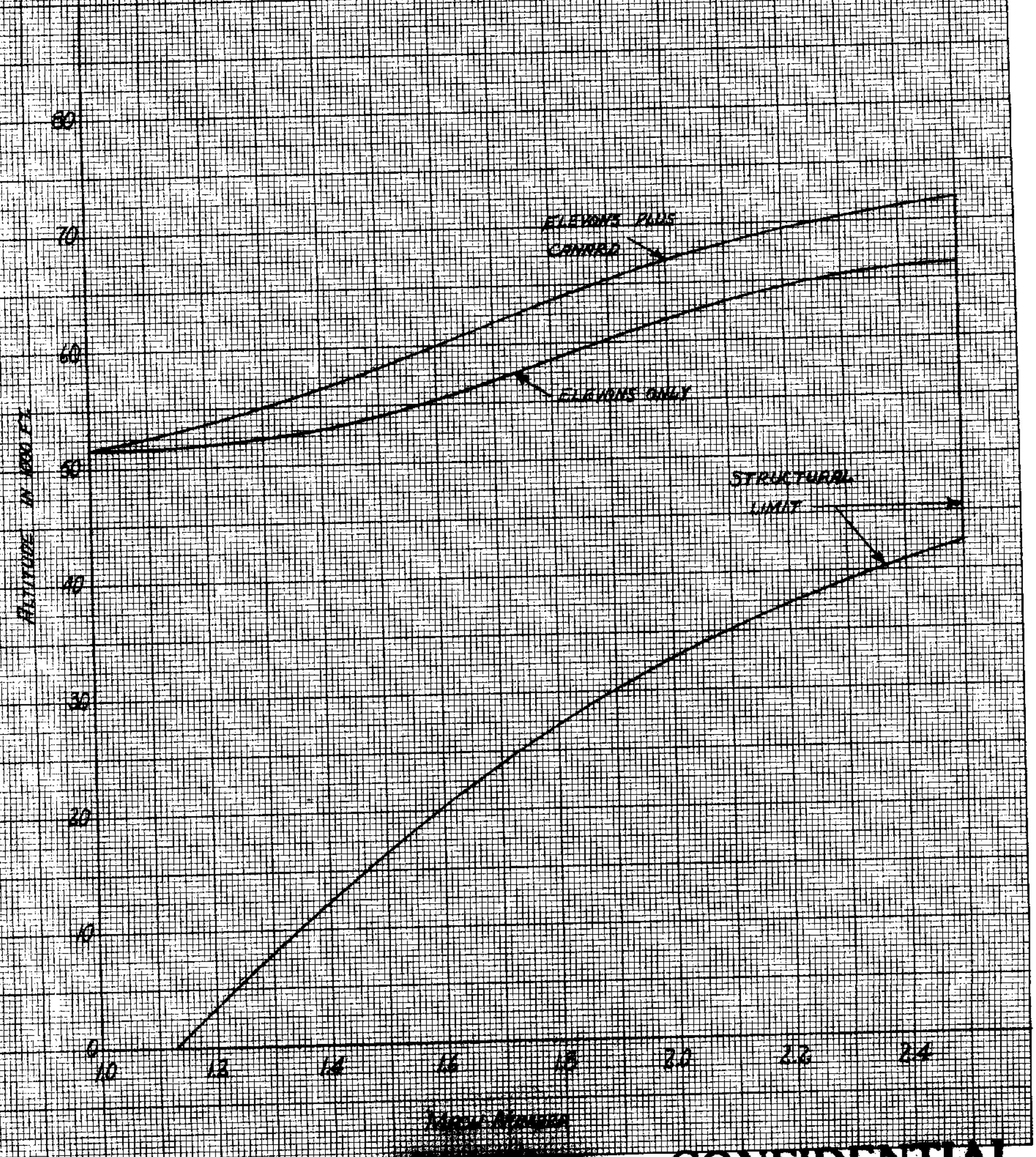
FIGURE

REPUBLIC F-105

EFFECT OF 8.8% CANARD ON SPEED ALTITUDE ENVELOPE

GROSS WEIGHT = 53,000 LB



K+E 10 X 10 TO THE CM. 359-14
KEUFFEL & ESSER CO. MADE IN U.S.A.

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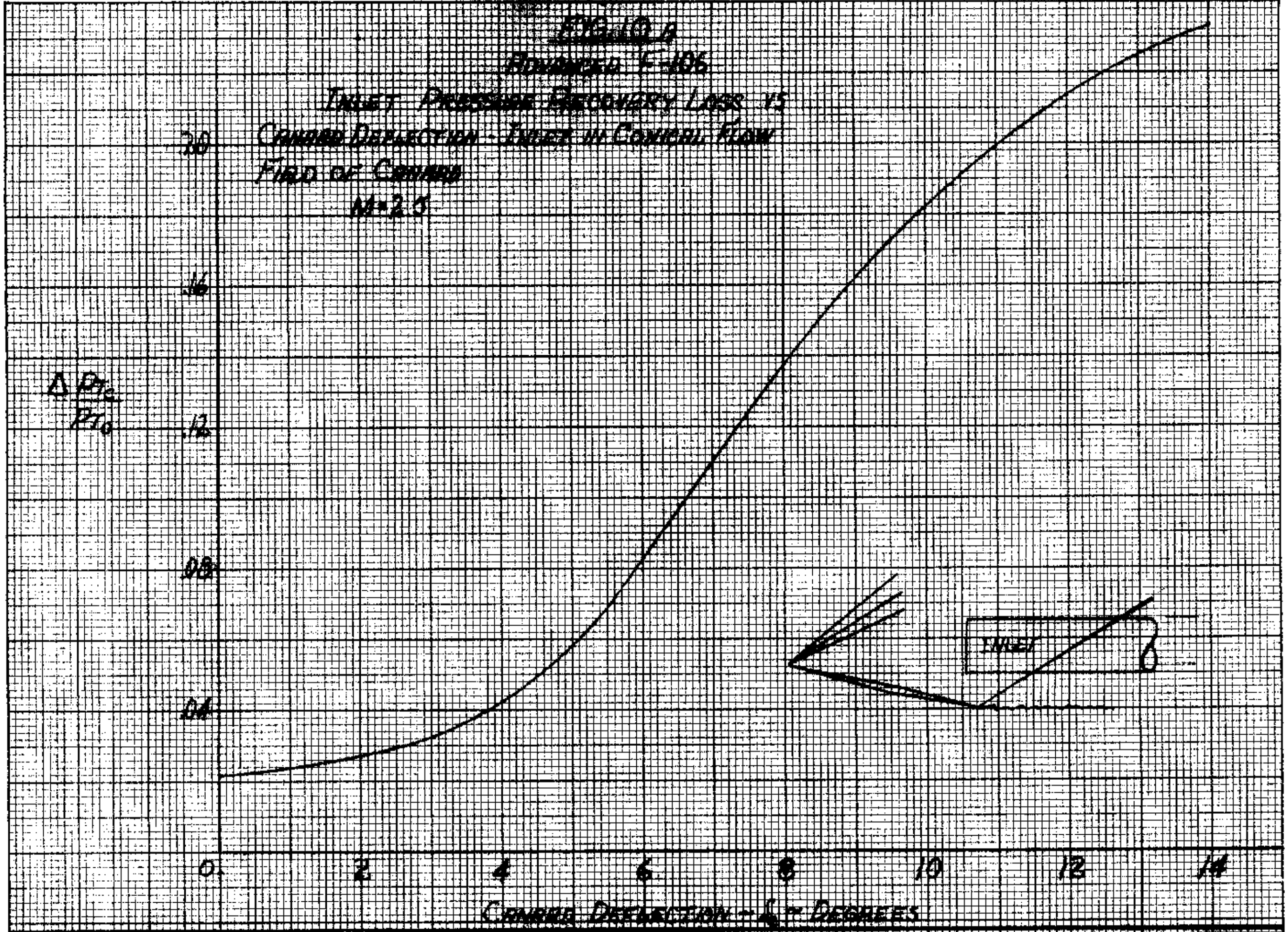


FIG. 10 A
ADVANCED F-106
INLET PRESSURE RECOVERY LOSS VS
CANARD DEFLECTION - INLET IN CONICAL FLOW
FIELD OF CANARD
M=2.5

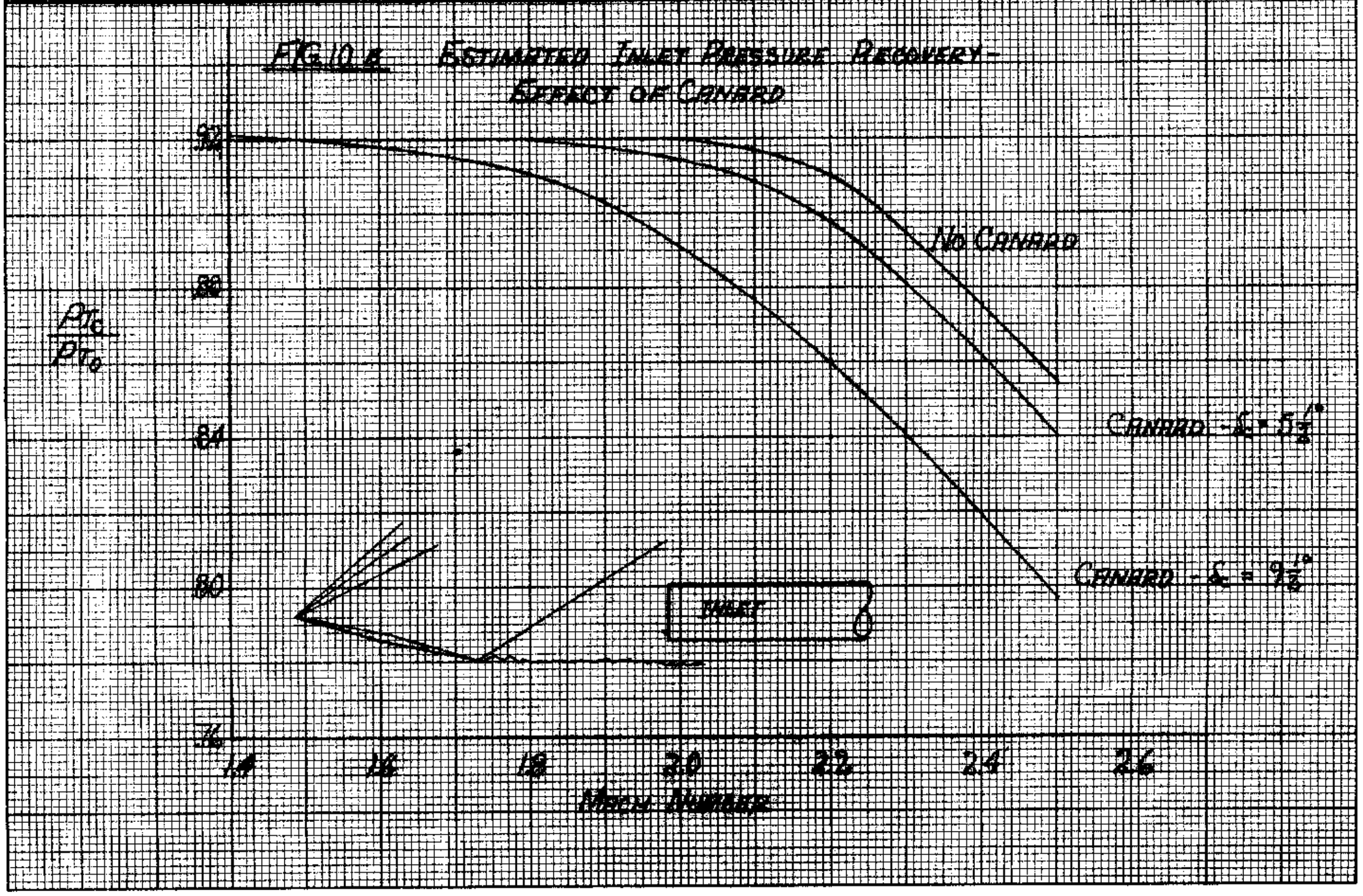
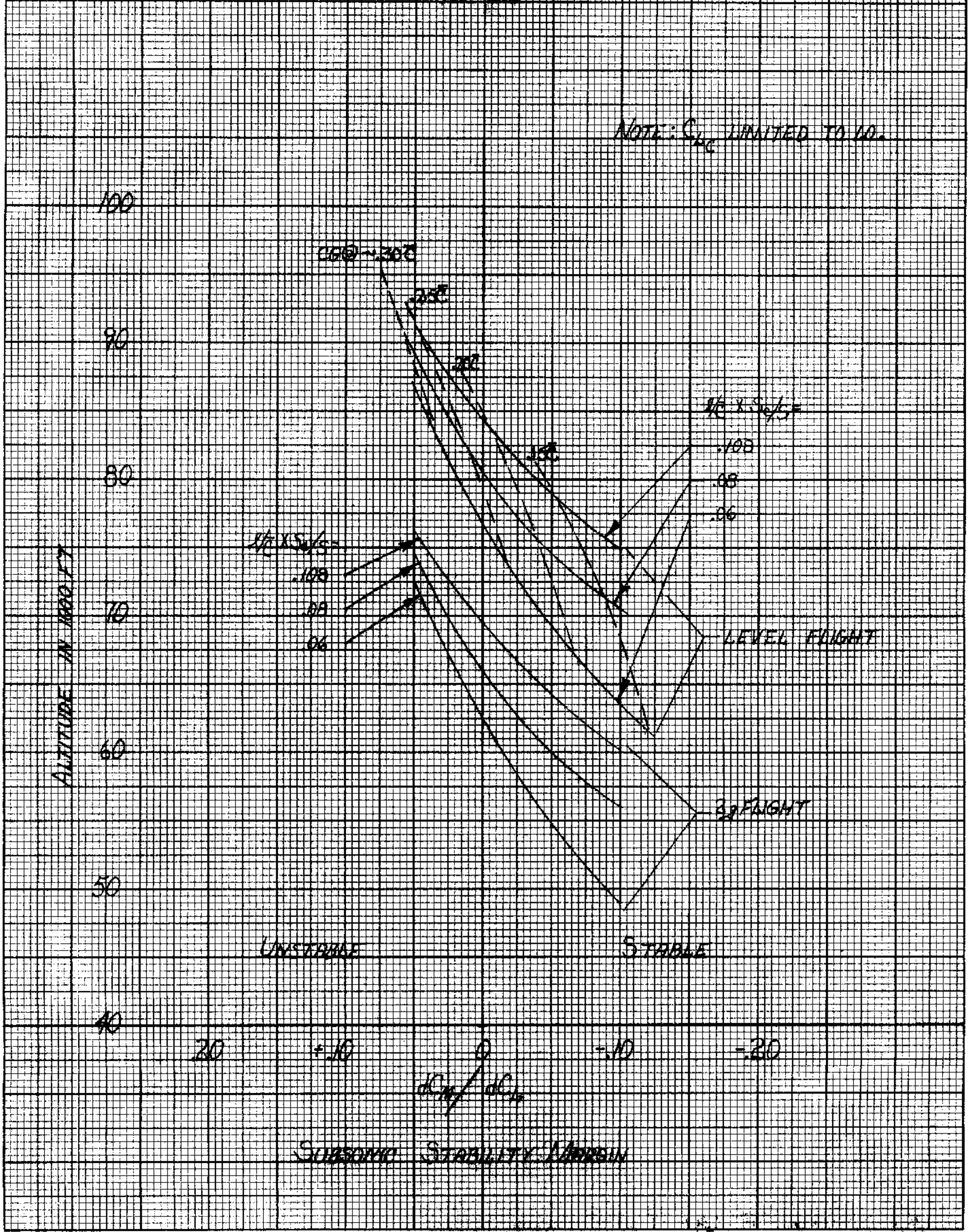


FIG 10 B ESTIMATED INLET PRESSURE RECOVERY - EFFECT OF CANARD

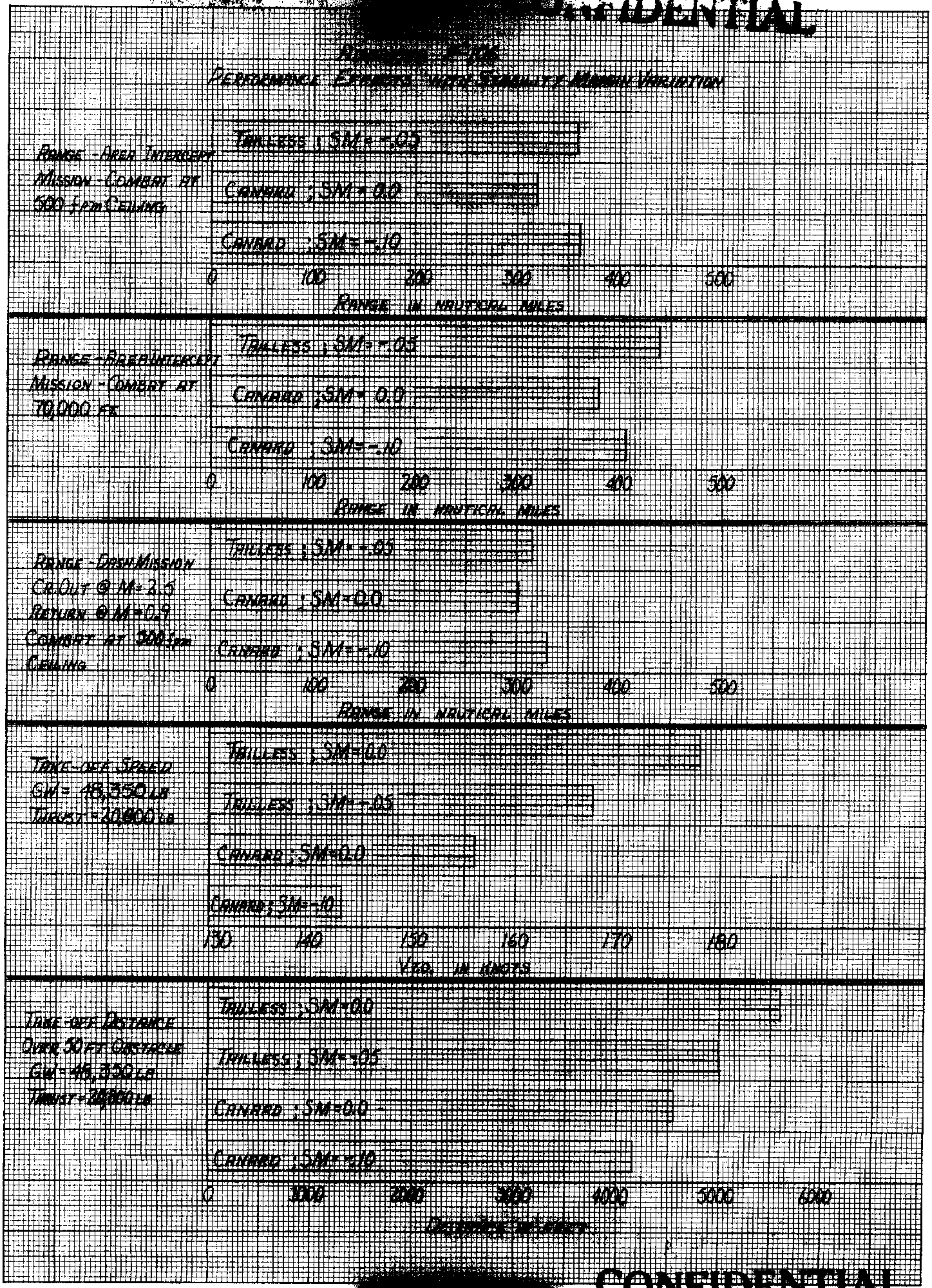
K+E 10 X 10 TO THE CM. 359-14 KEUFFEL & ESSER CO. MADE IN U.S.A.

FIG. 11
ADVANCED F-106
HIGH ALTITUDE MANEUVERABILITY
M=2.0



K+E 10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO. MADE IN U.S.A.

[redacted]DENTIAL



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[redacted]

10 X 10 TO THE CM. 359-14
KEUFFEL & ESSER CO. MADE IN U.S.A.

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FIG. 12 (CONT.)
ADVANCED F-106
PERFORMANCE EFFECTS WITH STABILITY MARGIN VARIATION

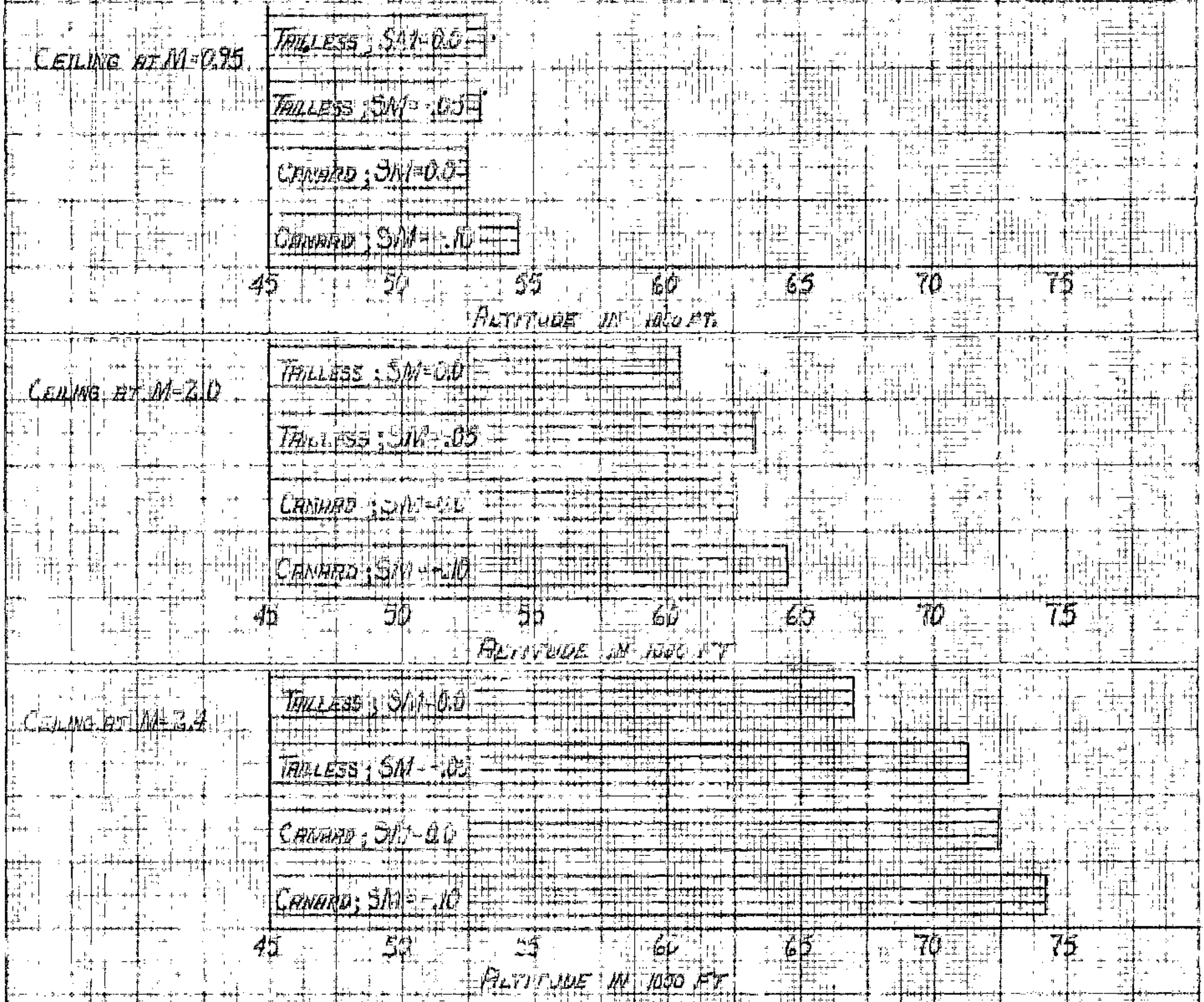

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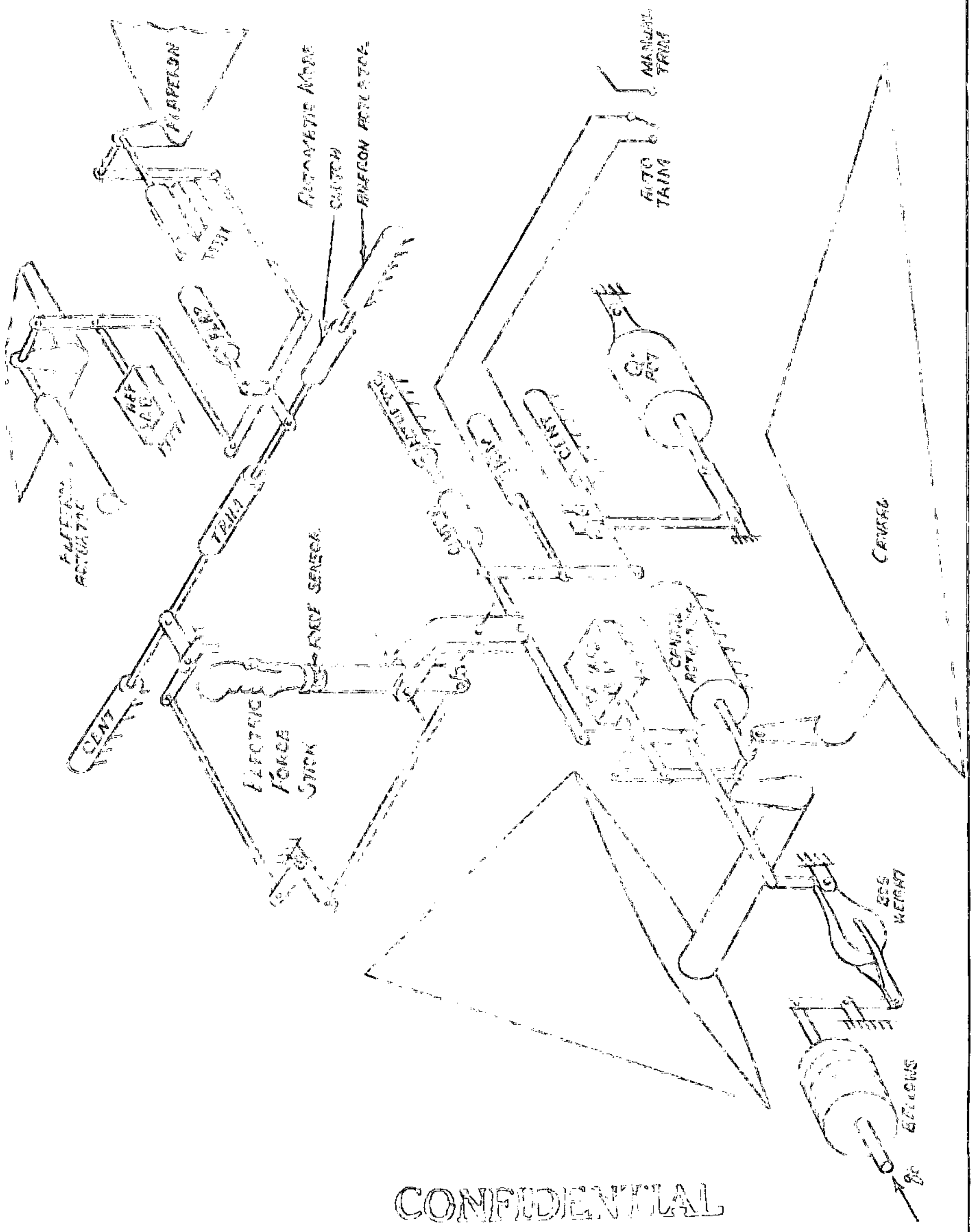
FIG. 13
ADVANCED F-106
CONTROL SYSTEM FUNCTIONAL SCHEMATIC



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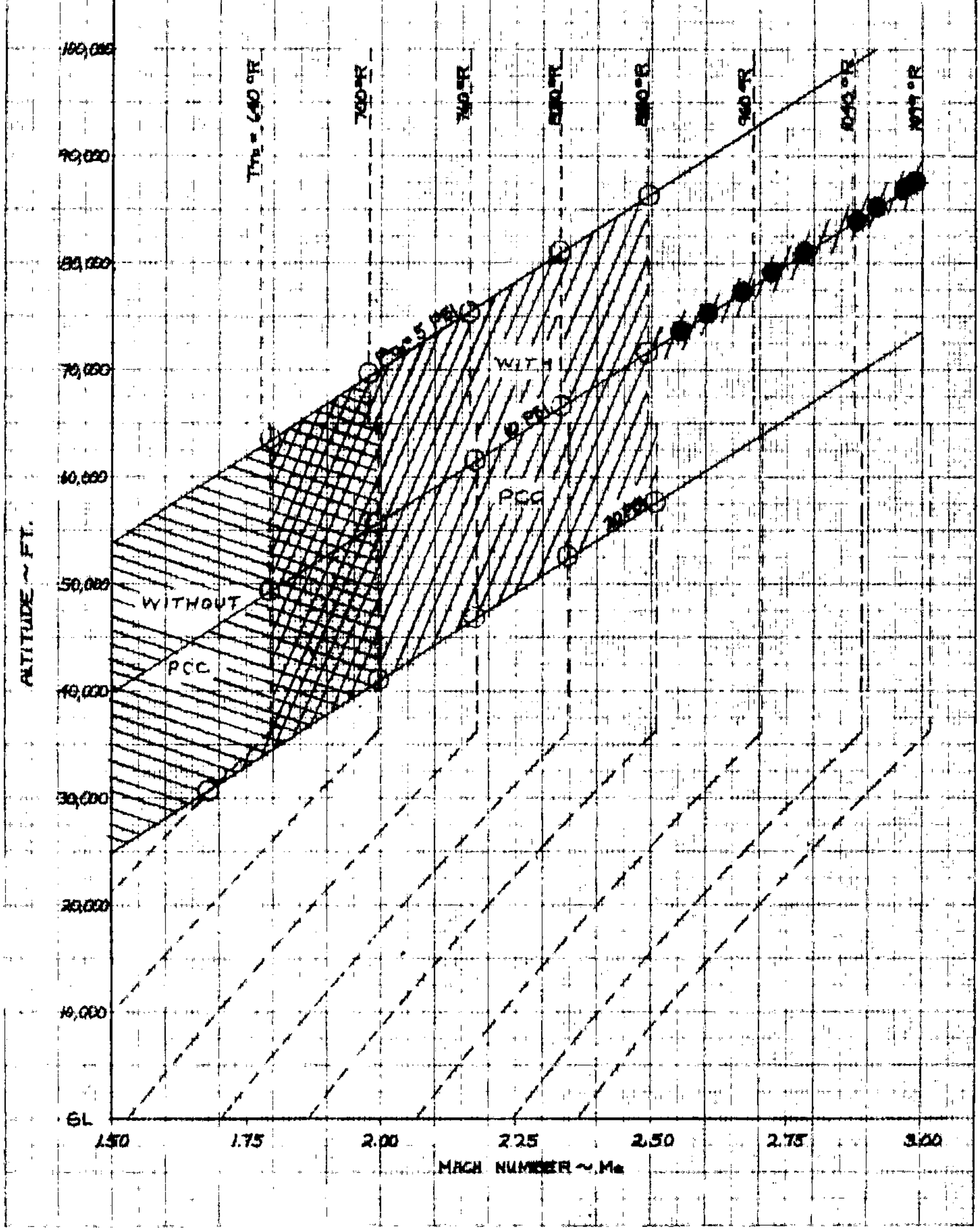
ALTITUDE VS MACH NUMBER
FOR CONSTANT FREE STREAM TOTAL TEMPERATURES AND CONSTANT FREE STREAM TOTAL PRESSURES

ICAO STANDARD DAY ATMOSPHERE FROM SL TO 65,000 FT.
NACA STANDARD DAY ATMOSPHERE FROM 65,000 FT. TO 100,000 FT.



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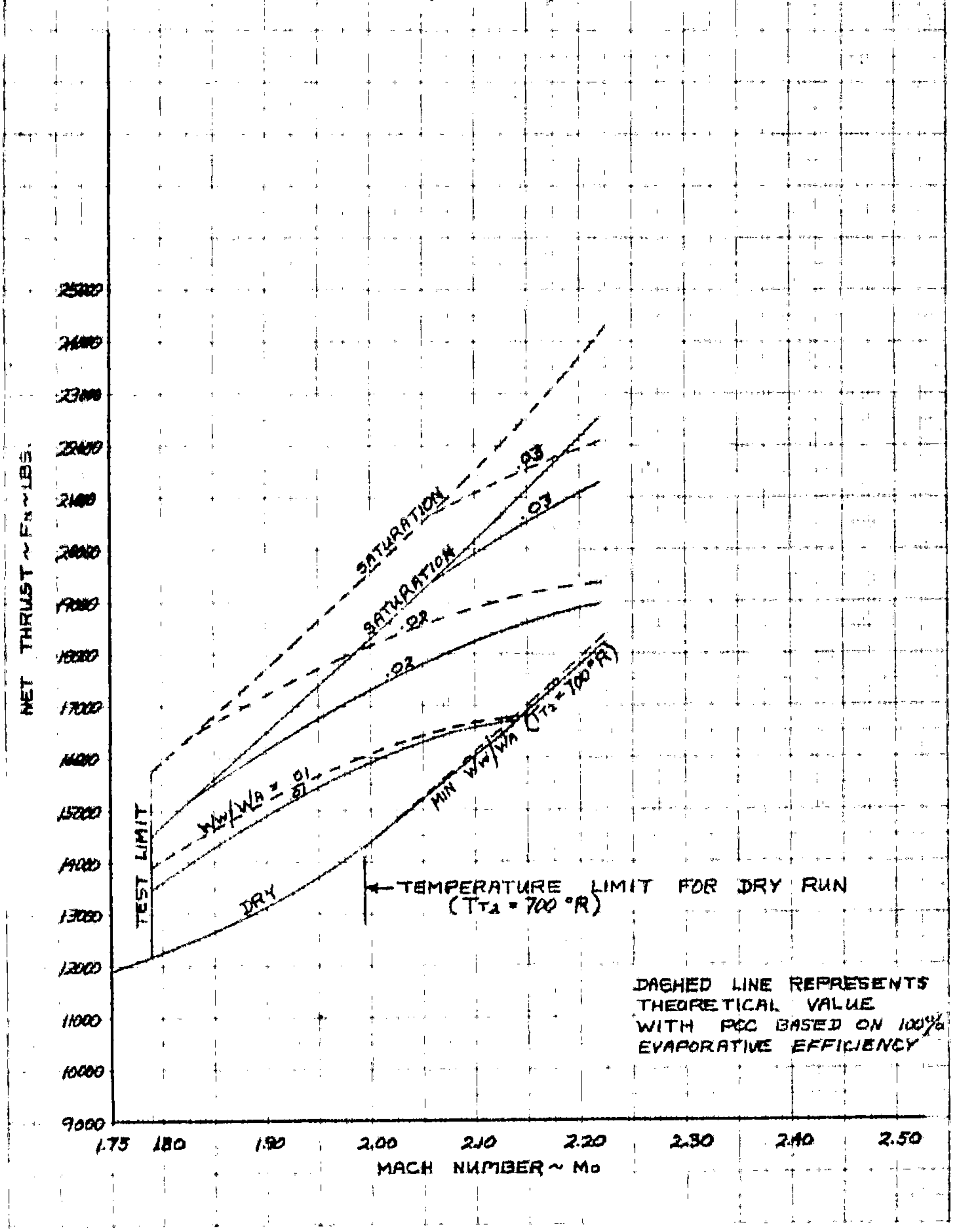
NET THRUST VS MACH NUMBER
YJ-75-P-3 P-610002, MAX /B, $P_{t_7} = P_{t_0}$
45,000 FT.



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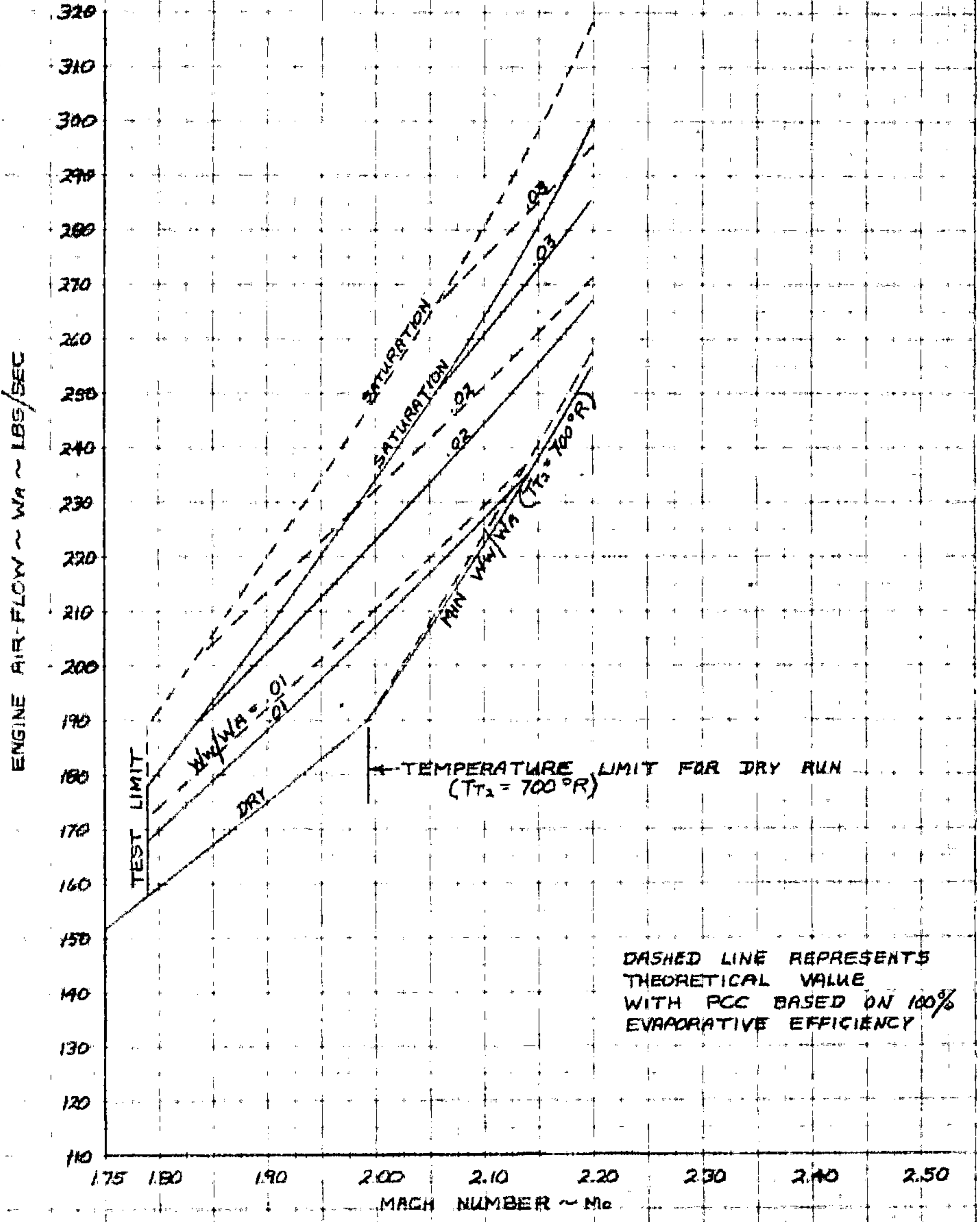
ENGINE AIR-FLOW VS. MACH NUMBER
YJ-75-P-3 P-610002, MAX AB, $P_{T1} = P_{T0}$

45,000 FT.



DASHED LINE REPRESENTS THEORETICAL VALUE WITH PCC BASED ON 100% EVAPORATIVE EFFICIENCY

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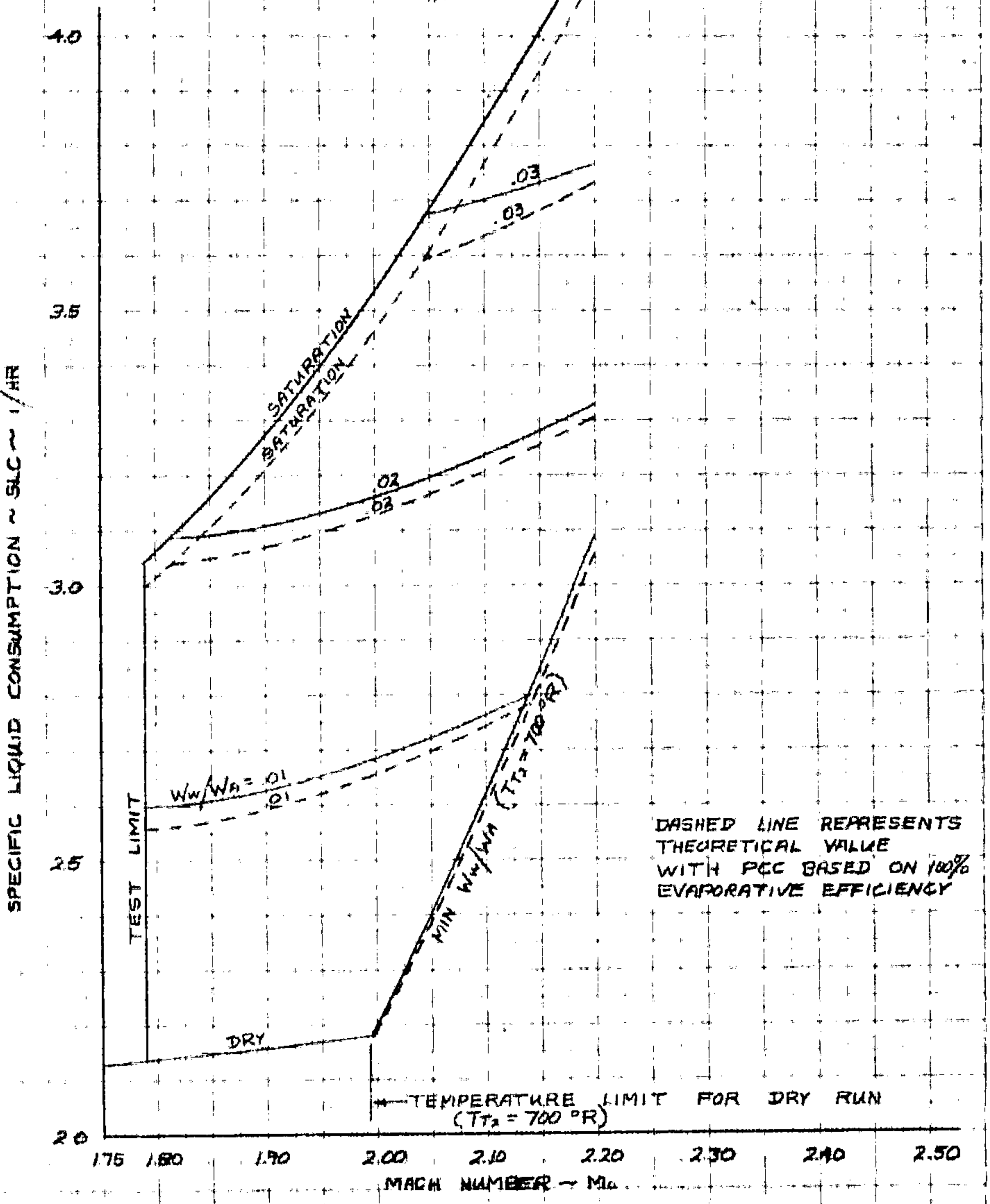
[illegible]

SPECIFIC LIQUID CONSUMPTION
VS
MACH NUMBER
YJ-75-P-3 P-610002, MAX AB, PT1 = PT0
45,000 FT



DASHED LINE REPRESENTS THEORETICAL VALUE WITH PCC BASED ON 100% EVAPORATIVE EFFICIENCY

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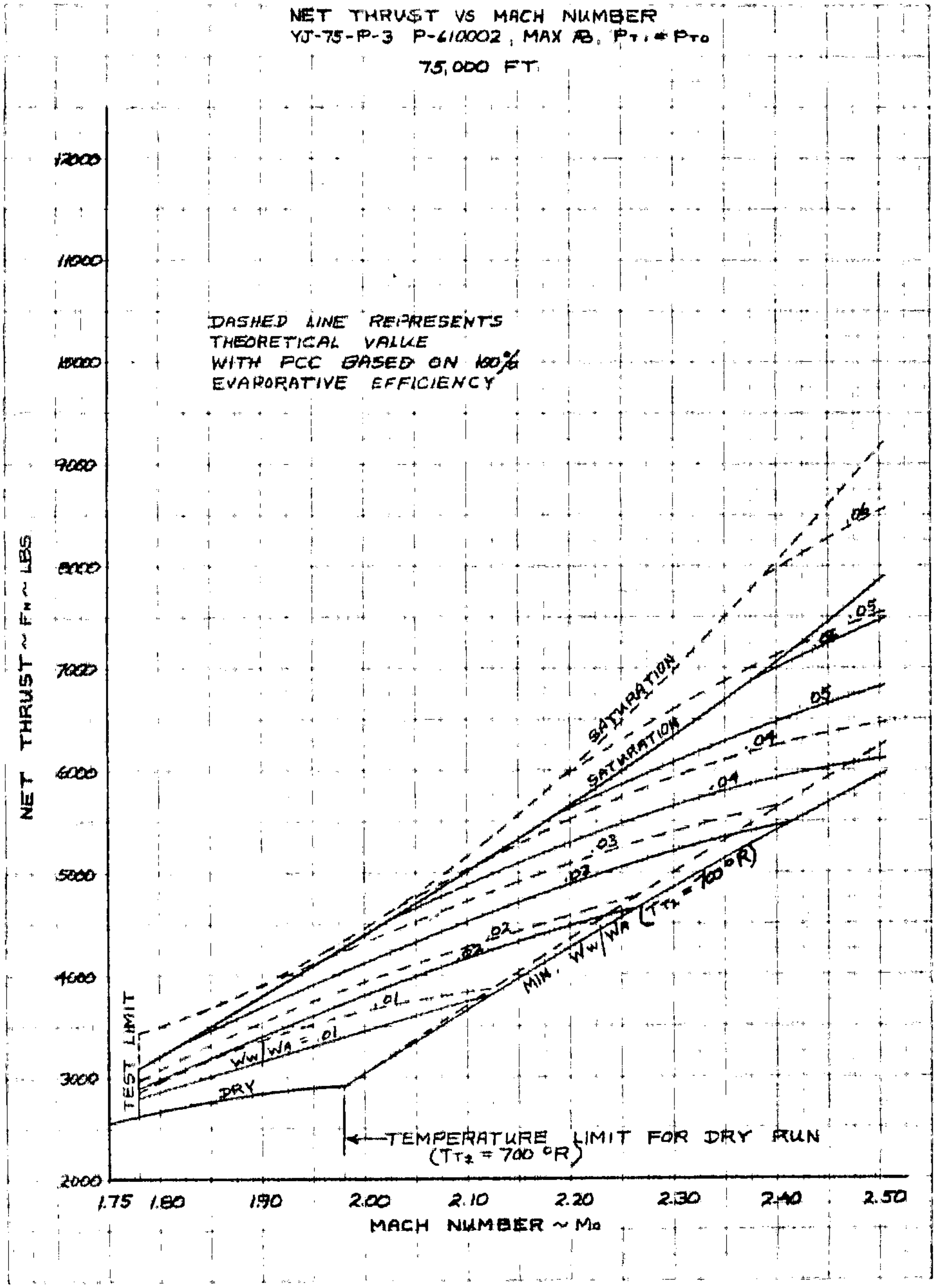

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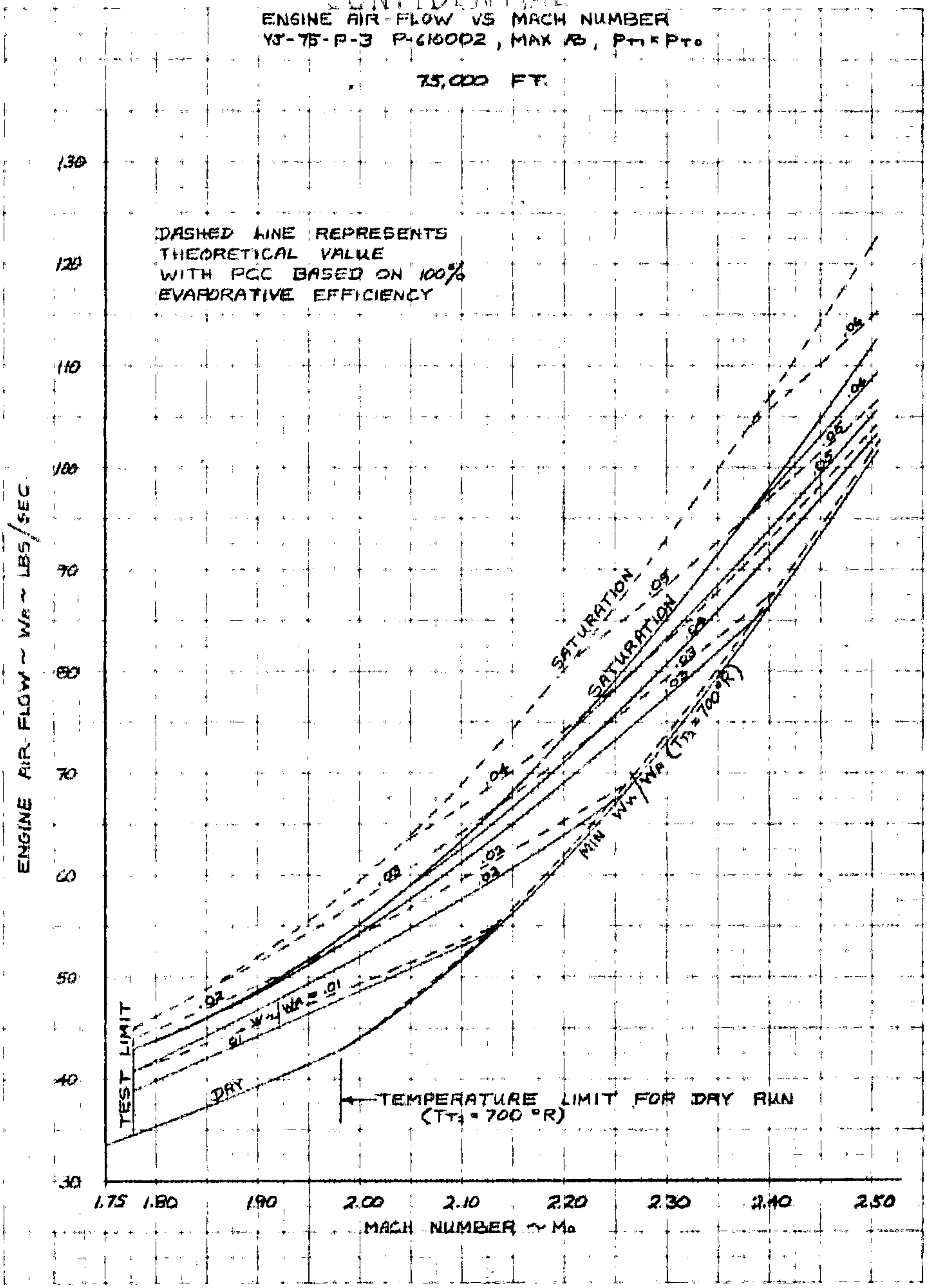

ENGINE AIR-FLOW VS MACH NUMBER
YJ-75-P-3 P-610002, MAX AB, $P_{T1} = P_{T0}$

75,000 FT.

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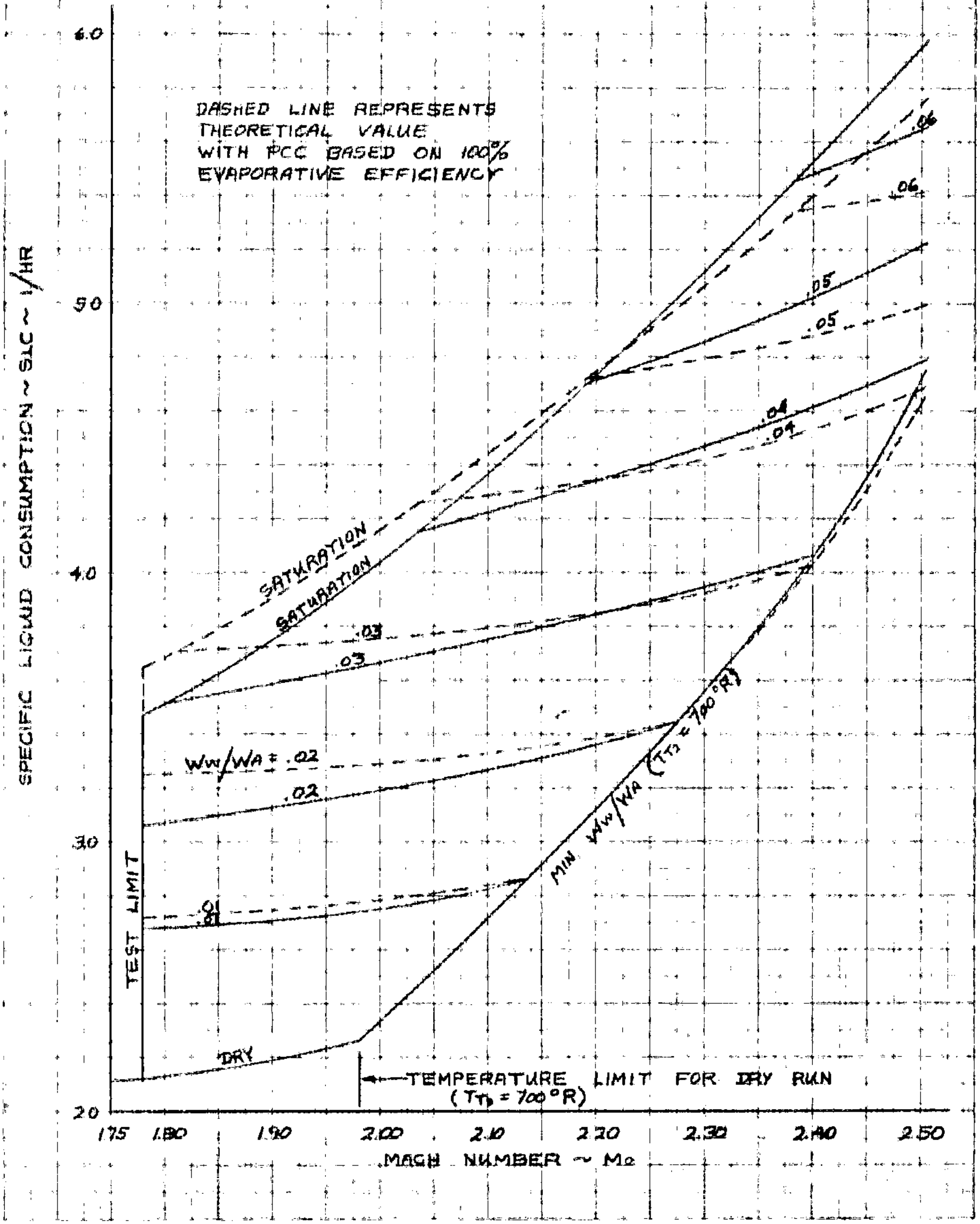
SPECIFIC LIQUID CONSUMPTION
VS
MACH NUMBER
YJ-75-P-3 P-610002, MAX AB, $P_{T1} = P_{T0}$
75,000 FT.



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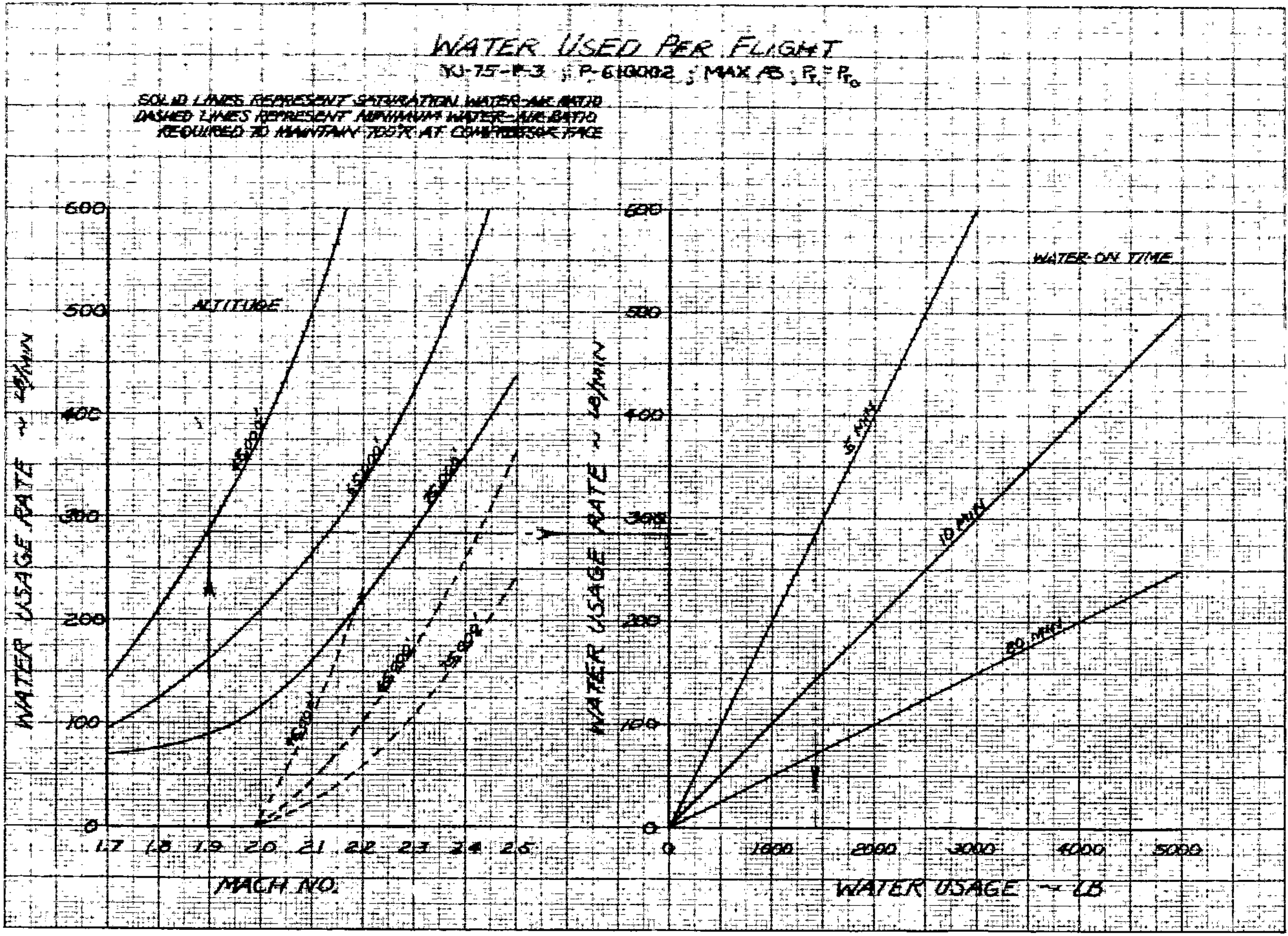
K+E 10 X 10 TO THE CM. 359T-14G
KEUFFEL & ESSER CO. MADE IN U.S.A.
ALBANENE



WATER USED PER FLIGHT

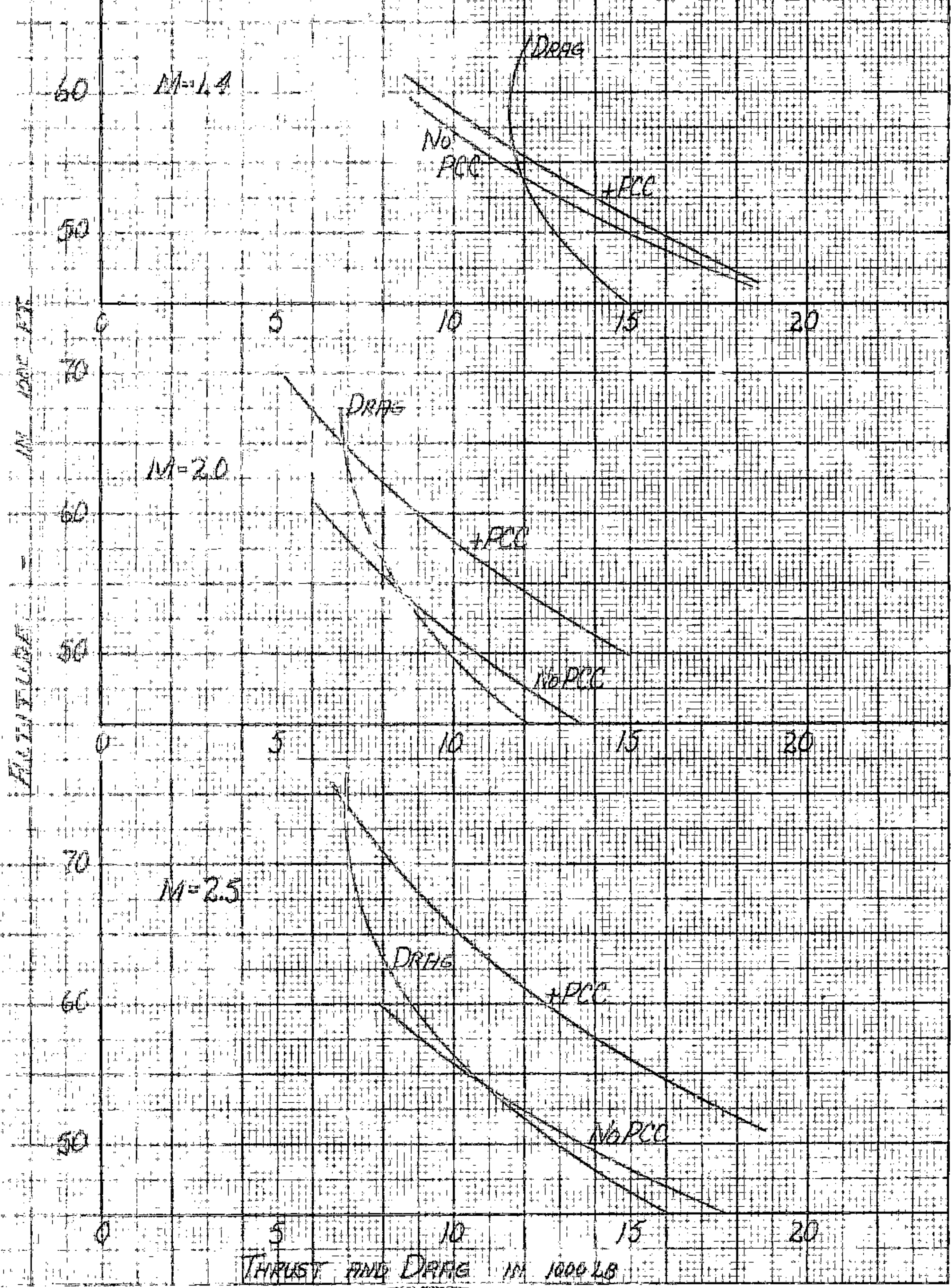
YJ-75-P-3 ; P-610002 ; MAX AB ; $P_T = P_{T_0}$

SOLID LINES REPRESENT SATURATION WATER-AIR RATIO
DASHED LINES REPRESENT MINIMUM WATER-AIR RATIO
REQUIRED TO MAINTAIN 700°R AT COMPRESSOR FACE

FIG. 2.11

ADVANCED F-106

EFFECT OF PRE-COMPRESSOR COOLING



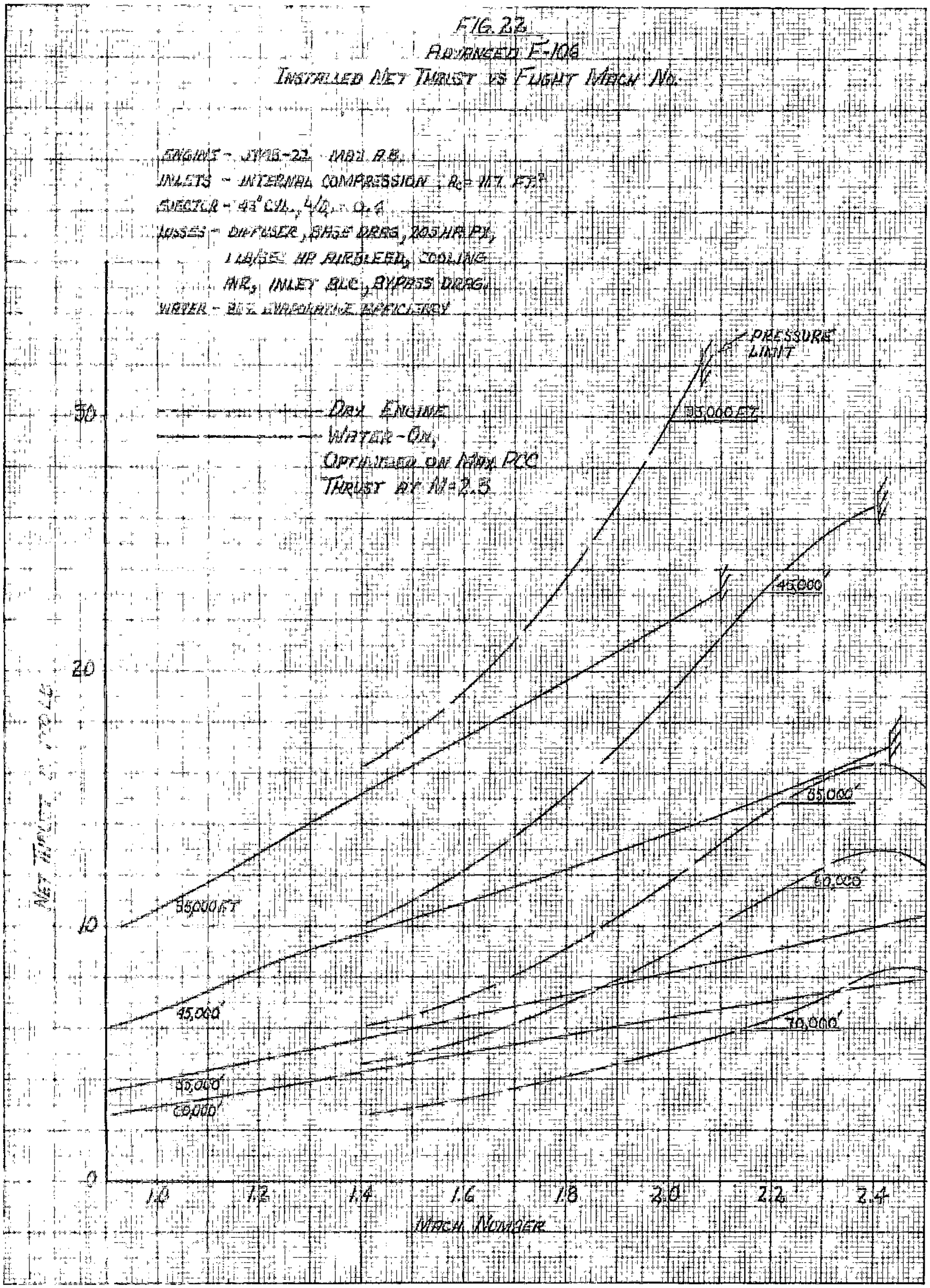


FIG. 22
ADVANCED F-106
INSTALLED NET THRUST VS FLIGHT MACH NO.

ENGINE - J75-22 MOD A.B.
INLETS - INTERNAL COMPRESSION; A_c = 11.7 FT.^{2}
EJECTOR - 43° CYL., L/D = 0.4
LOSSES - DIFFUSER, BASE DRAG, 205 HP P.X., 1 LB/SEC HP AIRBLEED, COOLING AIR, INLET BLC, BYPASS DRAG.
WATER - 85% EVAPORATIVE EFFICIENCY

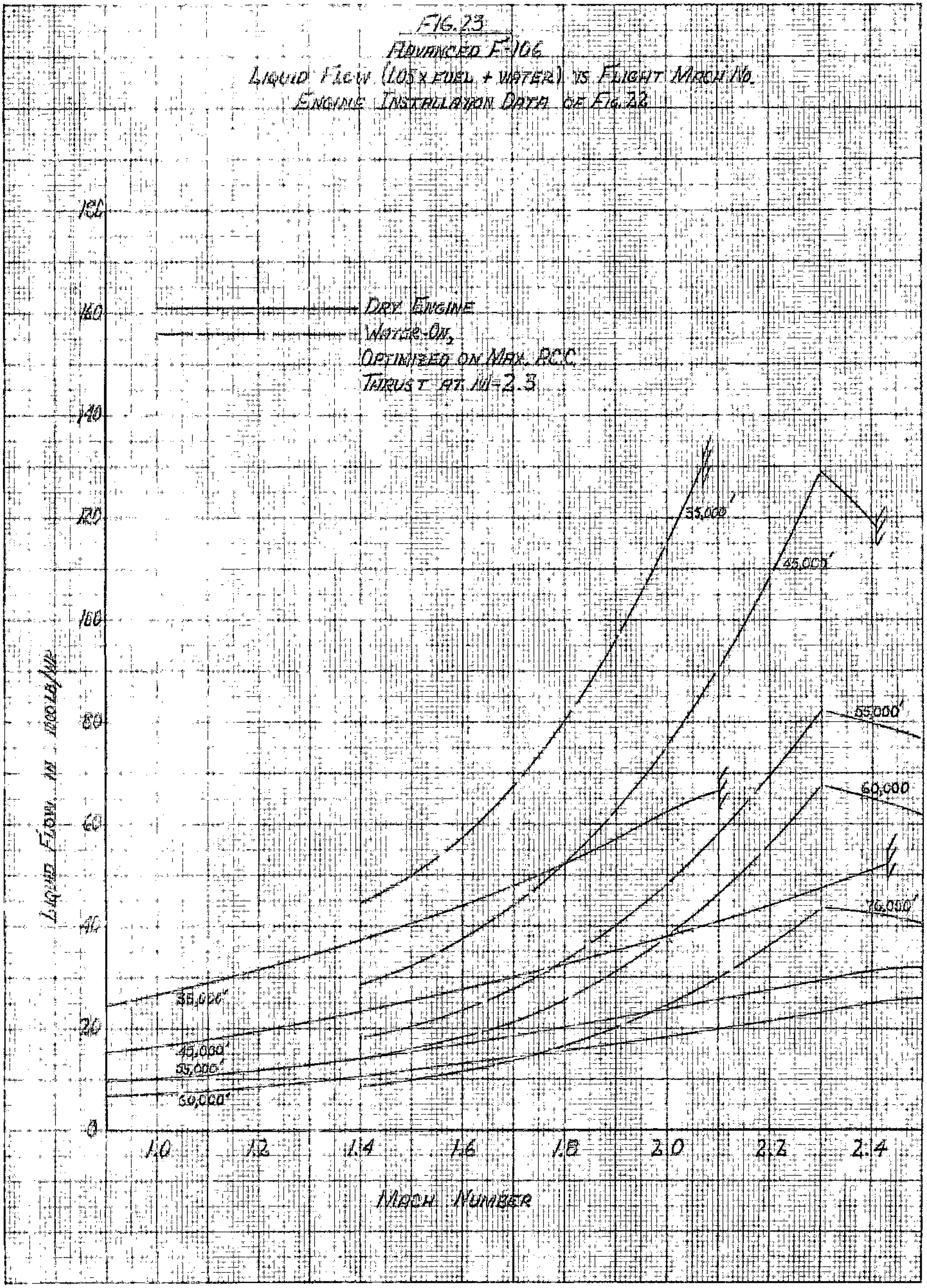


FIG. 23

ADVANCED F-106

LIQUID FLOW (1.05 x FUEL + WATER) vs FLIGHT MACH NO.

ENGINE INSTALLATION DATA OF FIG. 22

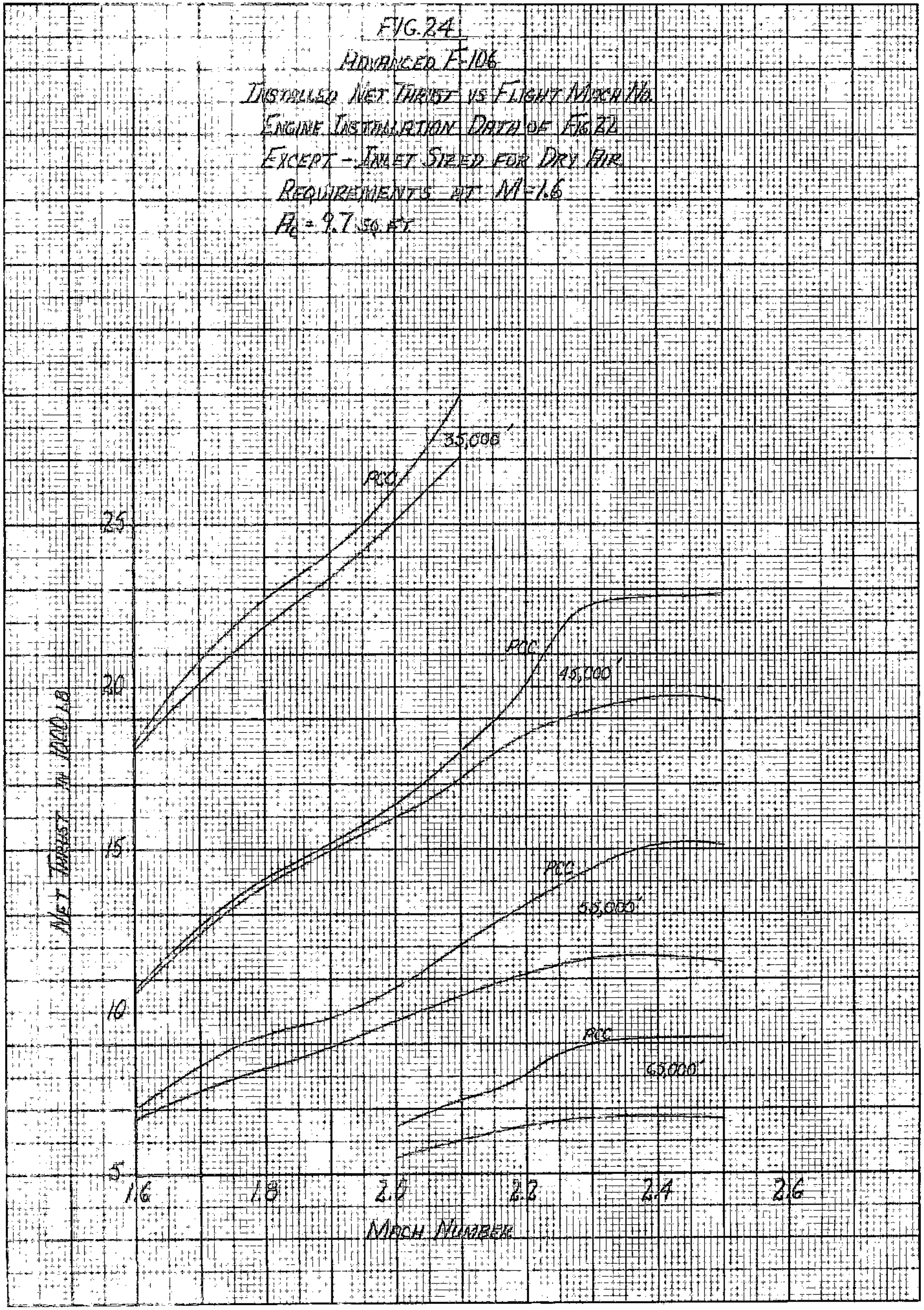


FIG. 24

ADVANCED F-106

INSTALLED NET THRUST VS FLIGHT MACH NO.

ENGINE INSTALLATION DATA OF FIG. 22

EXCEPT - INLET SIZED FOR DRY AIR REQUIREMENTS AT M=1.6

A_c = 9.7 SQ. FT.

10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO

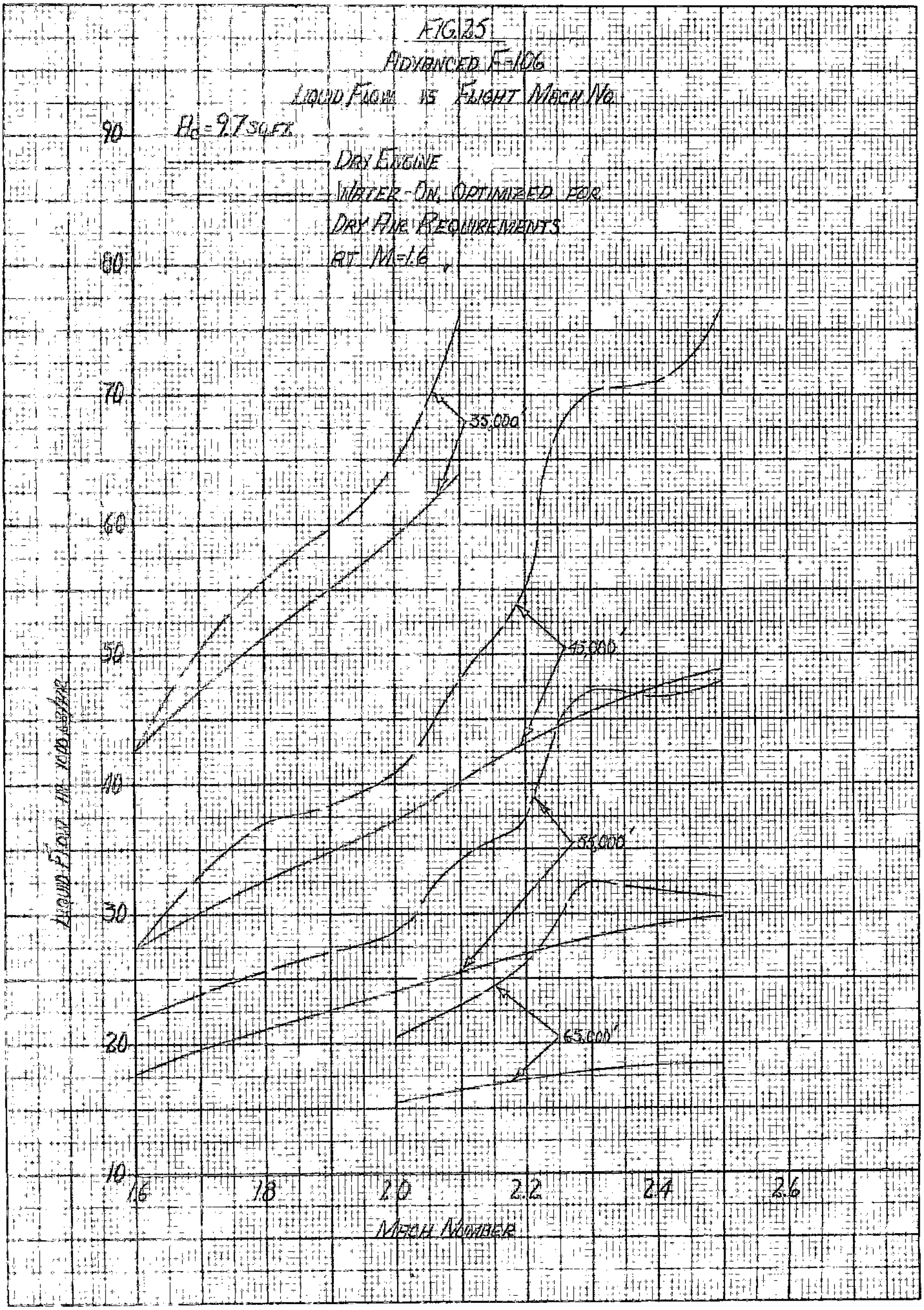
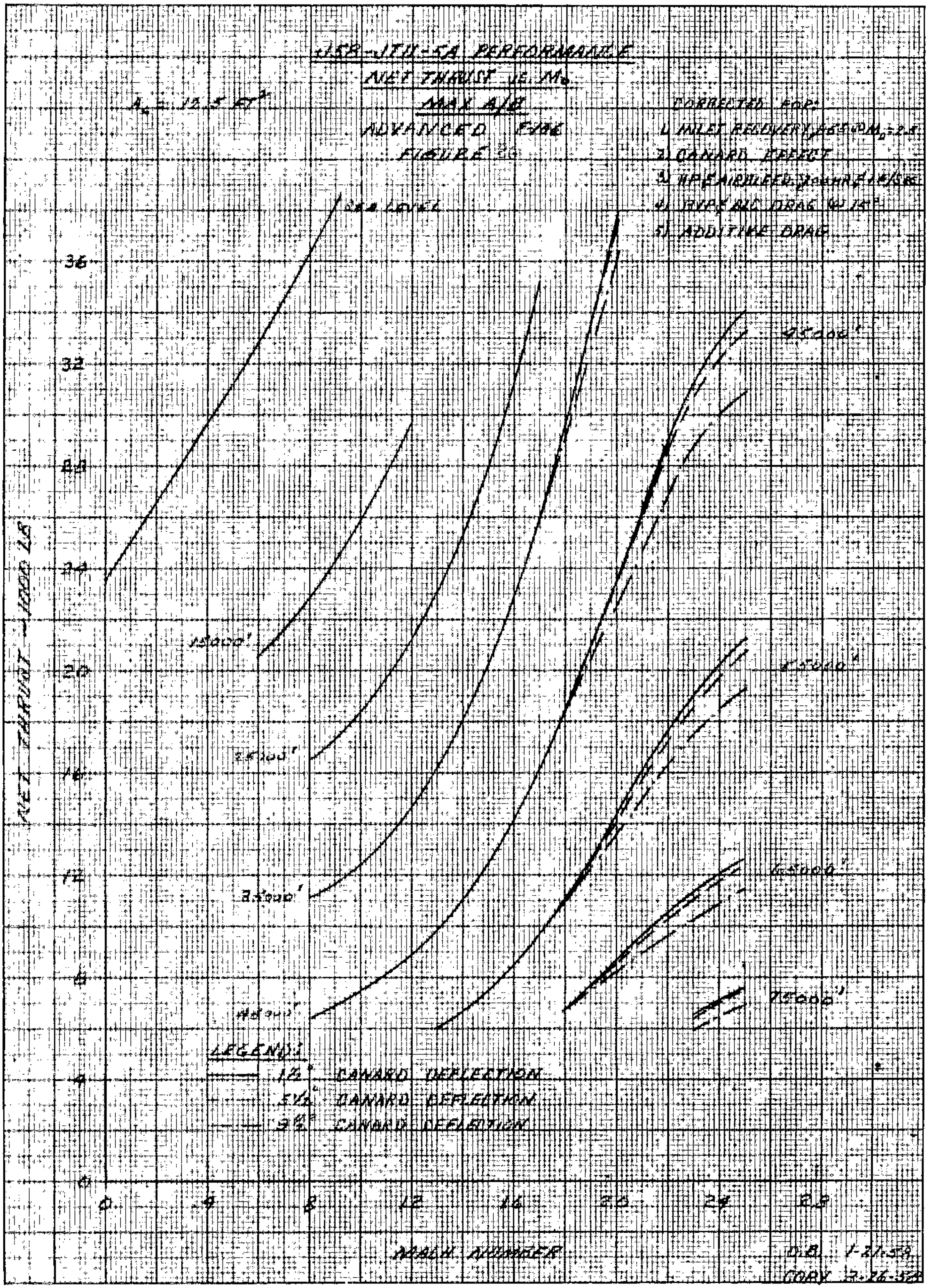


FIG. 25
ADVANCED F-106
LIQUID FLOW VS FLIGHT MACH NO.

K&E 10 X 10 TO THE 1/2 INCH 359-11

K+E 10 X 10 TO THE CM 359T-14G KEUFFEL & ESSER CO. MADE IN U.S.A.



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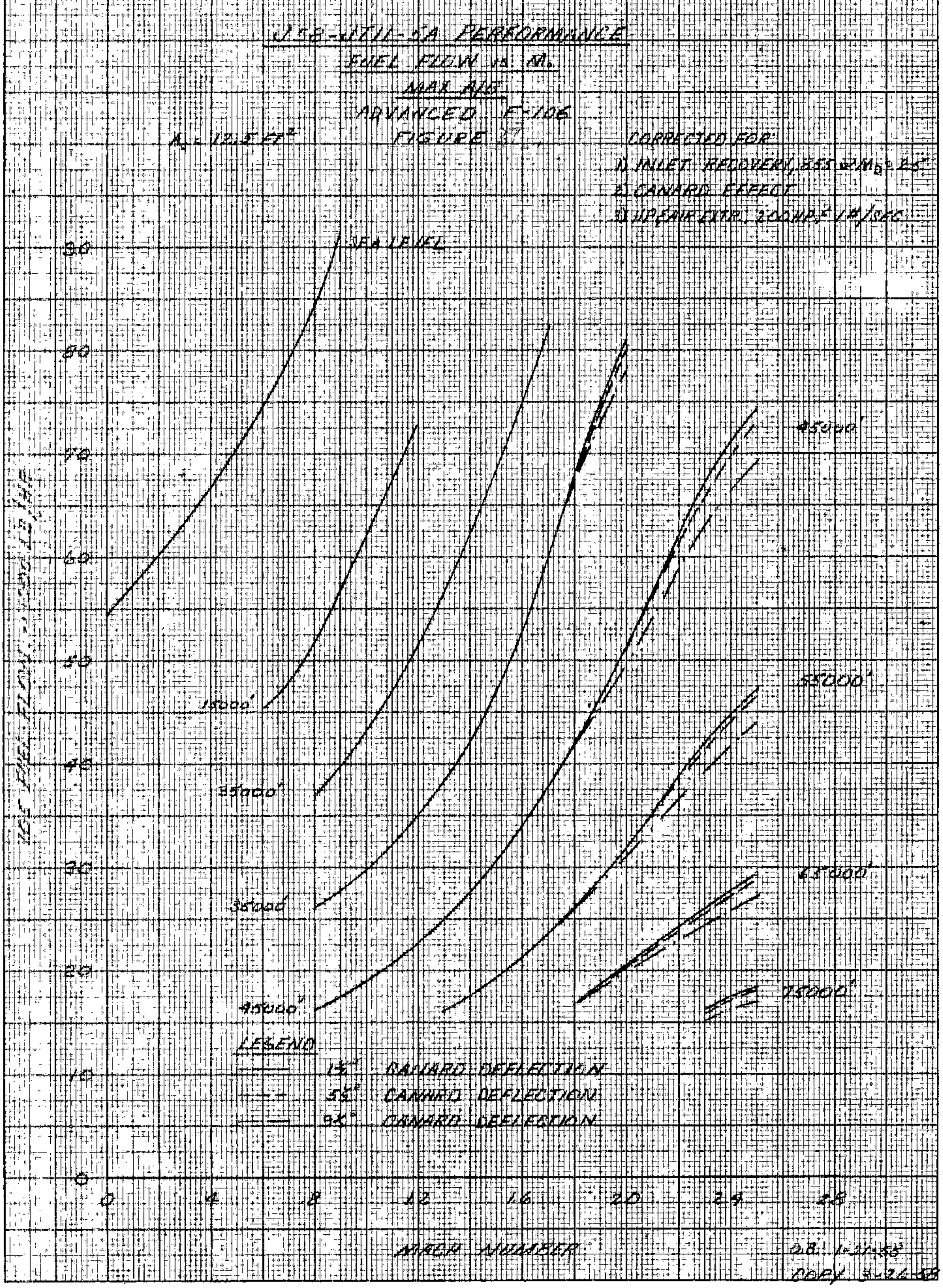

J58-JT11-5A PERFORMANCE
FUEL FLOW vs M₀
MAX A/B
ADVANCED F-106
FIGURE 37

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FIG. 28
ADVANCED F-103
FUEL SPECIFIC IMPULSE VS MACH NO.

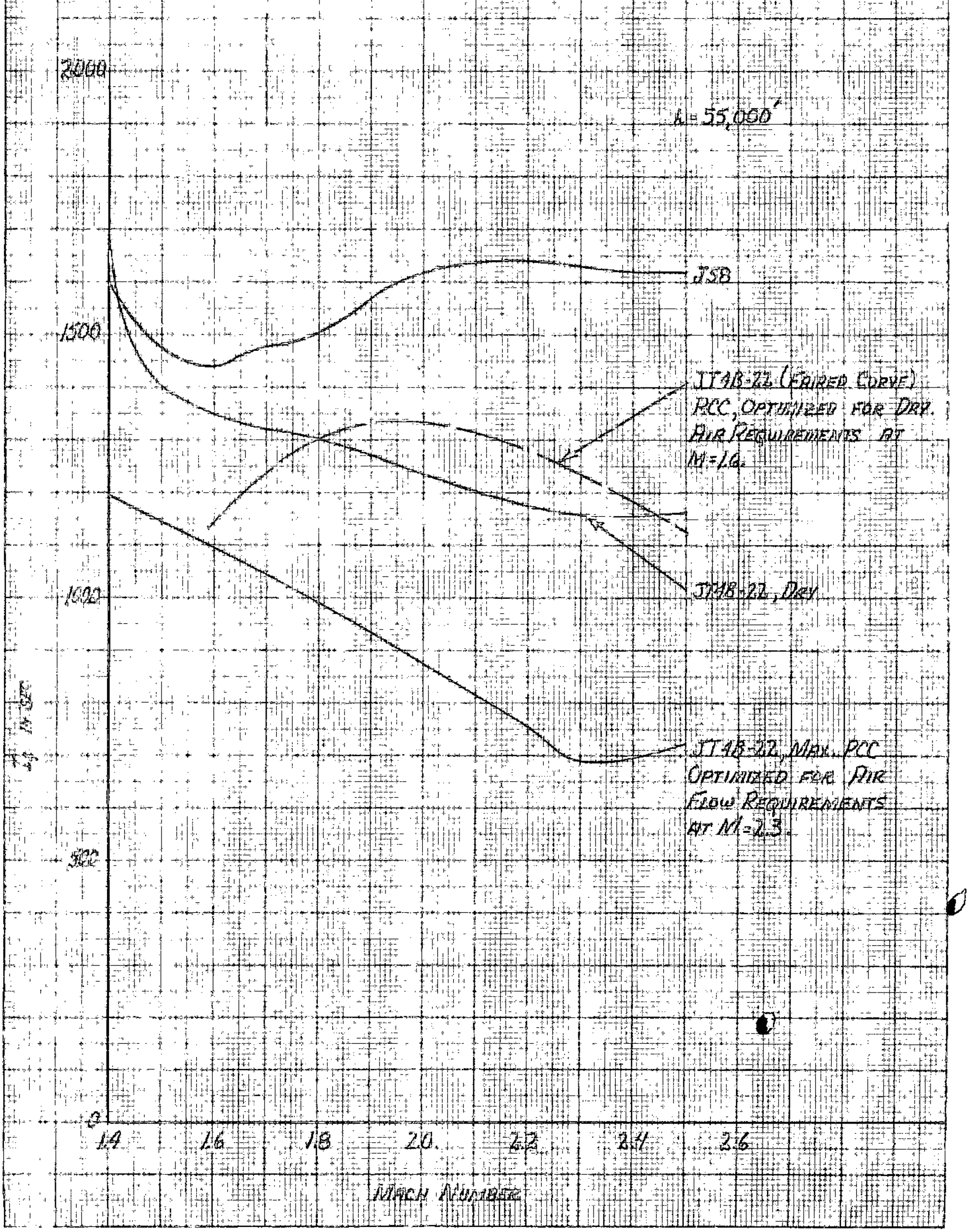
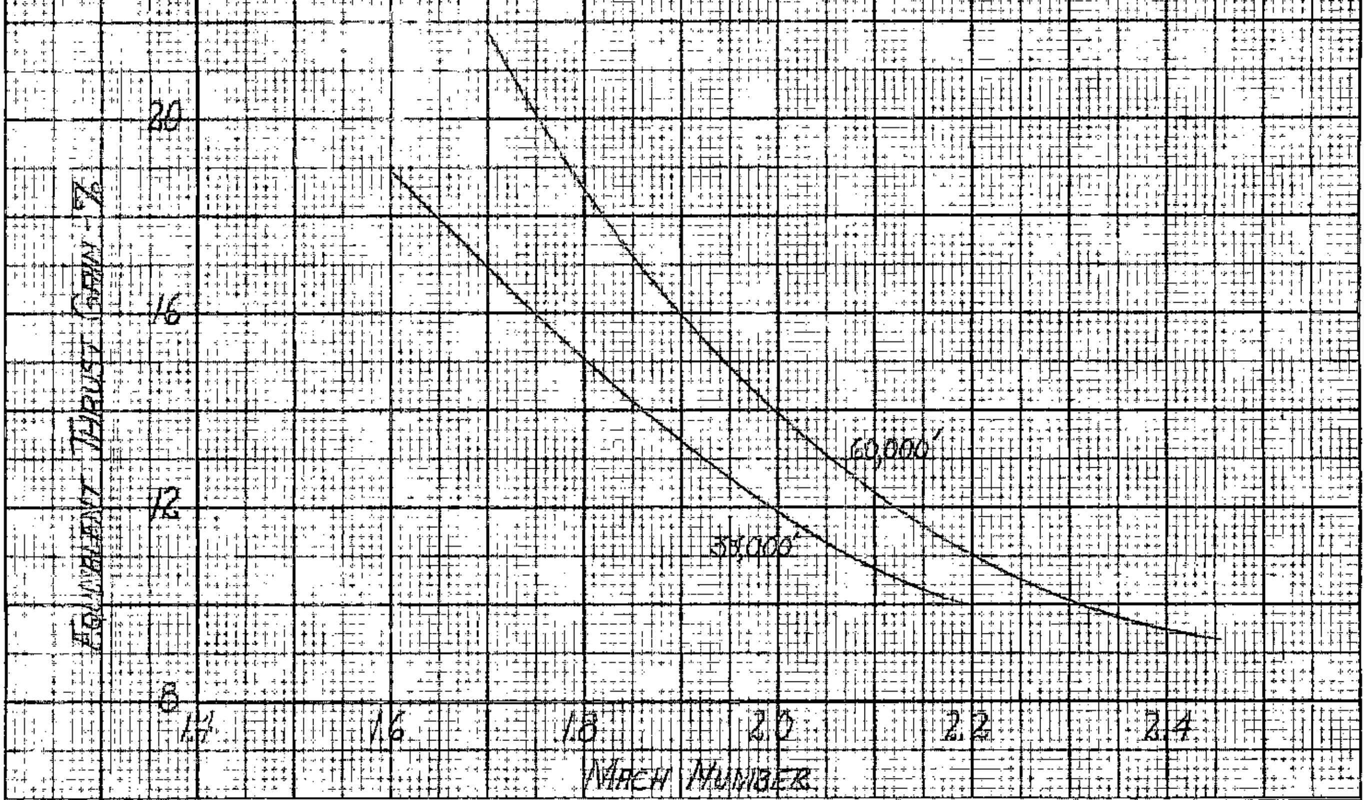


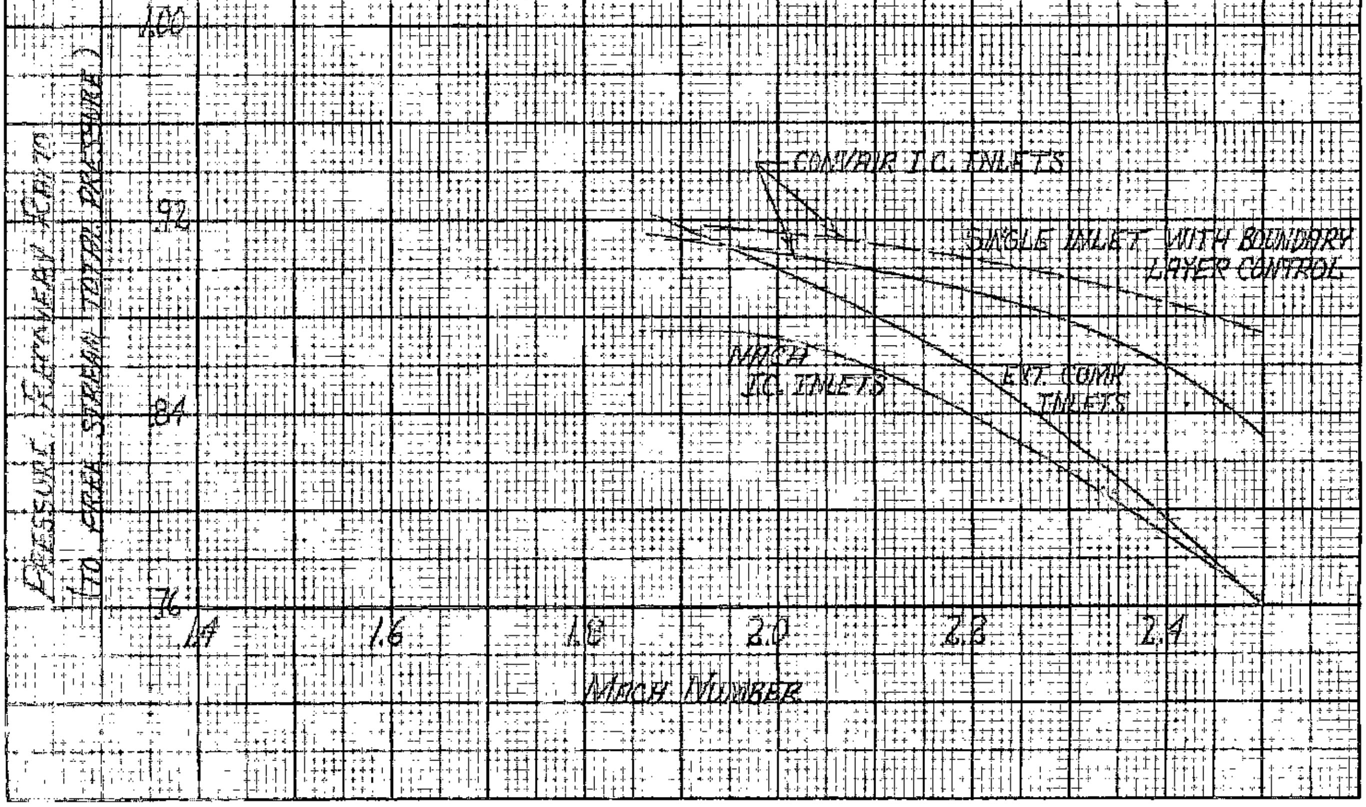
FIG. 28.1

ADVANCED F-106

A. DRAG REDUCTION THROUGH USE OF INTERNAL COMPRESSION INLETS



B. IMPROVEMENT IN PRESSURE RECOVERY
(ENVELOPE OF BEST RECOVERY)

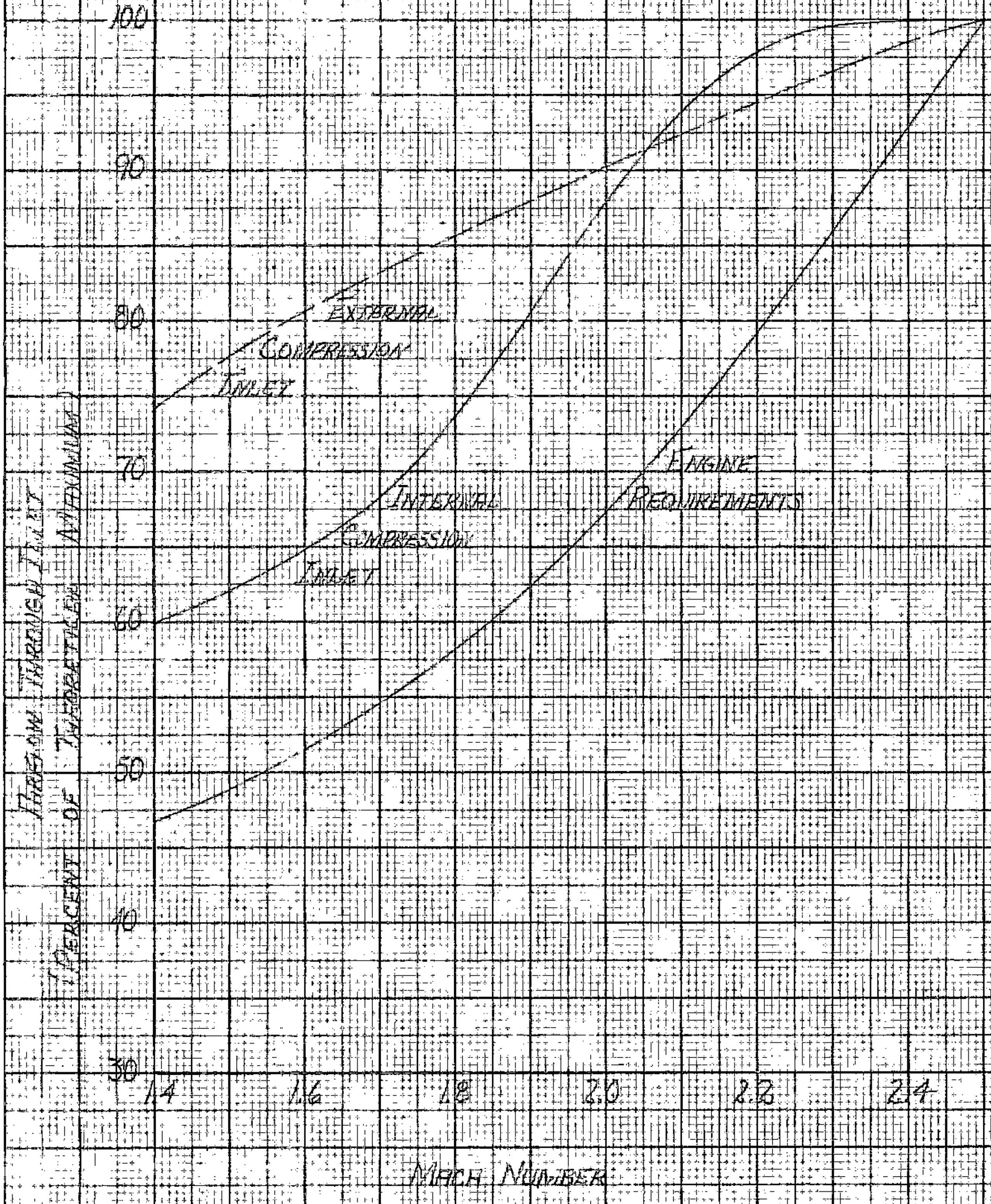


K&E 10 X 10 TO THE ½ INCH 359-11 KEUFFEL & ESSER CO.

FIG. 28.2

ADVANCED F-106

MATCHING OF THE INLET - ENGINE
AIRFLOW CHARACTERISTICS



K&E 10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO.

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FIGURE 30
EFFECT OF REYNOLDS NUMBER
ON INLET PRESSURE RECOVERY
1/12 SCALE ADV F-106
$M_0 = 2.50$

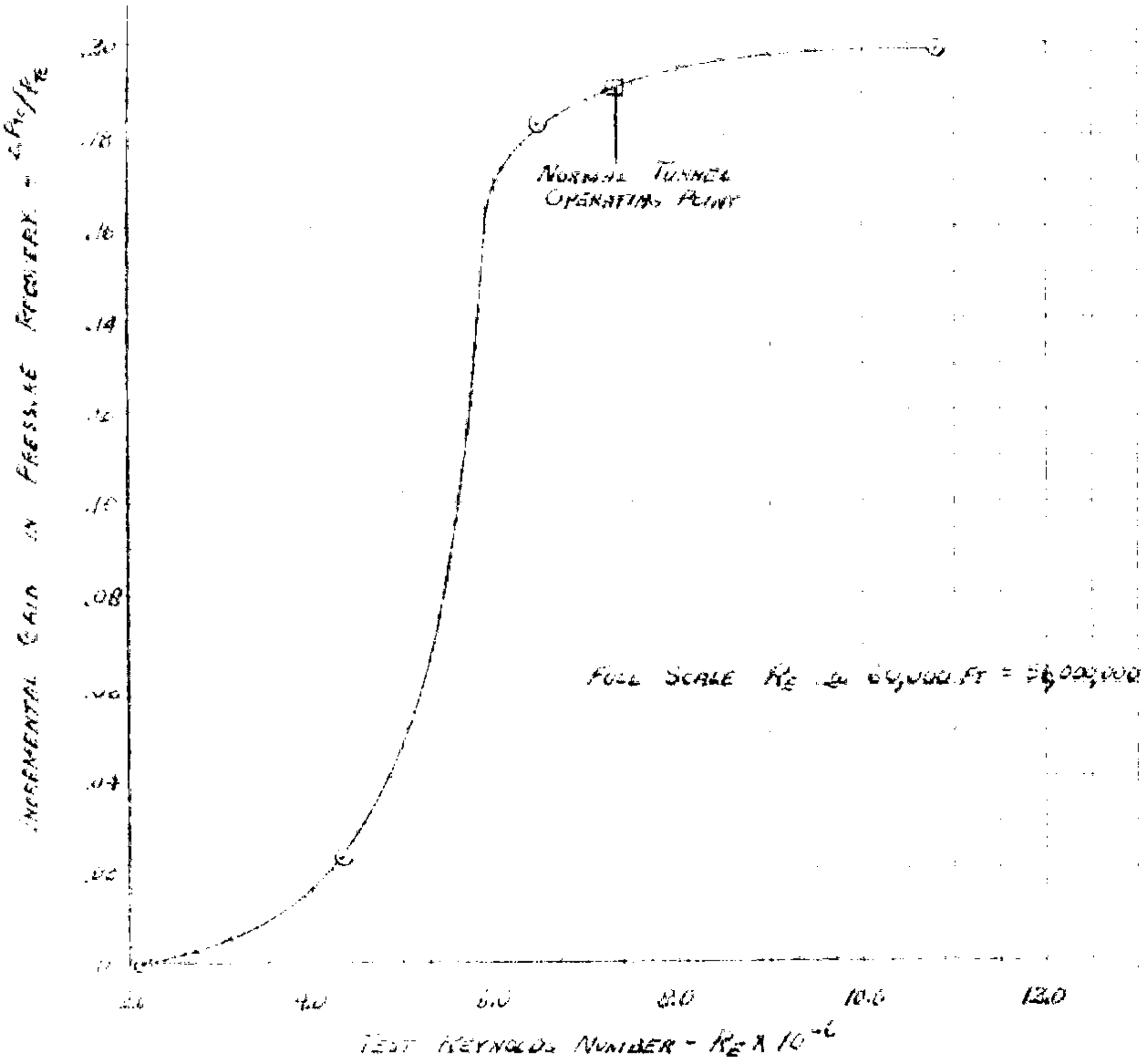

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FIGURE 31

INLET PRESSURE RECOVERY

VS MACH NUMBER

F106-ADV

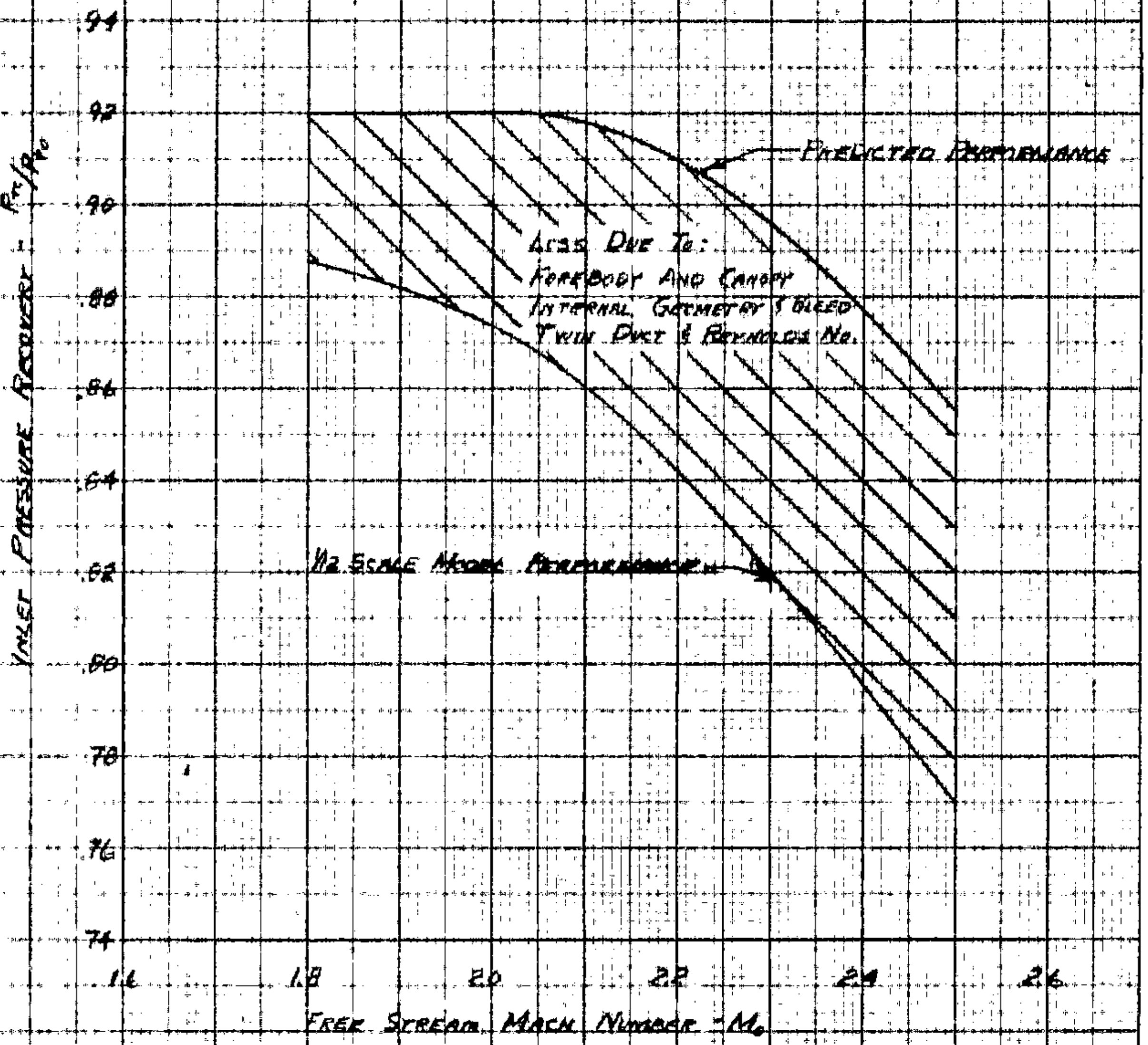

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FIG. 52
ADVANCED F-106
INTERNAL COMPRESSION INLET CONTROLS SYSTEM

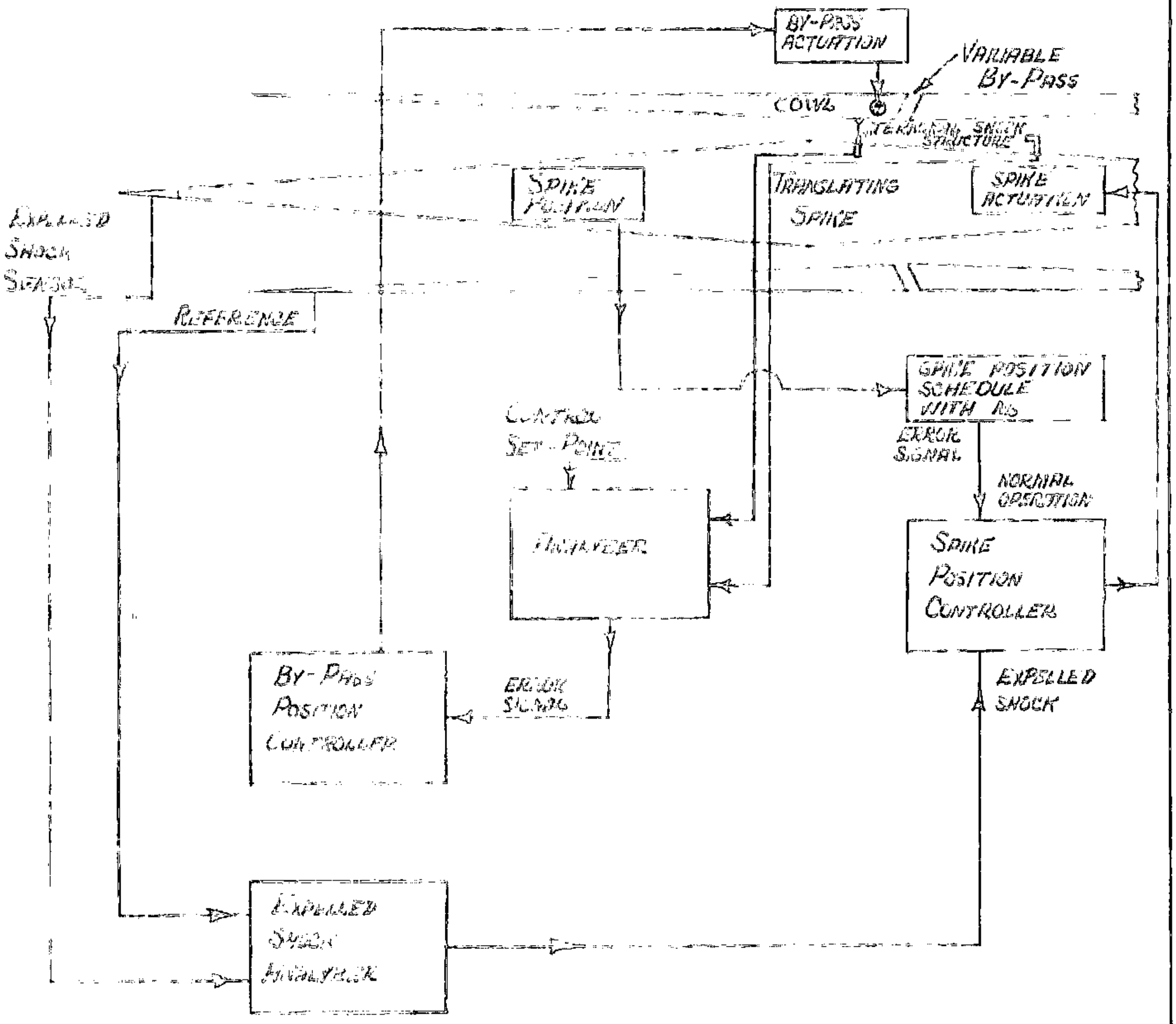

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FIGURE 29

ADVANCED F-106 CONFIGURATION

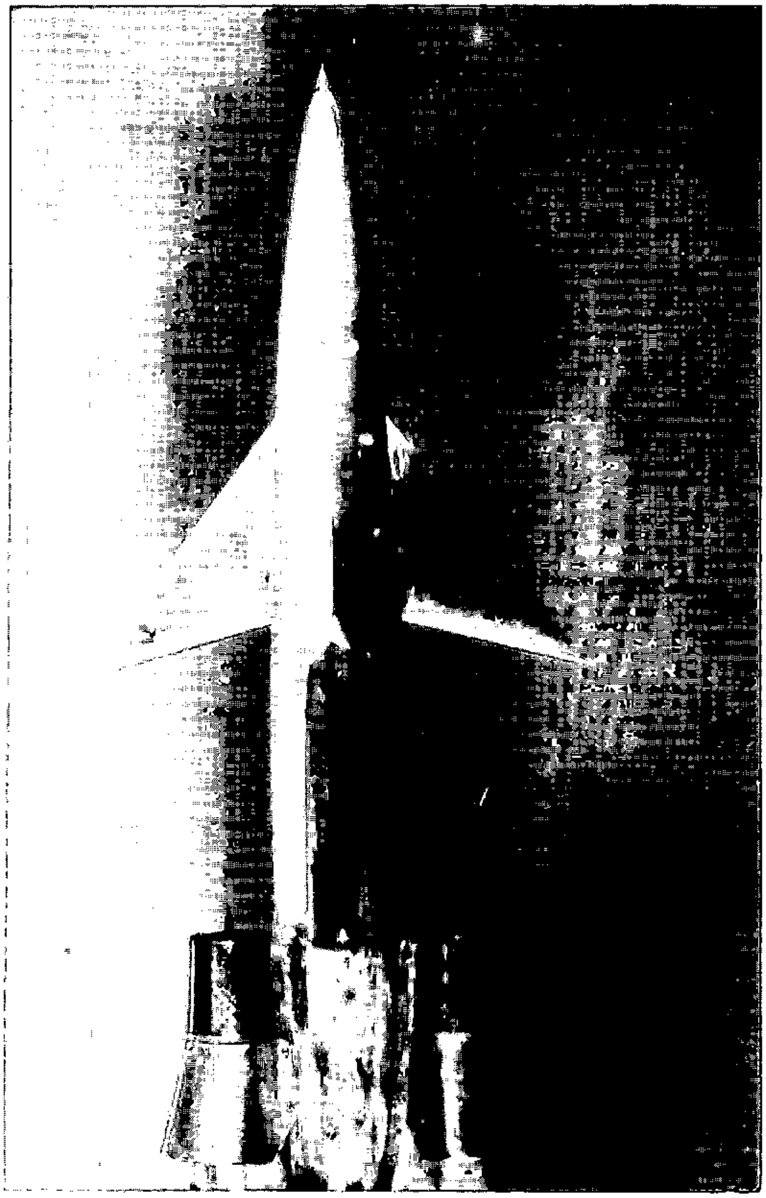

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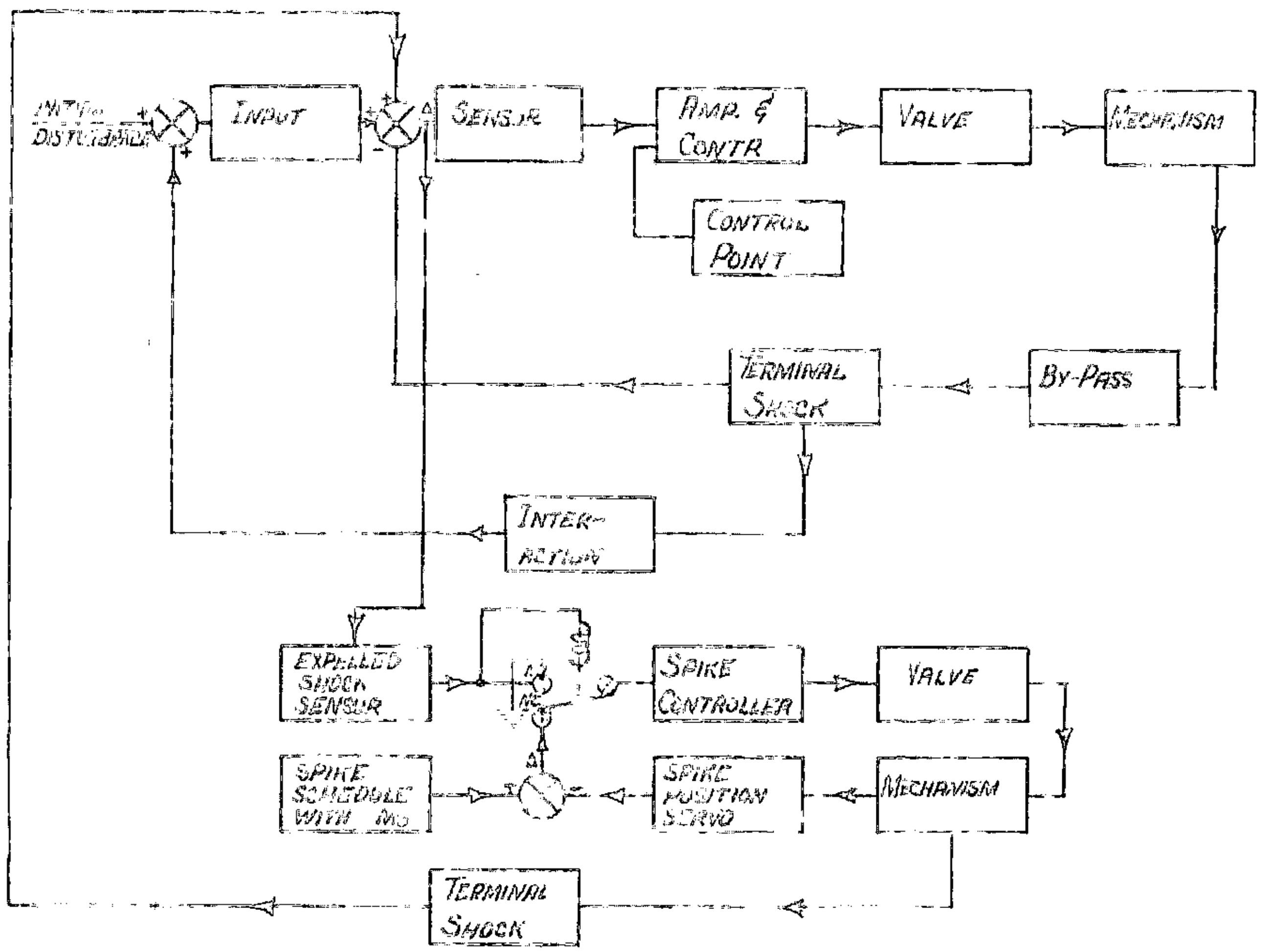
FIG. 33

ADVANCED F-106

INTERNAL COMPRESSION INLET CONTROLS BLOCK DIAGRAM



SYSTEM PARAMETERS CONSIDERED IN ANALOG STUDY

1. INITIATING DISTURBANCE : TIME HISTORY AND MAGNITUDE OF MOST SEVERE DISTURBANCE EXPECTED.

2. RESPONSE OF THE TERMINAL SHOCK AND MAGNITUDE OF SHOCK POSITION CHANGE FOR GIVEN DISTURBANCE.

3. RESPONSE OF TERMINAL SHOCK SENSING DEVICE TO SHOCK MOVEMENT INCLUDING EFFECTS OF SHOCK-BOUNDARY LAYER INTERACTION AND PROBE CONFIGURATION.

4. AMPLIFIER, VALVE AND MECHANISM CHARACTERISTICS INCLUDING NON-LINEAR EFFECTS, HYSTERESIS, AND NETWORK COMPENSATION.

5. CHANGE IN EXIT AREA WITH BY-PASS POSITION.

6. TERMINAL SHOCK MOVEMENT DUE TO BY-PASS AREA CHANGE INCLUDING EFFECT OF DIFFUSER-TERMINAL SHOCK DYNAMICS AND MAGNITUDE OF SHOCK MOVEMENT.

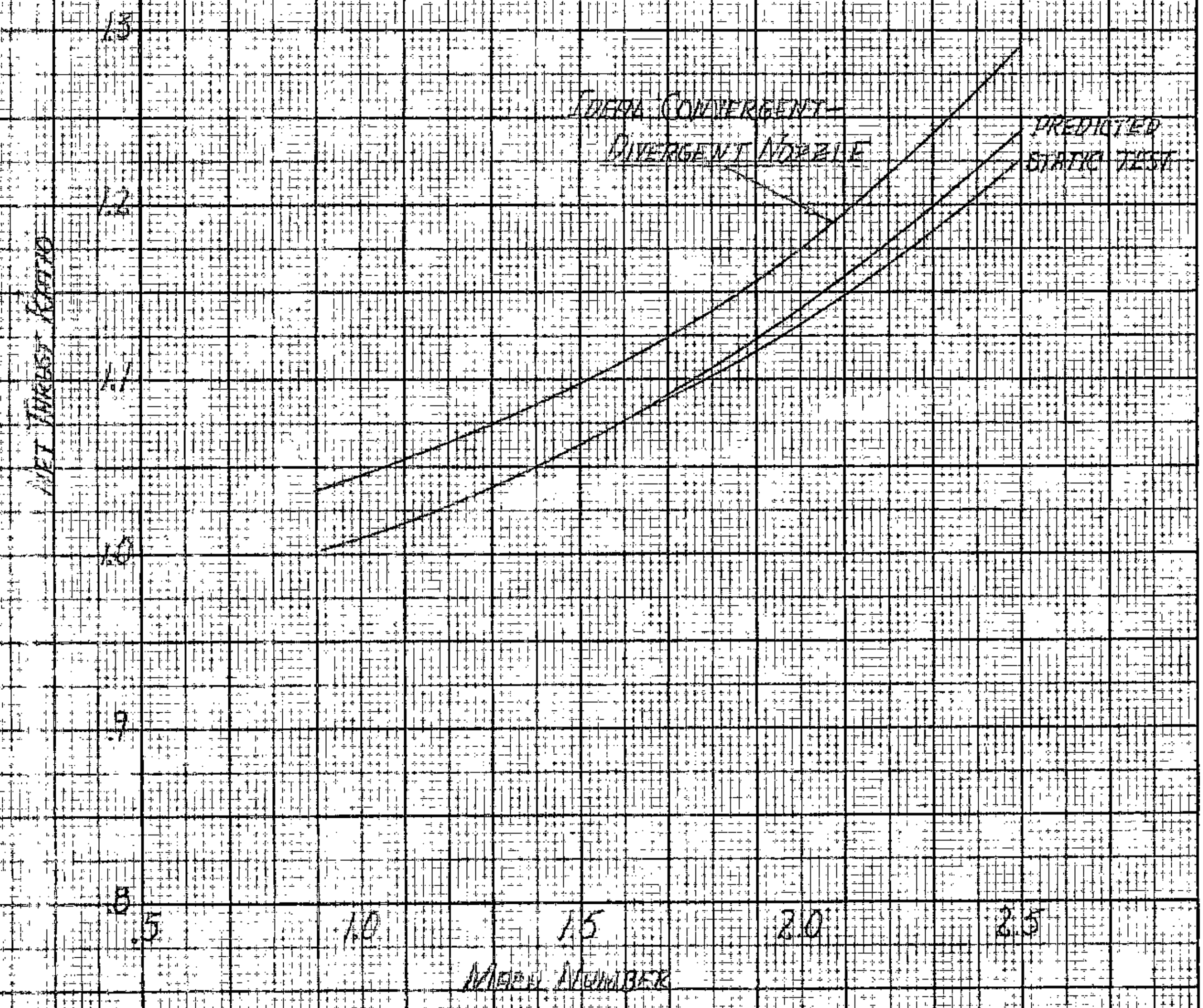
7. INTERACTION EFFECTS OF THE RESULTING PRESSURE CHANGES ON THE ORIGINAL DISTURBANCE

FIG. 34

ADVANCED F-106

VARIABLE EJECTOR PERFORMANCE

ENGINE: JT4B-22, MAX. A/B POWER WITH
WATER INJECTION
CONDITIONS: 35,000', STANDARD DAY



K&E 10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO.

FIG. 35A
ADVANCED F-166 WINDSHIELD AND CANOPY TEMPERATURES

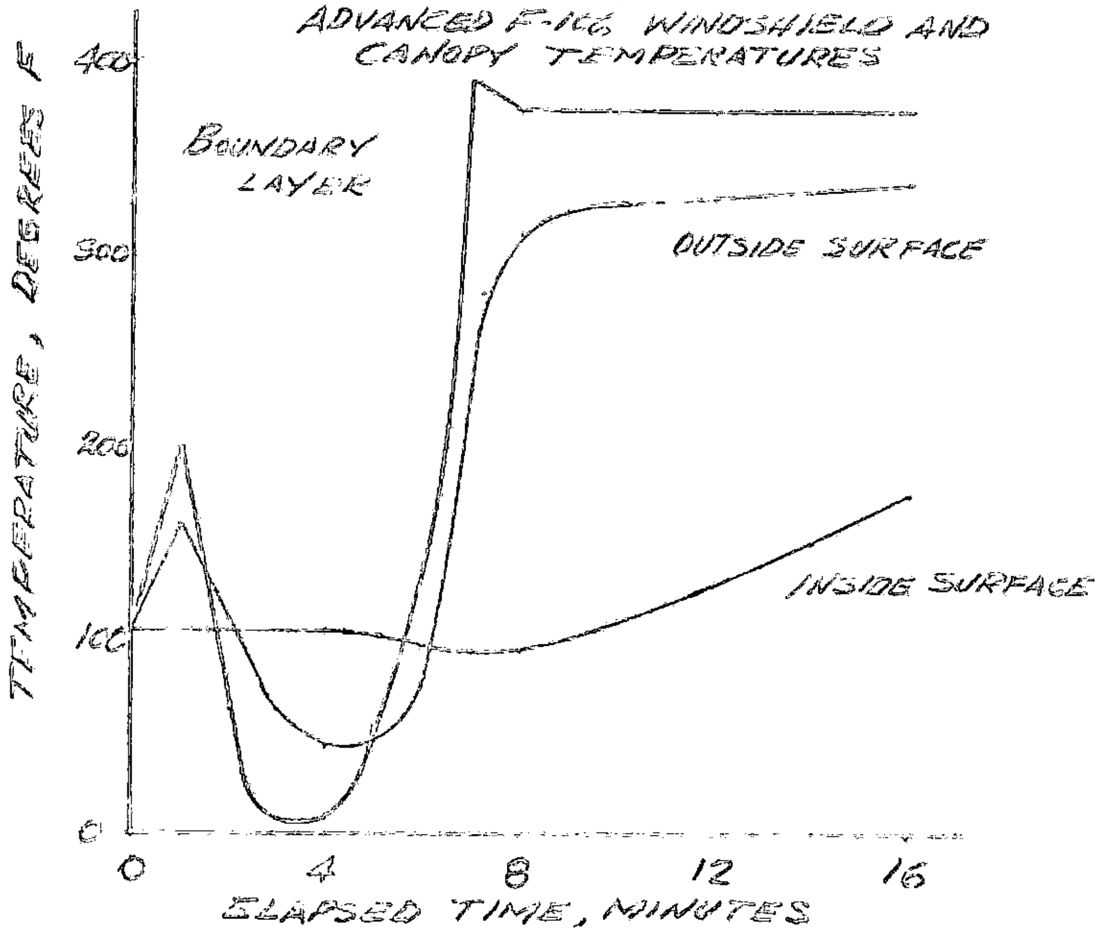


FIG. 35B
WINDSHIELD AND CANOPY GRADIENTS

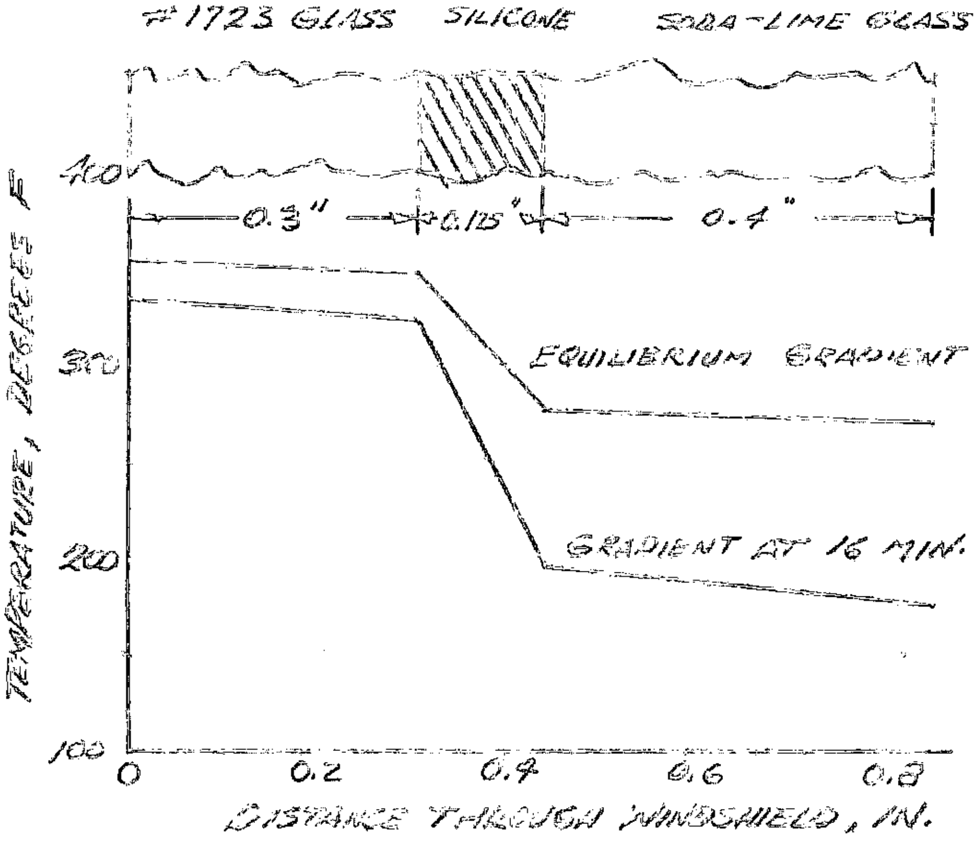


FIG 36
ENGINE OIL COOLING SCHEMATIC
ADVANCED F-106

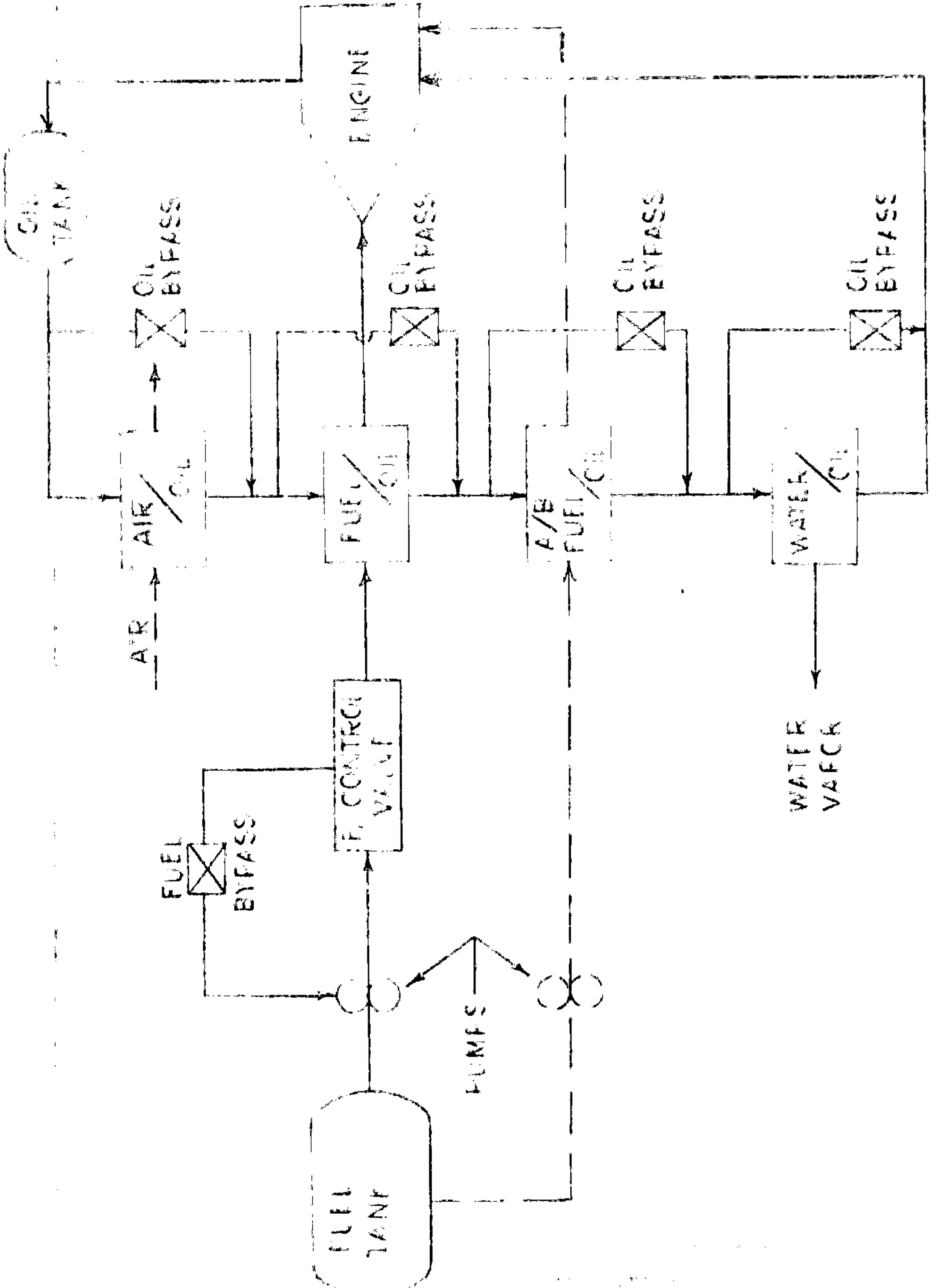


FIGURE 37

ADVANCED F-106 STRUCTURAL TEMPERATURE GRADIENTS

ALUMINUM STRUCTURE

SKIN = .060"
FLANGE = .020"
WEB = .10"

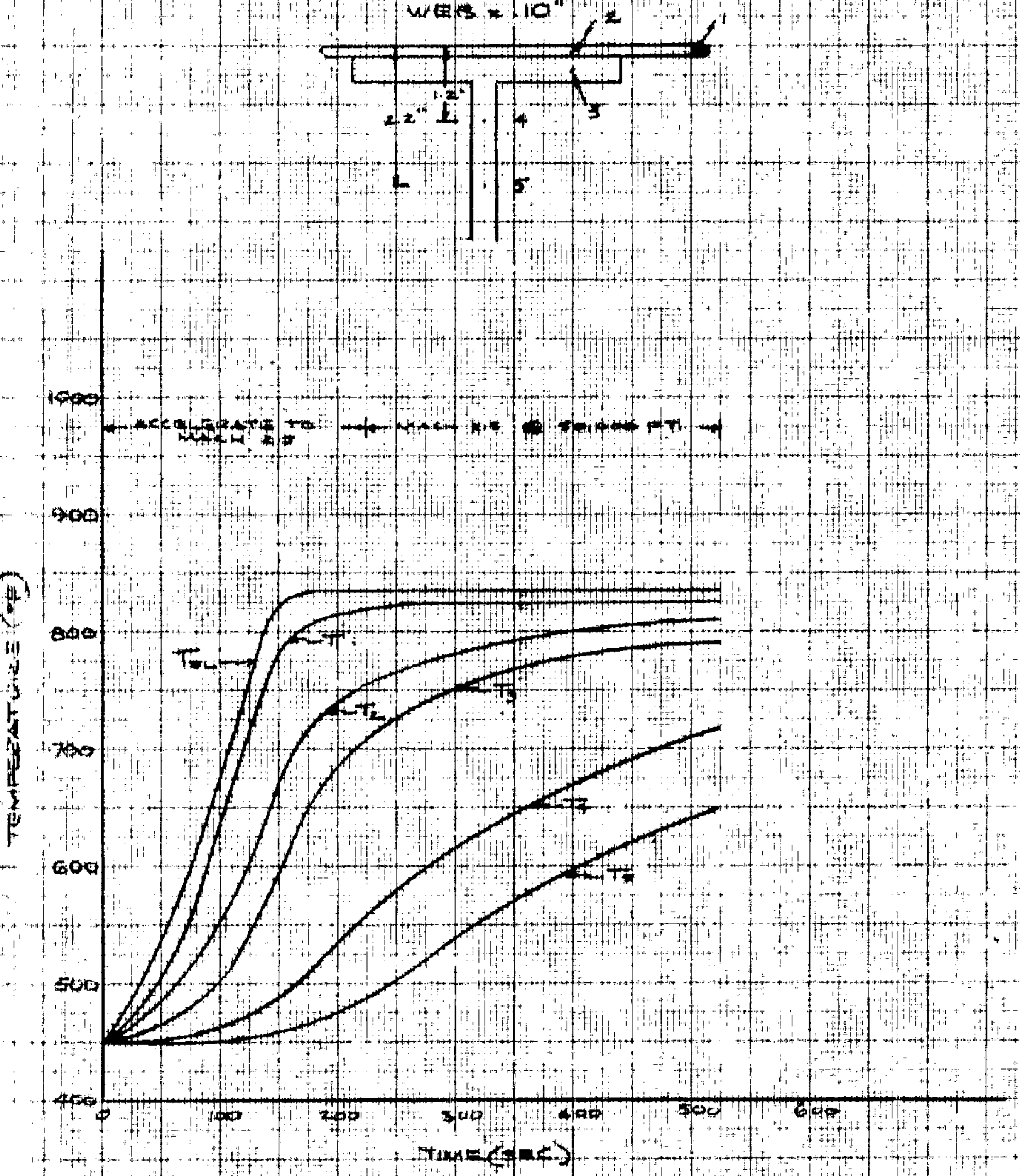


FIG. 38
FUSELAGE SKIN TEMPERATURES
ADVANCED F-106

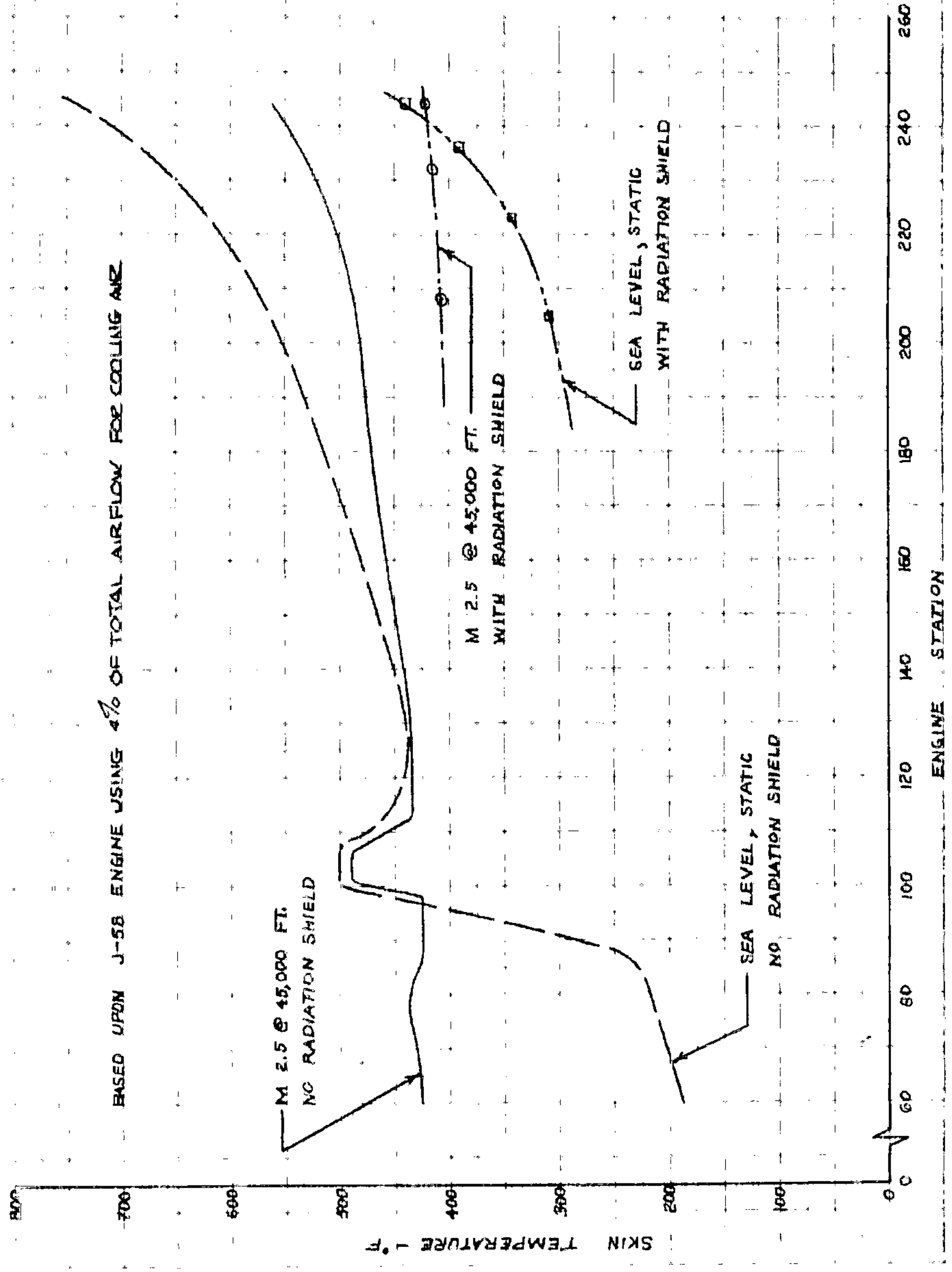


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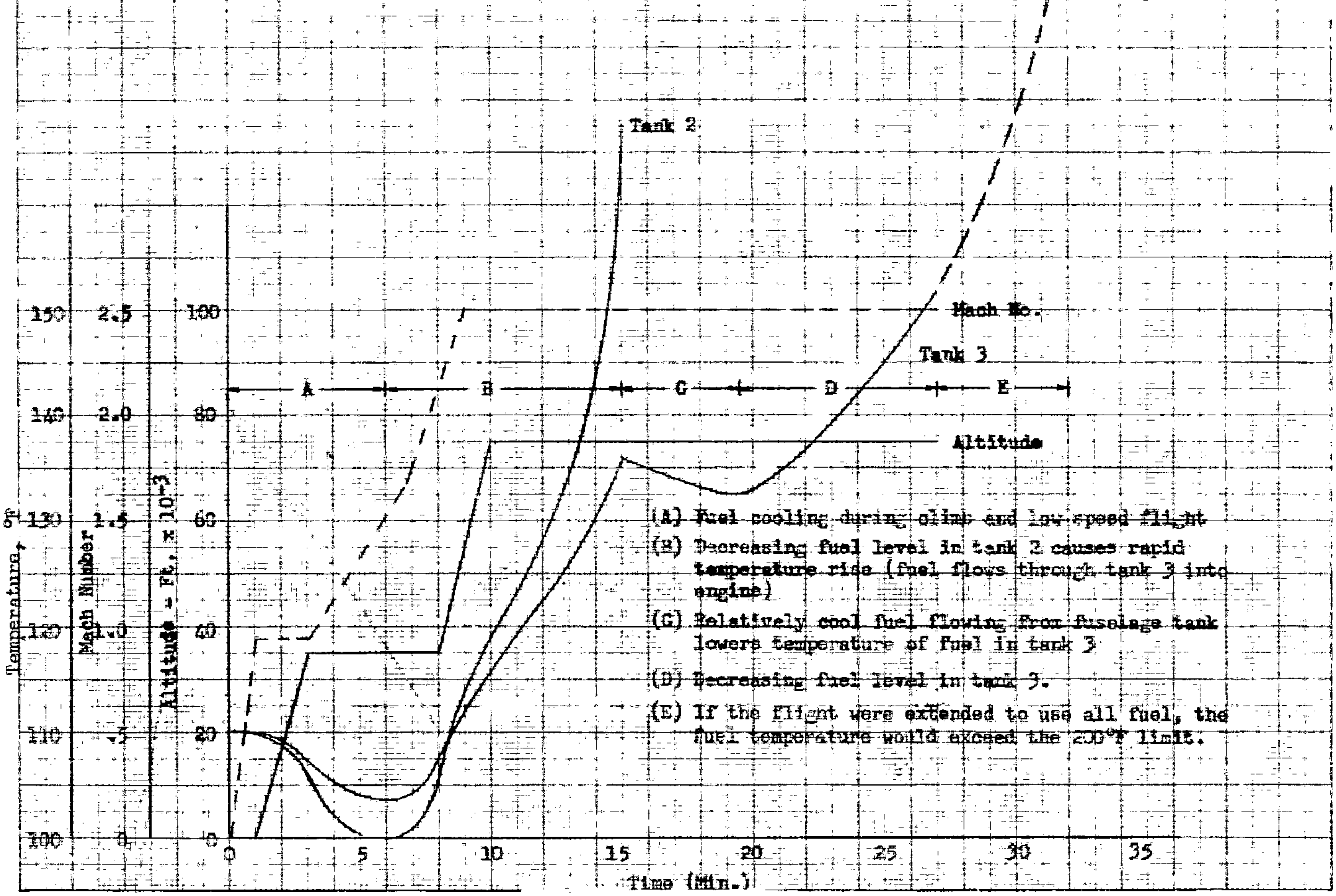
FIGURE E-[illegible]

Fuel Temperature vs. Time F-[illegible]



(A) Fuel cooling during climb and low speed flight

(B) Decreasing fuel level in tank 2 causes rapid temperature rise (fuel flows through tank 3 into engine)

(C) Relatively cool fuel flowing from fuselage tank lowers temperature of fuel in tank 3

(D) Decreasing fuel level in tank 3.

(E) If the flight were extended to use all fuel, the fuel temperature would exceed the 200°F limit.

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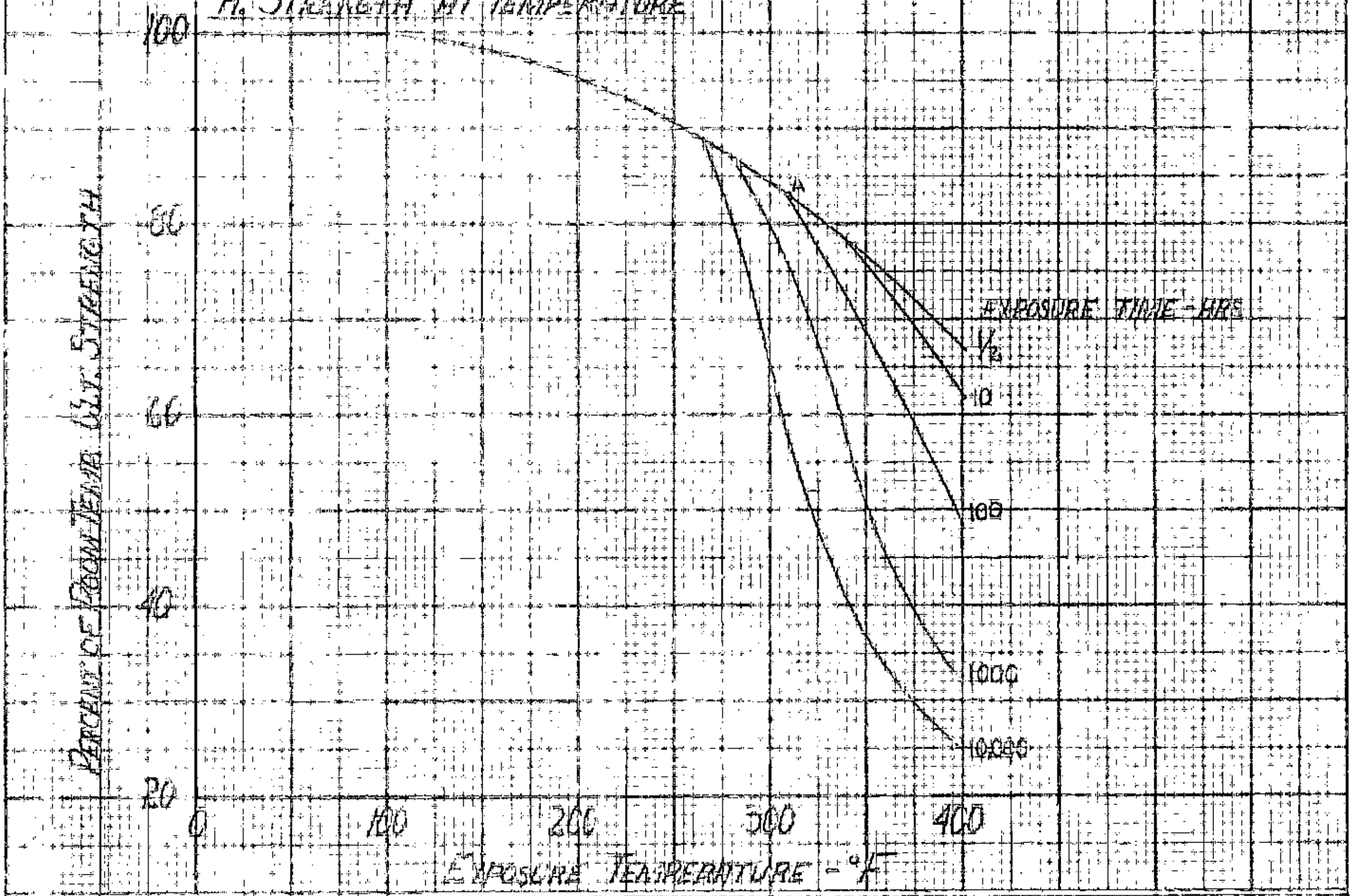
FIG 39.1

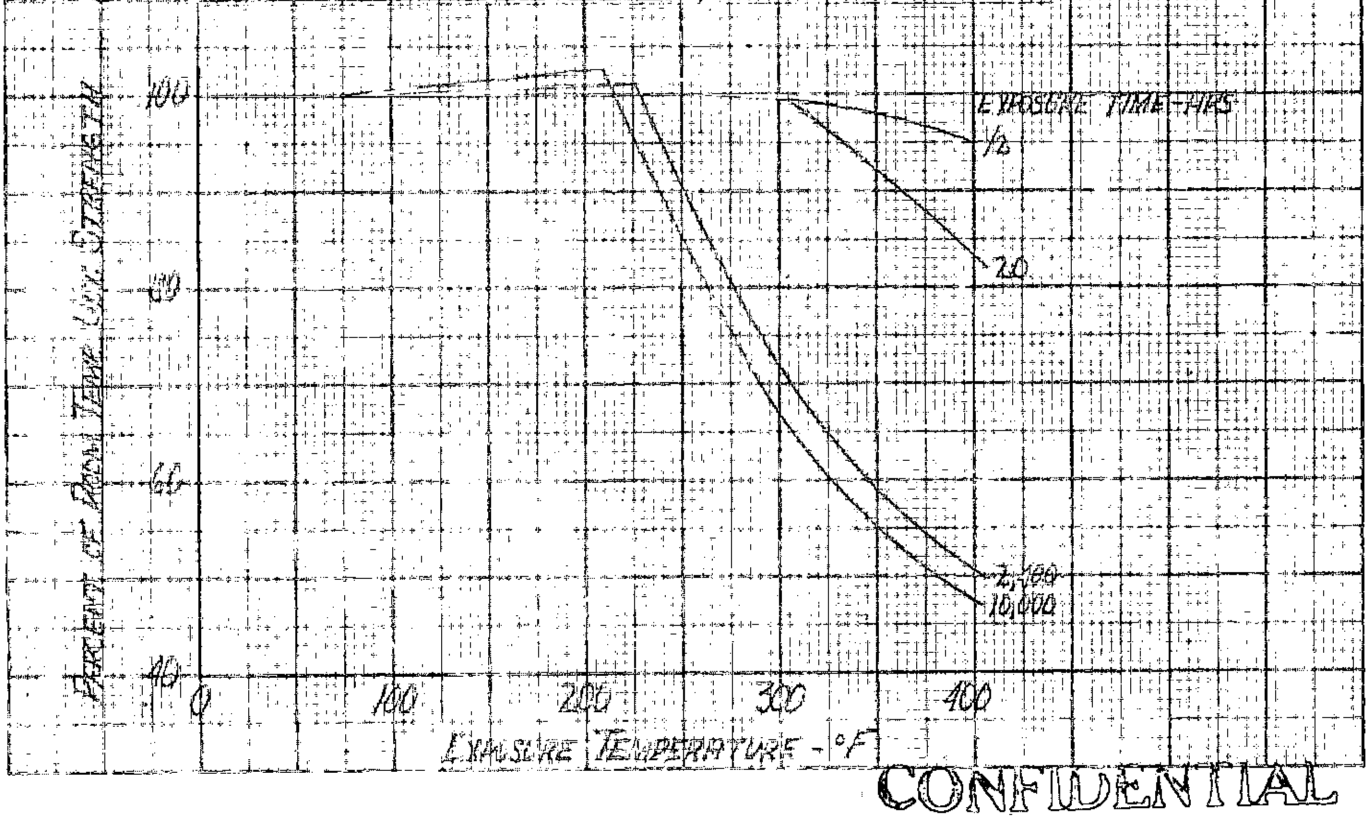
ADVANCED F-106

ANC-5 DATA FOR 2014-T6

A. STRENGTH AT TEMPERATURE



B. ROOM TEMPERATURE STRENGTH AFTER EXPOSURE



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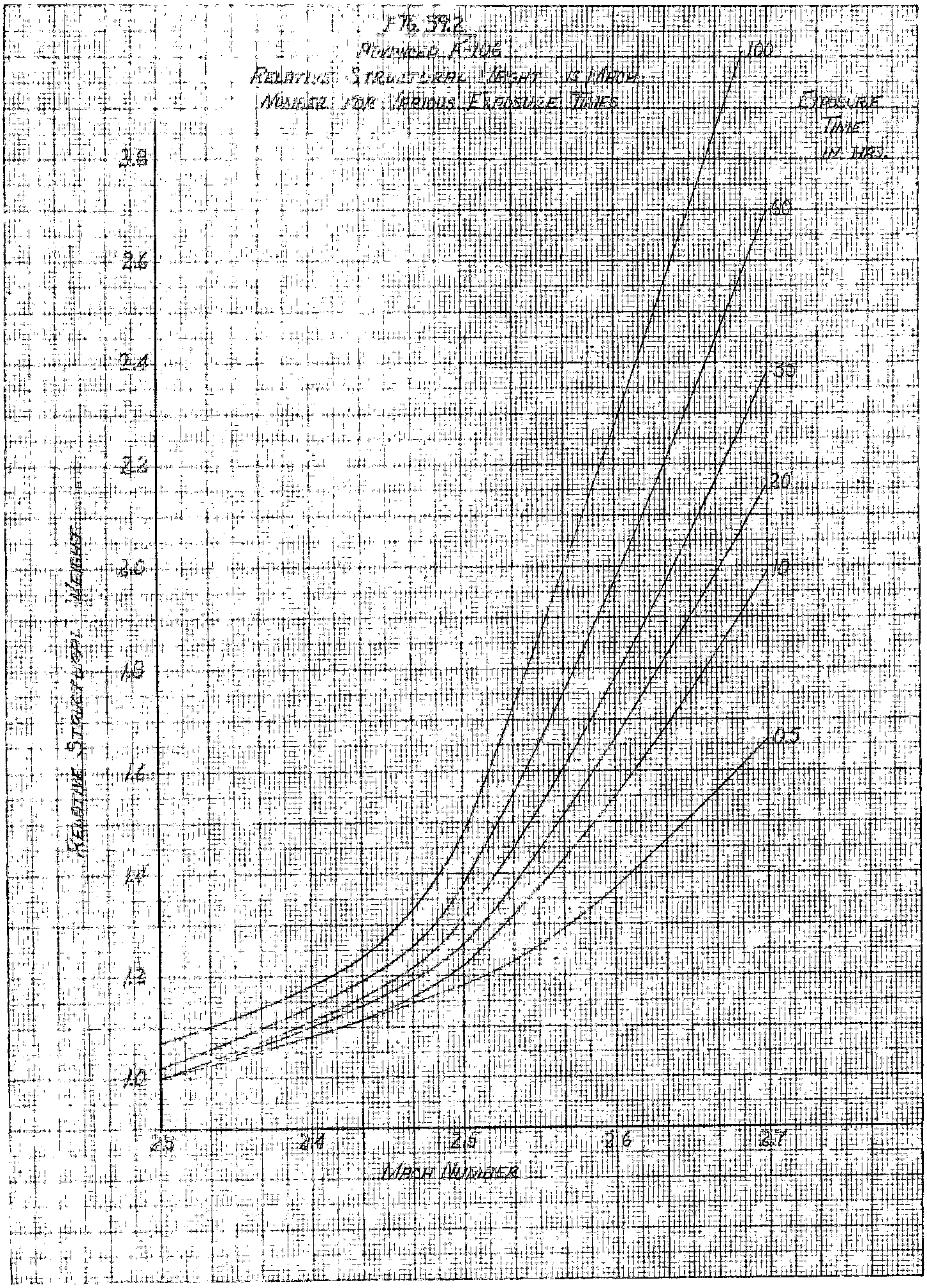

FIG. 59.2
Modified A-106
Relative Structural Weight vs Mach Number for Various Exposure Times

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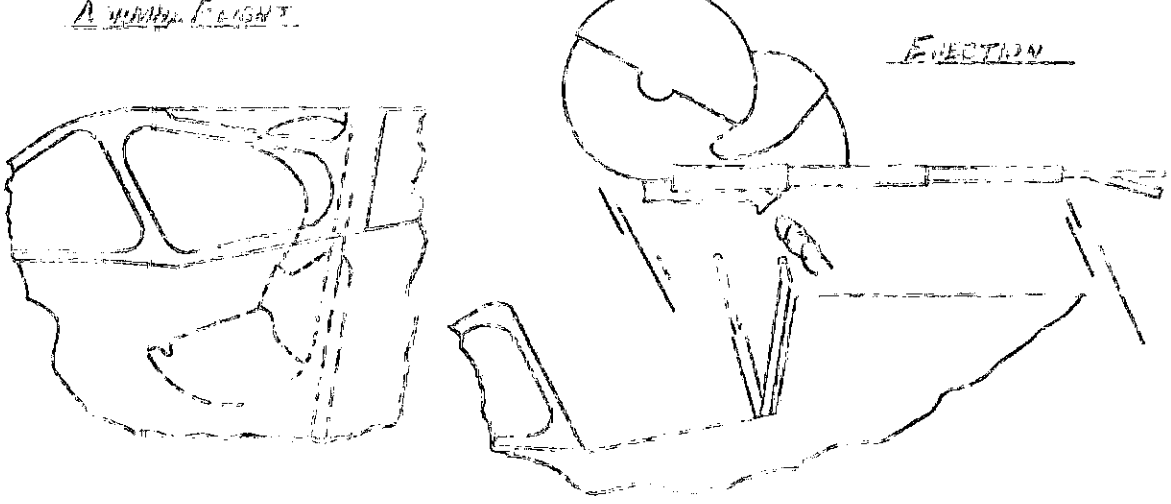
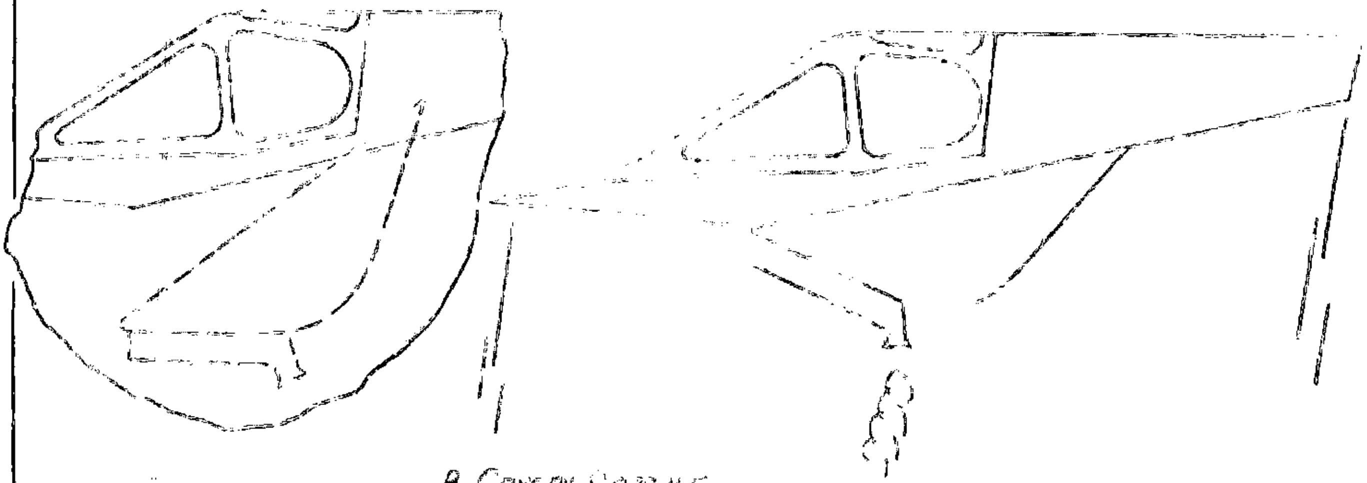
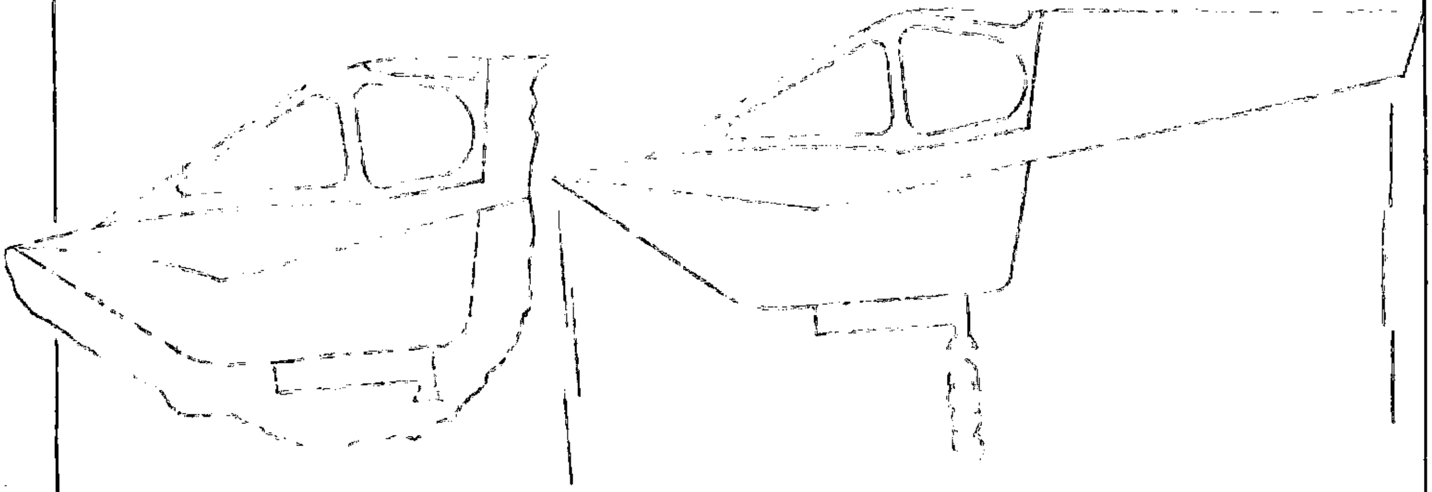
FIG. 40 ADVANCED F-106
BASIC CAPSULE TYPES



A. ENCLOSED SEAT



B. CANOPY CAPSULE



C. COCKPIT CAPSULE

FIG. 41
ADVANCED F-106
FREE FLIGHT MODEL TEST CONFIGURATIONS

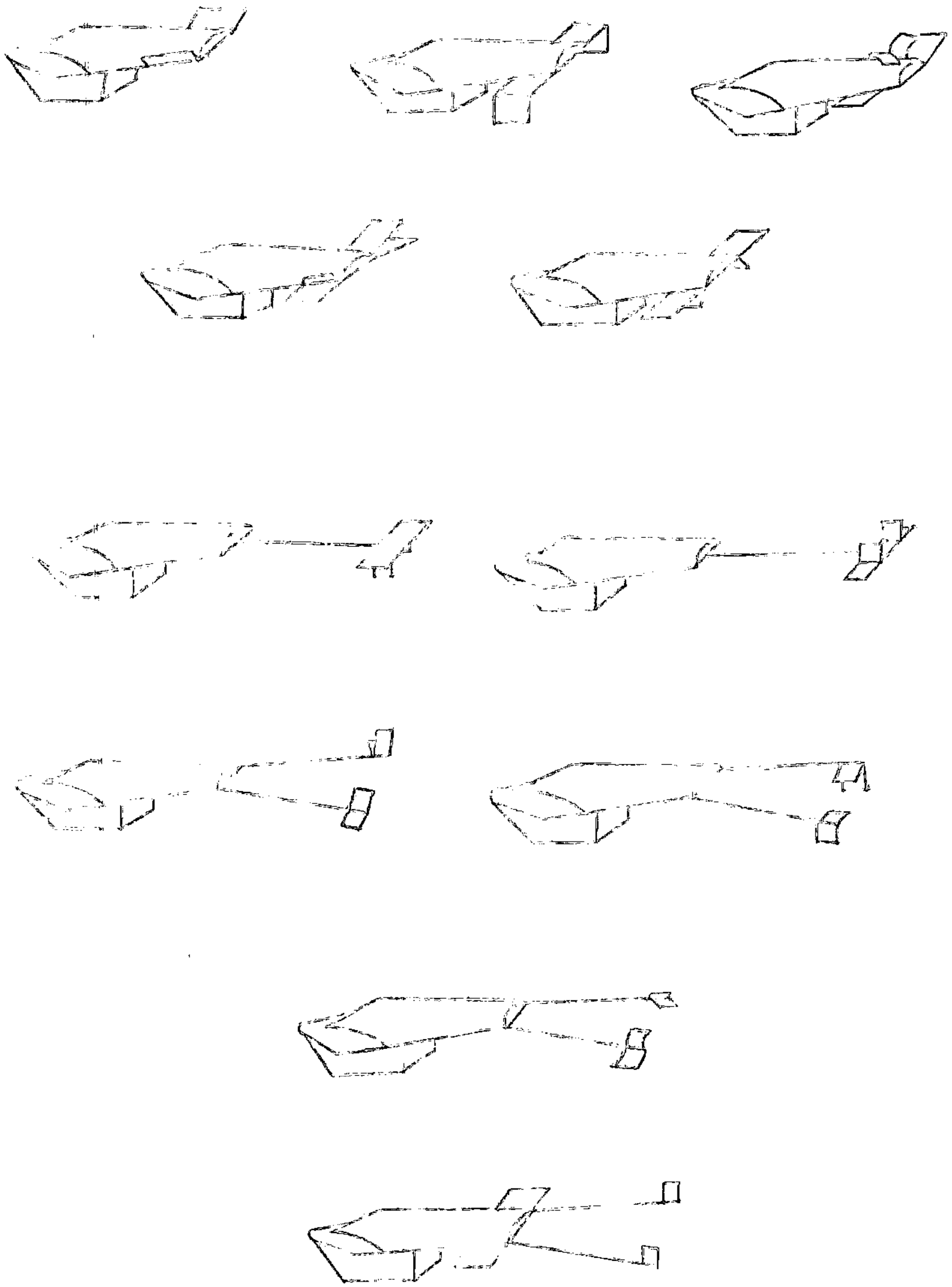


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Fig. 42
F-106 - 30
Speed Altitude Envelope
(Gross Wt = 37000 lb)

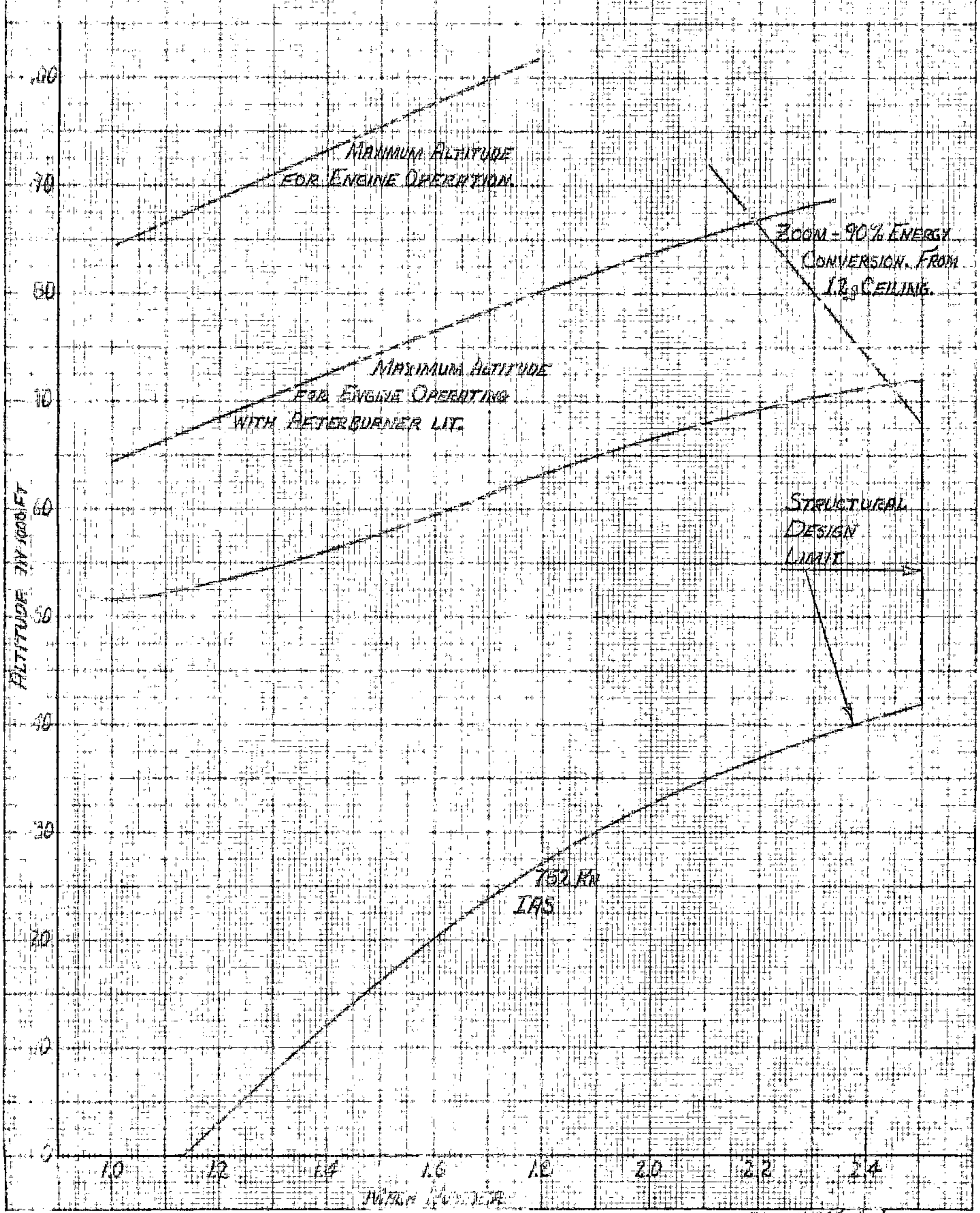

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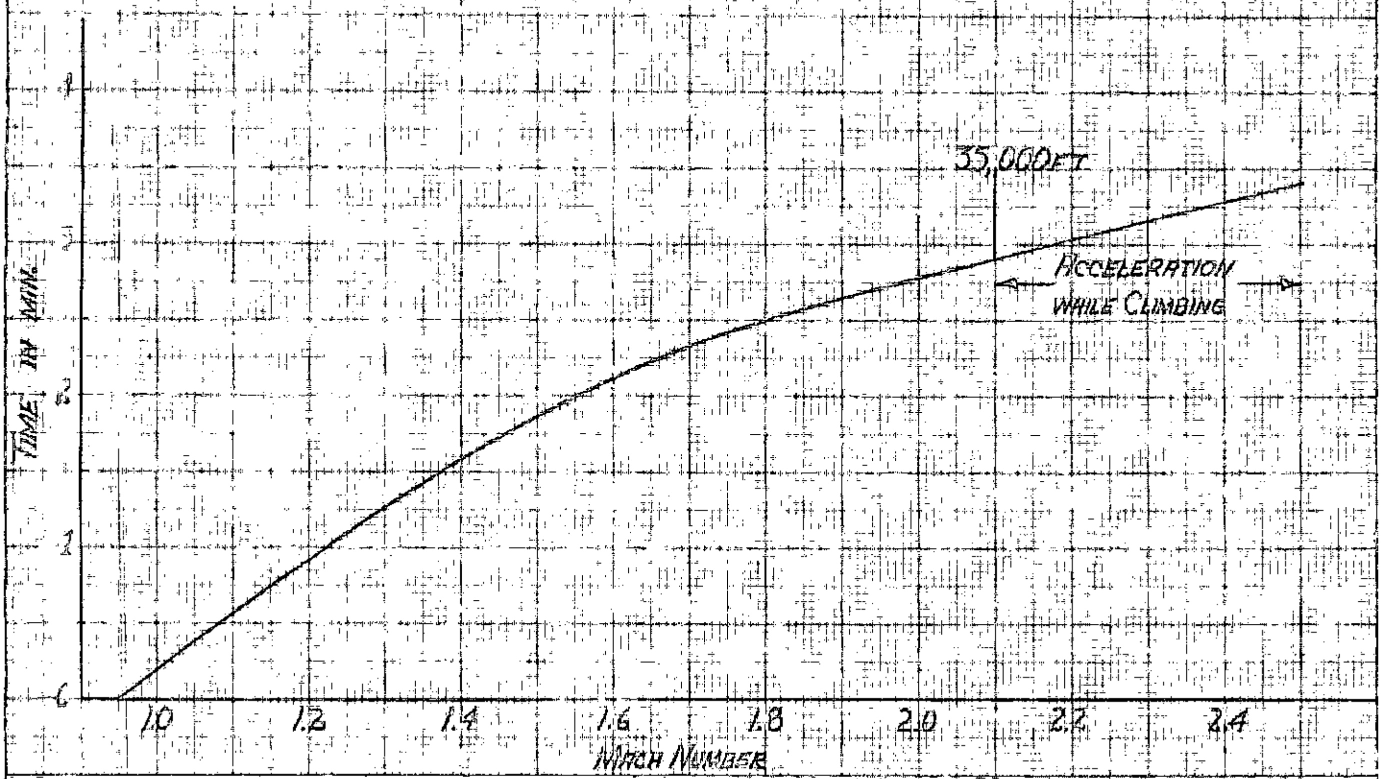
FIG-15
F108 30

A. TIME TO ACCELERATE FROM M=0.95 AT 35,000 FT.
INITIAL WEIGHT =45,500 LB.



B. TIME TO CLIMB TO 50,000 FT. FROM BRAKE RELEASE

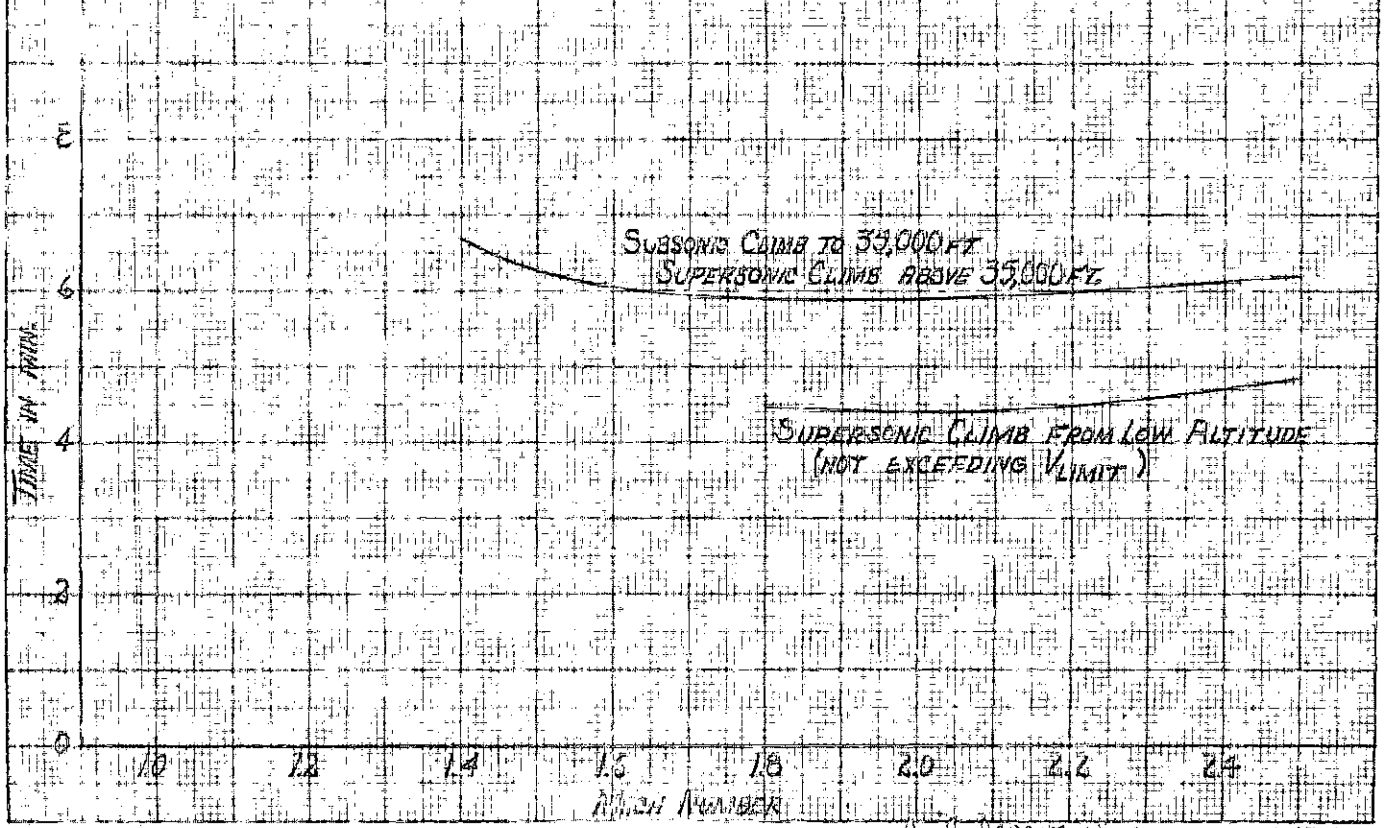


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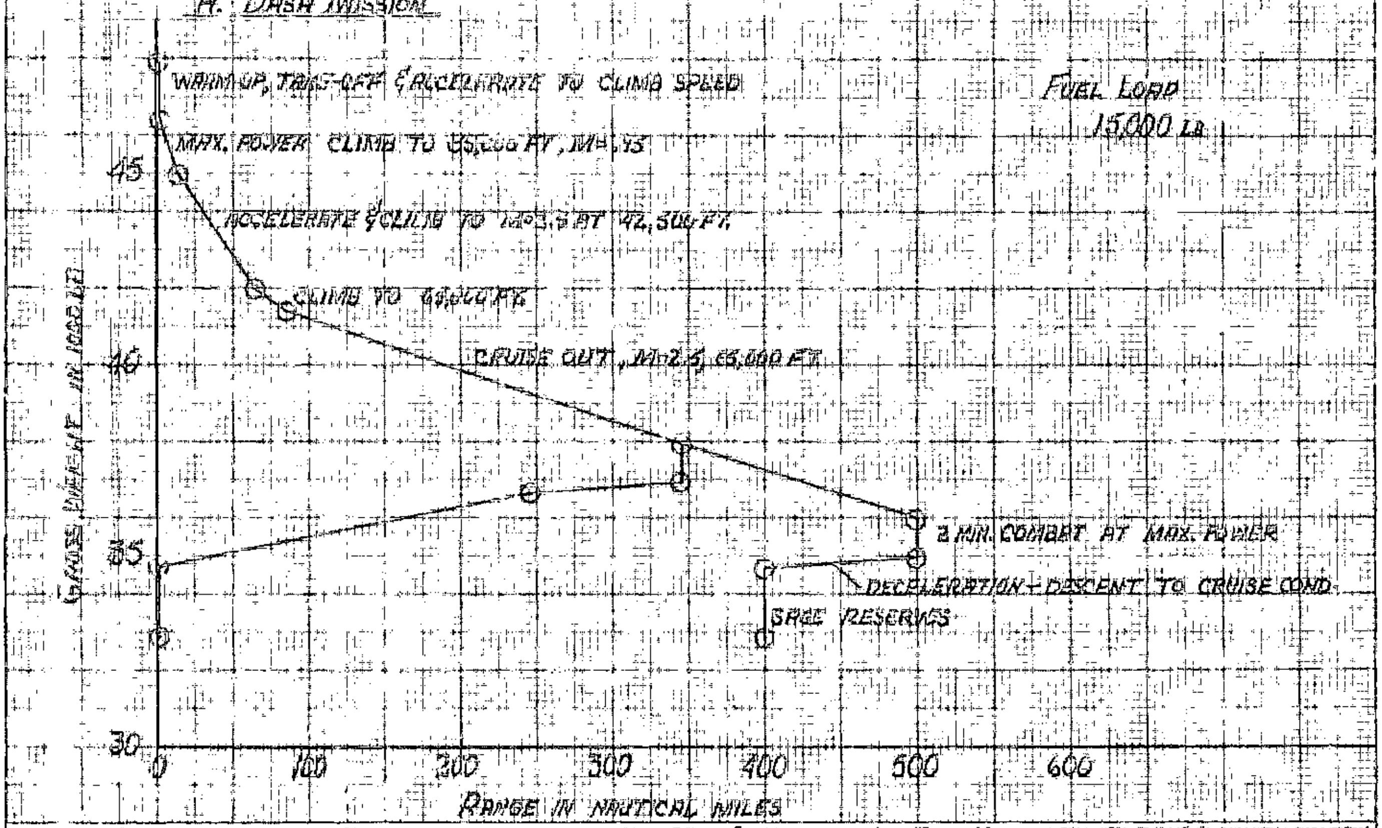
FIG. 4
F-102-50

A. DASH MISSION



B. MAXIMUM RANGE MISSION

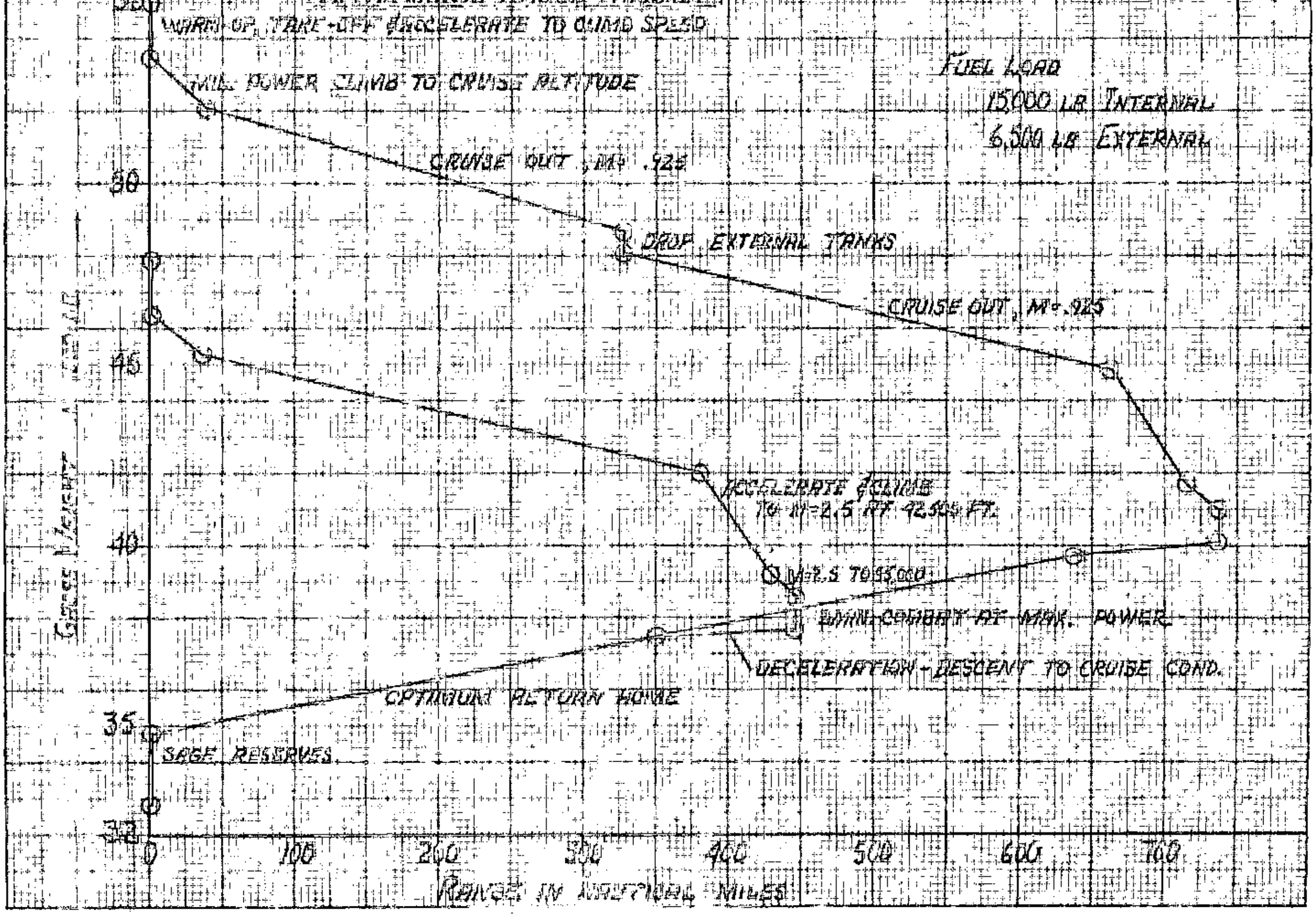


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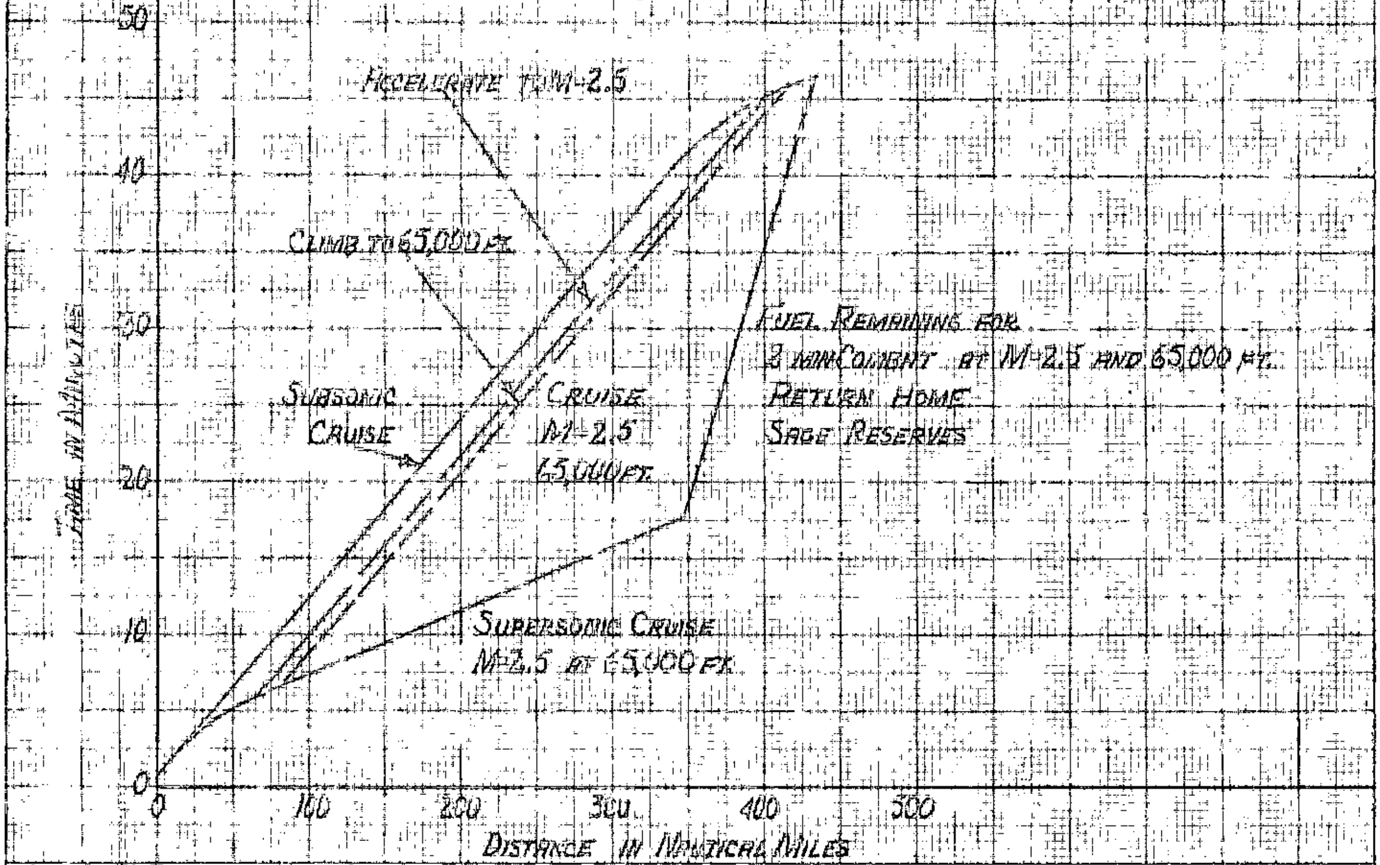
Fig. 75
A-106-30

A. Time vs Distance - Internal Fuel Only



B. Loiter Time vs Combat Distance from Loiter Point

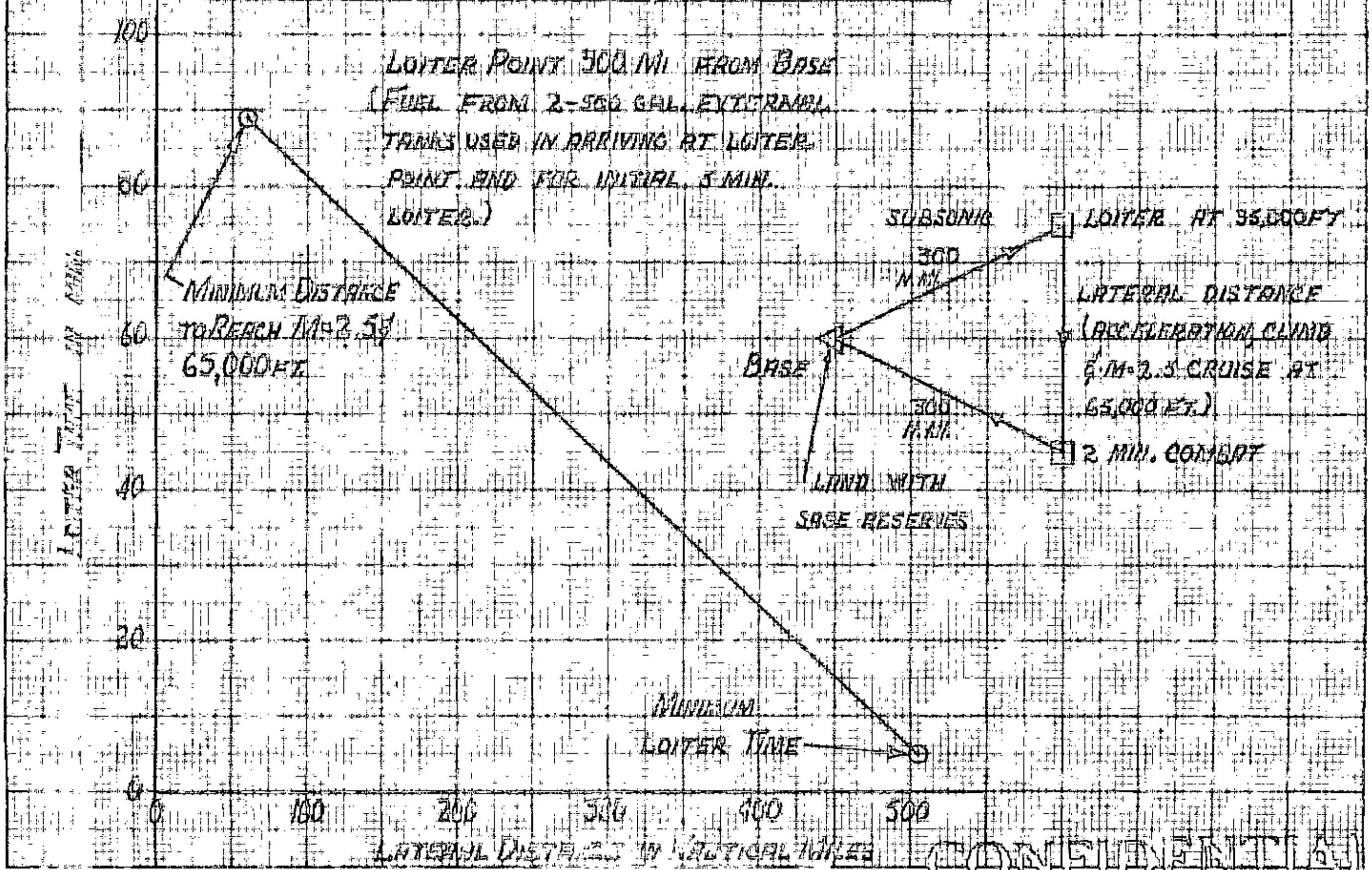


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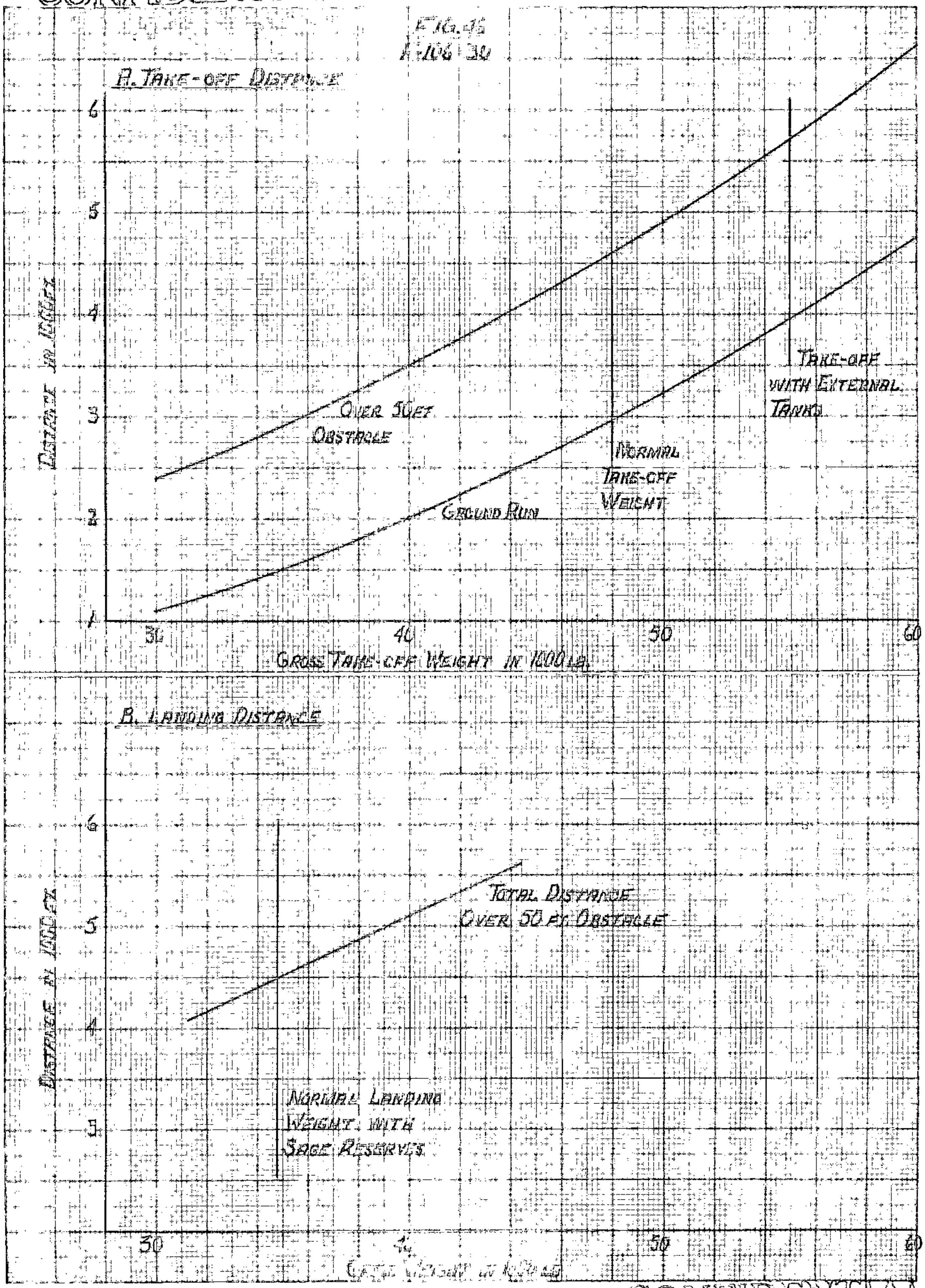
FIG. 45
A-106 30



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LIST OF DRAWINGS

1.	SD-56-08064	Study - Structure, Variable Camber Leading Edge, Advanced F-106 Interceptor
2.	SD-56-08065	Study - Mechanism, Variable Camber Leading Edge, Advanced F-106 Interceptor
3.	SD-56-08075	Three View - Advanced F-106
4.	SD-56-08076	Inboard Profile - Advanced F-106
5.	SD-56-08089	Advanced F-106A Canard
6.	SD-57-08028	Study - 80" Extended Fuselage - Advanced F-106
7.	SD-57-08119	Study - Fully Variable Ejector F-106C (JT4B-22 Engine)
8.	SD-57-08134	Escape Capsule, Canopy Type, Single Place Interceptor
9.	SD-57-08136	Escape Capsule, Cockpit Type, Two Place Pilot and R/O
10.	SD-57-08161	Study - Fixed Ejector - Bleed Doors F-106C - (CJT4B-22 Engine)
11.	SD-57-08174	Escape Capsule, Cockpit Type, Single Place Interceptor
12.	WT-57-82515	General Arrangement - IC Inlet for Lewis 10' x 10' Wind Tunnel
13.	SD-58-06001	Study - Interior Arrangement - F-106C Two-Man Interceptor - J-58
14.	SD-58-06002	General Arrangement F-106C Two Man Interceptor J-58
15.	SD-58-06012	Study - Interior Arrangement F-106-30 Two Man Interceptor
16.	SD-58-06013	General Arrangement F-106-30 Two Man Interceptor

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APPENDIX A

Proposed Test Vehicles Resulting From This Study

Under each of the sections of the study contract, the contractor was to determine modifications to the F-106A aircraft required to flight test the features. Chronologically, the following test proposals have been made:

1. Canard Test Bed for Unstable Aircraft Configuration:

The Advanced F-106, as proposed by Reference 1, was controlled longitudinally by a canard and the aircraft was balanced with the center of gravity unstable; i.e.: c.g. aft of the aerodynamic center. To stabilize the aircraft, an automatic control system as described by Reference 2 was proposed, called a "q-bob" system. The development of this canard control system was of such importance to the flight safety that a modified F-106A test bed was proposed by Reference 3. This configuration was to be tested prior to actual flight of the 'Advanced F-106' configuration.

2. 40" Antenna F-106A (Now Designated F-106C):

A related test bed program, but not specifically a part of the study reported herein, is the modification of two F-106A aircraft to house an enlarged antenna of 40" diameter rather than the current F-106A antenna of 23". This program has since been contracted under ECP 4173 of the F-106 program. References 4 and 5 describe this program.

3. F-106-30 Proposal

No F-106A test bed configuration has been proposed for the F-106-30. The new features of the F-106-30 configuration are not believed to be sufficient development items to require a test bed configuration prior to the time the design is committed to production. The canard, in this proposal, for example, is employed with a conventional stable aircraft configuration, allowing the control system to be similar to present systems. A normal program of prototype and slow production during a 24 month test program (Cook-Craige type plan) is proposed; Reference 6.

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APPENDIX A - REFERENCES

1. F-102/F-106 Improvement Program; ZP-57-08000 dated February 1957.

2. Summary of Canard Control System Studies; DC-8-422 dated 17 January 1957, Revised 25 January 1957.

3. Convair Letter 11-0-645 dated 22 March 1957, Subject: Canard Flight Test on F-106A Airplane, Proposal for.

4. ECP-CAC-F-106-4173.

5. Incorporation of a 40" Radar Antenna in the F-106A/B Aircraft; ZP-8-513 dated 2 December 1957.

6. The F-106-30 Interceptor; ZP-58-06000 dated March 1958.

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APPENDIX B

This appendix consists of a list of references that have been completed through and in association with this study contract.

1. CVAL43 "Wind Tunnel Investigation of the Longitudinal Effects of Horizontal Surfaces Located Fore and Aft of a 60° Delta Wing Model"
2. A-8-108 "Langley 4 x 4 Ft. Supersonic Wind Tunnel, Langley Test 88"
3. A-8-130 "Preliminary Evaluation of Canard Surfaces for Advanced Versions of the F-102 Interceptor"
4. ZA-8-010 "Estimated Performance, Stability and Control Characteristics - Advanced F-102"
5. ZA-8-013 "Preliminary Aerodynamic Analysis of the F-102B with Modifications"
6. A-8-140 "Ames 135 Performance Results"
7. A-8-147 "Results of a Transonic Test on an .04956 Scale F-102B Model in the Southern California Coop. Wind Tunnel ($M = .45$ through 1.20)"
8. A-8-152 "Auxiliary Longitudinal Control Test Program"
9. A-8-153 "Discussion of Flight Tests of a Canard Arrangement of the F-102A Airplane"
10. A-8-169 "Results on a Test on an .0991 Scale Model of the F-106A Airplane in the CVAL 8 x 12 Foot Low Speed Wind Tunnel, Including Ground Plane Effects and Configuration Changes"
11. A-8-174 "Some Stability and Control Characteristics of the Advanced F-106 Airplane With Various Longitudinal Control Devices"

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12. A-8-178 "Data Report on Transonic and Supersonic Wind Tunnel Tests of .04956 Scale Model of the Advanced F-106 at S. C. C. W. T. Wind Tunnel"

13. A-8-179 "Results of Supersonic Tests of a .025 Scale Advanced F-106 Model at J.P.L. 20" Wind Tunnel"

14. A-8-162 "NOL 40 cm. x 40 cm. Supersonic Wind Tunnel WTR 331"

15. A-8-184 "Results of Additional Transonic Wind Tunnel Tests of The F-106A from CWT Tests 528 and 535"

16. ZA-8-502 "Supplementary Aerodynamic Data for Advanced F-106A Airplane"

17. Zero Doc. 8-019 "Estimated Stability Derivatives of the F-106A Airplane with a Canard"

18. A-8-186 "Advanced F-106 Airplane Transonic Test CWT 579 of 27 December 1956"

19. Aero Doc. 8-003 "Preliminary Drag Buildup of Advanced F-106 with Aft Horizontal Tail"

20. A-8-196 "Advanced F-106 Airplane Low Speed Test CVAL 230 28 January - 1 February 1957"

21. A-8-197 "Estimated Performance Characteristics of F-106A. Canard Flying Test Bed"

22. A-8-203 "Study of the Aerodynamic Effects of Canard Geometry and Body Fineness Ratio - Advanced F-106 Airplane"

23. A-8-207 "Preliminary Wing Area Study for the Advanced F-106 Airplane"

24. A-8-210 "Stability Derivatives of the Advanced F-106 Airplane"

25. CVAL 1816 "Additional Low Speed Wind Tunnel Tests of a .0991 Scale Model of the F-106 A Airplane to Determine the Characteristics of a Horizontal Tail and a Canard"

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26. A-8-212 "Advanced F-106 Alternate Engine Studies"

27. A-8-215 "Results of 1/20 Scale Advanced F-106 Test, CWT 608, 15 - 17 May 1957"

28. A-8-217 "Comparison of Ceiling Performance of Canard and Tailless and Canard Arrangements of the Advanced F-106 Airplane"

29. A-8-223 "The Effect of Stability Margin Variation on the Tailless and Canard Arrangements of the Advanced F-106 Airplane"

30. A-8-225 "Configuration Comparisons for the Advanced F-106 Based on 1/20 Scale Model Tests in the Southern California Coop. Wind Tunnel"

31. CVAL 204 "Low Speed Wind Tunnel Tests of a .04956 Scale Model of the F-106A Airplane with a Horizontal Canard"

32. A-8-231 "Results of Low Speed Tests on an 0.4 Scale Canard Nose Model of the Advanced F-106 at CVAL Wind Tunnel"

33. Aero Doc. 8-027 "Comparison of J58 and Twin GE X279E Engine Installations in an Advanced F-106 Airplane"

34. A-8-241 "Performance Comparisons of Three Advanced F-106 Airplanes Having Different Engine and Inlet Arrangements"

35. DC-8-422 "Summary of Canard Control System Studies"

36. DC-8C-103 "Summary of Nonlinear Studies on Q-Bob Control System"

37. DC-8C-105 "Derivation of Linearized Equations of Motion for Q-Bob Stabilized Airframe"

38. DC-8C-106 "Additional Nonlinear Studies on Q-Bob Control System"

39. DC-8C-107 "Moving Cockpit Investigations of 'Bring Home' Conditions"

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40. DC-8C-112 "Electric Stick Control System for the Advanced F-106"

41. DC-8C-115 "Derivation of Advanced F-106 Longitudinal Stability Derivatives Used in Q-Bob Studies"

42. DC-8C-114 "Final Nonlinear Studies on Q-Bob Control System"

43. DC-8C-109 "Maneuvering Flight Study of the Canard Airplane - Advanced F-106"

44. DC-8C-110 "Evaluation of the Advanced F-106 Canard Airplane With Q-Bob During Landing Roll"

45. DC-8B-159 "Analysis of the Structural Flexibility Effects on the Longitudinal Control System of Two Advanced F-106 Configurations"

46. CONTR-8-20-725 "Advanced F-106 Feel System Study"

47. DC-8-417 "Results of the Convair Electric Stick Program"

48. SG-99 "Reliability of Electric Control Systems"

49. DF-8B-141 "Supersonic Tests of 1/8 Scale Flutter Models of 45° Delta and 52° Swept Canard Surfaces"

50. DF-8B-148 "Low Speed Tests on a Semi Span 1/4 Scale 52° Swept Canard Flutter"Model"

51. ZV-8-503 "Wind Tunnel Investigation of the Effects of a 45° Delta Canard on the Aerodynamic Characteristics of a 60° Delta Wing"

52. CVAL-241 "Low-Speed Wind Tunnel Tests of a 1/5 Scale Model of a Pilot Escape Capsule"

53. SCCWT-622 "Transonic Wind Tunnel Tests of the Sting-Mounted 0.20 Scale Convair San Diego Pilot Escape Capsule"

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54. Aero Memo A-Misc-35 "Wind Tunnel Test Results of a 1/5 Scale Model Pilot Escape Capsule with Twin Boom Stabilizing Surfaces"

55. Aero Memo A-Misc-29 "Two-Man Estimated Capsule Estimated Aerodynamic Coefficients"

56. ZA-276 "Summary of Aerodynamics Studies of a Pilot Escape Capsule"

57. ZP-57-08000 "F-102, F-106 Improvement Program"

58. ZP-8-507 "Advanced F-106 Study, First Quarterly Progress Report"

59. ZP-8-509 "Advanced F-106 Study, Second Quarterly Progress Report"

60. ZP-8-516 "Advanced F-106 Study, Third Quarterly Progress Report"

61. ZP-58-06000 "F-106-30 All-Weather Interceptor"

62. T-106Adv-1 "Comparison of the Estimated Performance of the Advanced F-106 Airplane Propulsion System with Two Types of Diffuser Inlets"

63. T-106Adv-2 "Preliminary Performance Estimate for the Advanced F-106A Airplane"

64. T-106Adv-4 "Determination of Size and Coefficients for Mass Flow Measuring Nozzles to be Used in the 1/12 scale Twin Duct Model Having Internal Compression Inlets"

65. T-106Adv-5 "Design of Inlet Control System Simulation on the 1/3 Scale Wind Tunnel Model of the Advanced F-106A"

66. T-106Adv-6 "Pre-Test Summary Data for the Advanced F-106 Static Ejector Test Program"

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67. T-106Adv-7 "Proposed Test Program for the Single Inlet Internal Compression Wind Tunnel Model of NAMTC, Point Magu, California.

68. T-106Adv-8 "Tentative Test Schedule for OAL Pre-compressor Cooling Tests"

69. T-106Adv-9 "Thermodynamics Laboratory Pre-compressor Cooling Tests"

70. T-106Adv-10 "Proposed Test Program for the 1/12 Scale Advanced F-106A Inlet Model in the 4' x 4' Wind Tunnel at Langley Field Virginia"

71. F-106Adv-11 "Performance Calculation for Turbojet Engine Utilizing Pre-compressor Evaporative Cooling"

72. T-106Adv-12 "Pre-compressor Cooling Tests to be Conducted at Arnold Engineering Development Center"

73. T-106Adv-14 "Investigation of the Effects of Canard Location on the Performance of An Internal Compression Inlet"

74. T-106Adv-15 "Performance of a Circular Internal Compression Side Inlet"

75. T-106Adv-16 "General Information Relating to Tests of the 1/12 Scale Advanced F-106 Inlet Model"

76. T-106Adv-17 "Wind Tunnel Investigation of Internal Compression Inlets at Mach Numbers 1.6 - 3.5"

77. T-106Adv-18 "Pre-Test Summary Data for the Advanced F-106 Ejector Afterbody Test Program at Lewis 8' x 6' Wind Tunnel"

78. T-106-Adv-19 "A Theoretical Calculation on the Effects of Water Injection Cooling"

79. T-106Adv-20 "Proposed Test of Advanced F-106 Full-Scale Inlet Controls Model in the Lewis 10' x 10' Unitary Wind Tunnel"

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80. T-106Adv21 "Completely Variable Ejector Performance"

81. T-106Adv22 "Feasibility of Using Bleed Air from an Engine with Pre-Compressor Cooling for Operation of Auxiliary Equipment"

82. T-106Adv-23 "Proposed Test Program for Phase II of the Test of the 1/12 Scale Adv. F-106 Inlet Model in the 4' x 4' Unitary Plan Wind Tunnel, Langley Field, Virginia"

83. T-106Adv-25 "Alternate Engine Study for the Advanced F-106, Considering the J79 = (X207), J89 and J91 Engines"

84. T-106Adv-26 "Estimated Performance for the Installed J93-1 Engine in these Versions of the Advanced F-106"

85. T-106Adv-29 "Analytical Investigation of the Dynamics and Steady State Gains of the Internal Compression Inlet"

86. T-106Adv-30 "Instrumentation Requirements for Full Scale Internal Compression Inlet Controls Model Test in NACA Lewis 10' x 10' Supersonic Wind Tunnel"

87. T-106Adv-31 "Preliminary Summary of Results from Static and Wind Tunnel Tests for the Adv. F-106 Ejector Configuration"

88. T-106Adv-33 "Pre-Compressor Cooling Tests with a J57 Turbojet Engine"

89. T-106Adv-34 "Structural and Equipment Cooling Study Summary Report"

90. T-106Adv-35 "Installed Performance of the JT4B-22 Engine with Partial PCC in the Advanced F-106"

91. T-106Adv-36 "Pre-Compressor Cooling Tests with a J75 Turbojet Engine - Phase II"

92. T-106Adv-37 "Pre-Compressor Cooling Tests with a Straight Duct - Phase I"

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93. T-106Adv-38 "Pre-Compressor Cooling Tests with a Bifurcated Duct - Phase III"

94. T-8/30-1 "F-106C Air Conditioning System Early Pre-Design Study"

95. T-8/30-2 "Characteristics of J58 Engine (P&W JT-11-5A)"

96. T-8/30-2 "Addendum"

97. T-8/30-3 "Installed Performance of J58 Engine in the F-106-30"

98. T-8/30-4 "Selection of Fuselage - Canard - Inlet Configuration for the F-106-30 Airplane"

99. T-8/30-5 "F-106-30 MA-1 Advanced and Pulse Doppler Fire Control Systems Cooling Requirements"

100. T-I-338 "Pre-Compressor Cooling Studies for the Pratt and Whitney JT4B-21 Engine"

101. RT-360 "Water Injection Pre-Compressor Cooling Tests"

102. T-G-68 "An Analytical Study of Inlet Bypass Drag and a Comparison with Additive Drag"

103. ZA-8-506 "Advanced F-106 Summary Report, Aerodynamic Features, Pressure Distribution, Control System and Flutter Data"

104. ZP-8-519 "Pilot's Escape Capsule - Advanced F-106 Airplane Study Report"

105. T-I06Adv-32 "Summary Report - Pre-Compressor Cooling and Related Studies"

106. ZP-8-520 "Advanced F-106 Summary Report of Design Studies"

107. DF-8B-145 "Summary of Canard Flutter Model Wind Tunnel Tests".

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108. DC-8C-111 "Dynamic Analysis of Pilot Escape Capsule"

109. ZU-030 "A Preliminary Proposal for the Development of a Control System for Internal Compression Inlets"

110. DC-8B-165 "Dynamic Analysis of a Shock Position Control System for an Internal Compression Inlet"

111. DC-8B-165 Addendum I, "Addendum to an Advanced F-106 Internal Compression Inlet Control System Dynamic Analysis"

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Binding Instructions

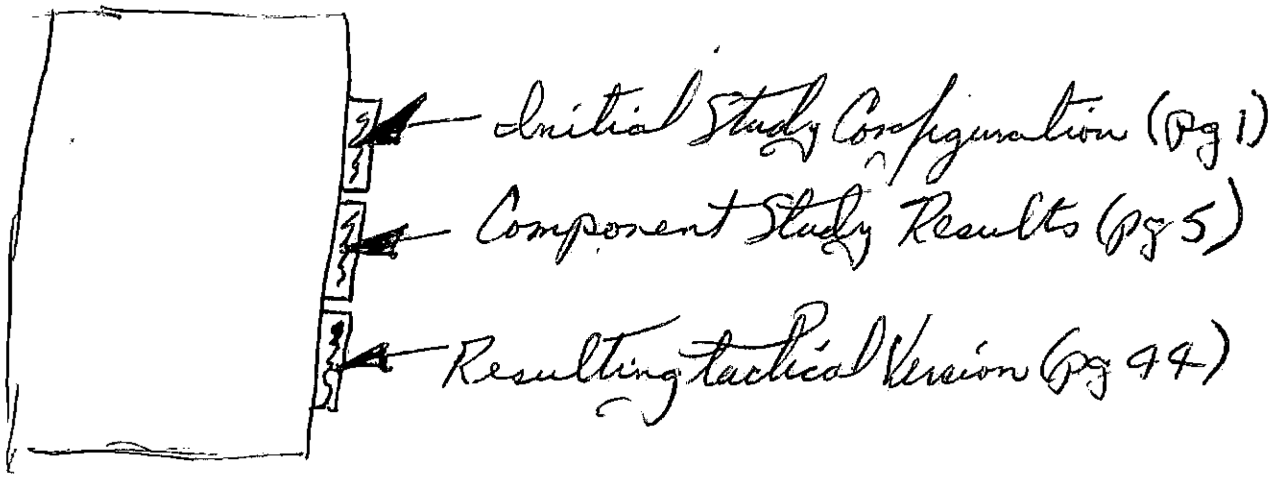
The 16 drawings, numbered 1-16 in upper right hand corner are inserted in order from page 109 to pg 124 immediately following the List of Drawings (pg 108).

The extra sheet prior to the title page is used to prepare tabs for the bound report. The tabs are used as follows: (TCX PAPER)

Initial Study Configuration is applied towards the top of page 1

Component Study Results is applied ~~towards~~ in the Center of page 5.

Resulting Tactical Version is applied towards the bottom of page 44.


Initial Study Configuration (Pg 1)
Component Study Results (Pg 5)
Resulting tactical Version (Pg 44)

Initial Study Configuration	Component Study Results	Resulting Tactical Version
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Attach to Pg 1	Pg 5	Pg 44

ZP-8-046
ZP-8-046

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-046
DATE Nov. 12, 1956
MODEL F-102A

TITLE

A REPORT ON
INFLIGHT REFUELING POTENTIALITIES WITH THE F-102A INTERCEPTOR AS A RECEIVER AIRPLANE

ZP-8-046

PREPARED BY P. W. Snijders

GROUP Preliminary Design

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INTRODUCTION:

The purpose of this report is to invite the attention of USAF personnel to the employment of the F-102A interceptor as a receiver airplane with inflight refueling.

A study was made to determine the feasibility of modifying the F-102A to incorporate inflight refueling using the probe and drogue system. The study covered various possible locations for installation of retractable probe structures and the connection of the probe with the normal pressure refueling adapter supplying the fuel system of the F-102A airplane.

In addition, the possibility of installing the Buddy System refueling package was studied, using the F-102A as a tanker aircraft.

SUMMARY:

This study shows that a retractable refueling probe can be installed on the F-102A airplane with a minimum of structural changes.

The location immediately forward of the windshield, (Position 5 on Drawing SD-56-08071), affords the pilot an unobstructed view of the refueling operation. This location causes no interference with the electronic compartments. No aerodynamic feasibility tests have been made.

The weight increase for the F-102A as a receiver aircraft is approximately 65 lbs., including hydraulic actuator, doors, fuel lines, couplings, fittings, clamps, etc.

If the Buddy System is employed, the F-102A as a tanker aircraft will have a weight increase of approximately 300 lbs.

RECOMMENDATIONS:

1. It is recommended that operational analysis studies be made to determine the performance improvements of the weapon system that can be obtained by the employment of inflight refueling.

2. A simulated probe should be installed in a fixed position on an F-102A to determine the aerodynamic interference effects during the refueling operations.

DISCUSSION:

The trailing tube or hose of the tanker airplane carries a standard type MA-2 reception coupling and cone conforming to U.S.A.F. Drawing No. 55C9062 and Spec. MIL-C-25162, which the F-102A receiver airplane engages with a type MA-2 nozzle (probe), U.S.A.F. Drawing No. 55C9061 and Spec. MIL-N-25161.

Upon engagement, refueling starts automatically.

Fuel into the fuel system of the F-102A receiver-airplane is transferred at the rate of 200-250 g.p.m. by pumps or by jet engine bleed air pressure via the refueling package of the tanker airplane.

DRAWING - SD-56-08070 - Inflight Refueling F-102A - 3 View
(various locations - not recommended)

POS. 1 - Shows a conventional non-retractable type MA-2 nozzle-probe located forward of the windshield of the F-102A.

Although a minimum of changes in structural design in the aircraft are involved and the pilot also has an excellent view of the probe and drogue connection during inflight-refueling, a retractable probe is preferred for better flight-characteristics.

POS. 2 - Shows a retractable telescoping type MA-2 nozzle-probe located in the nose section of the F-102A.

Due to interference with the compact electronic system in this nose section, several structural problems are to be overcome to make this design possible.

POS. 3 - Shows a retractable telescoping type MA-2 nozzle-probe located in the top of the fuselage, behind the canopy of the F-102A airplane. The pilot has to look upward in a 60° angle to view the drogue and probe connection.

Due to the 0.9 Mach speed of the tanker and receiver airplanes during inflight refueling; a high bow wave effect is anticipated. The design of this retractable telescoping type nozzle-probe structure as well as the modification on the airplane-structure are complicated by this high-bow wave effect.

POS. 4 - Shows a conventional non-retractable standard type MA-2 nozzle-probe located in the leading edge of the wing of the F-102A airplane.

To decrease wave effects at this point during high speed inflight refueling the length of the probe has to be increased to approximately 8 feet.

The pilot has to look backwards to pinpoint the drogue and probe connection.

DRAWING - SD-56-08071 - Inflight Refueling F-102A - Inboard Profile (Recommended Locations)

POS. 5 - Shows a retractable type MA-2 nozzle-probe in the upper panel assembly of the nose section of the F-102A. The pilot has an unobstructed view in all conditions.

With the exception of a small well in the top shelf of the forward electronic compartment and a fuel-drain line connection in the same, there is no interference with the electronic packages layout.

POS. 6 - Shows a retractable hinging type MA-2 nozzle-probe at the same location. Instead of the small well in the top shelf of the forward electronic compartment, a fairing on top of the nose section is shown.

Both configurations are possible for Pos. 5 and Pos. 6.

DRAWING - SD-56-08072 - Inflight Refueling F-102A - Retractable Probe

This drawing shows the more detailed retractable probe as described in Pos. 5.

This design shows the most feasible location for the retractable probe, while a minimum of structural changes in the aircraft are involved.

DRAWING - SD-56-08073 - Inflight Refueling F-102A - "Buddy System"

This drawing shows the possibility of using the F-102A interceptor as a tanker aircraft. The center armament bay provides ample space for the retractable tube system of the inflight refueling package.

Recognition lights are provided to give the receiver pilot indication of proper probe alignment and a warning if the extremities of the safe alignment range are approached.

In emergency cases the entire retractable tube system can be jettisoned.

Another F-102A interceptor is shown as the receiver airplane and is able to refuel from any other tanker aircraft, provided the probe and drogue system is used.

Note:

Although it is technically possible to install refueling probes in the nose section of the external fuel tanks, this is not recommended for the following reasons:

Considerable redesign of the fuel system of the F-102A would be required since the present tanks are not included in the pressure refueling system and are employed for ferry missions only.

The external tanks as well as the wing connections would have to be redesigned to withstand the "engagement" and "break away" forces during inflight refueling.

The receiver pilot has to depend on his depth-perception to engage the probe and drogue connection.

Jettisoning of the external fuel tanks including the refueling probe automatically limits the inflight refueling capability of the aircraft.

ZP-M-041

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REPORT ZP-M-041

DATE 5 January 1959

MODEL SD-58-60

TITLE

PERFORMANCE DATA REPORT

NAVY/AIR FORCE EXPENDABLE (U)
POWERED TARGET

CONVAIR MODEL SD-58-60

REPORT ZP-M-041
DATE 5 Jan 1959

Prepared Under Navy Bureau of Aeronautics
Request for Proposal No. EN11-2517-59

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ABSTRACT

An analysis of the aerodynamic performance characteristics of an expendable, liquid rocket powered target designed to meet the requirements of [illegible] MS-11-2517-59 Request for Proposal has been made and the results presented herein. A vehicle with a launch weight of 714 pounds and a usable fuel weight of 419 pounds will have the required endurance at M = 2 at 70,000 feet altitude of five minutes, and at M = 1.5 at 70,000 feet altitude of eight minutes when launched at M = .9 at 35,000 feet altitude. Performance data were based on estimated lift and drag characteristics of a 72.9° sweep delta wing configuration, including trim drag for a 10% static stability margin, and engine performance based on an average specific impulse of 245 seconds during boost and climb and an I_{sp} of 235 seconds at maximum cruise altitude.

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List of Symbols

		Units
AR	aspect ratio	-
b	theoretical wing span	ft.
$\bar{c}$	mean aerodynamic chord	ft.
C_D	drag coefficient = D/qS	-
C_L	lift coefficient = L/qS	-
C_N	normal force coefficient = C_L at small angles of attack	-
c.g.	center of gravity	-
d	body diameter	in.
D	drag	lbs.
g	acceleration due to gravity	ft/sec.^2
h	altitude	ft.
I_{SP}	specific impulse	sec.
L	lift	lbs.
M	Mach number	-
q	free stream dynamic pressure	lbs./ft.^2
R/C	rate of climb	fps
S	wing reference area	ft.^2
T	thrust	lbs.
t	time	min.
W	weight	lbs.
α	angle of attack	deg.
μ	mass ratio = $W_{initial}/W_{final}$	-

SUMMARY

This report presents the aerodynamic performance characteristics and methods of analysis used in the study of an expendable, liquid rocket powered target vehicle to meet the requirements of PR EN-11-2517-59 Request for Proposal.

The basic configuration of the proposed vehicle is shown in Figure I and Table I. The basic performance of this vehicle is:

Endurance at M = 2.0 at 70,000 ft., 5 min. 40 sec.

at M = 1.5 at 70,000 ft., 8 min. 9 sec.

This vehicle meets or exceeds the performance requirements.

INTRODUCTION

The performance objectives for the target were defined in terms of maximum speed, endurance at high altitude, and a region for satisfactory launch and flying ability. The maximum speed required is from M = 2 to M= 2.2. The endurance required is 5 minutes at M = 2 at 70,000 feet altitude and 8 minutes at M = 1.5 at 70,000 feet altitude when launched from 35,000 feet at M = .8. The launching region is defined by M = .4 to .95 at an altitude of 5,000 feet to M = .8 to 2.0 at 35,000 feet with the ability to fly straight and level within this region.

An expendable target must use a simple powerplant in order to achieve reliability with low cost. To this end, a possible choice of a ramjet or rocket motor, both using liquid propellants, was considered. However, the necessity of providing the ramjet target with a separate boost system to allow operation at Mach numbers less than about .8 and the possibility of an extended development program to achieve operation at M = 1.5 at the 70,000 ft. endurance altitude made this choice undesirable. Thus, the liquid rocket motor appeared most feasible.

Study of the target was concentrated on the liquid rocket configuration to design the vehicle which fulfills the requirements. The results of the performance analysis and a discussion of methods used in determining performance are presented in this report.

1. Target Geometry, Weight and Engine Data

The target dimensions, weights and power plant data are shown in Tables I and II. These data represent the values used in determining the aerodynamic characteristics and performance except as indicated in the text. Wing characteristics do not include the tip cutoff. Figure I presents a three-view sketch of the target.

Table I

Tabulated Geometry

Wing:

Area	19.65 ft^2
Span	4.84 ft
Mean aerodynamic chord	5.41 ft
Aspect ratio	1.19
Taper ratio	0
Tip chord	0
Root chord	8.13 ft
Incidence angle	0°
Dihedral angle	10°
Leading edge sweepback	72.9°
Trailing edge sweep forward	10°
Elevon area	1.75 ft
Distance from 50% $\bar{c}$ to elevon hingeline	2.08 ft
Leading edge diameter	.25 in
Trailing edge diameter	.25 in
Thickness ratio (MAC)	2.67%

Vertical Tail:

Area	2.5 ft^2
Span	1.5 ft^2
Mean aerodynamic chord	1.667 ft
Aspect ratio	.9

Taper ratio	.7
Tip chord	1.373 ft
Root chord	1.918 ft
Leading edge sweepback	35°
Trailing edge sweepback	18.5°
Distance from 50% $\bar{c}$ to 50% fin $\bar{c}$	4.37 ft
Rudder area	.8 ft^2
Leading edge diameter	.0625 in
Trailing edge diameter	.25 in
Thickness ratio	5%

Body: (without nose boom)

Length	18.1 ft
Diameter	13.5 in
Nose length	4.15 ft
Base area (not including motors)	.50 ft^2
Boattail length	3.42 ft

Motor housings:

Length	3.42 ft
Base area	.252 ft^2

Table II

Tabulated Weight & Power Plant Data

Gross weight	714 lbs
Fuel weight	428 lbs
Residual fuel	10 lbs
Empty weight	286 lbs
Max booster motor thrust	920 lbs
Max sustainer motor thrust	180 lbs
Throttled sustainer motor thrust	90 lbs
Specific impulse @ 50,000 ft altitude (all motors)	245 sec.
Specific impulse @ 70,000 feet altitude (all motors)	253 sec.

2. Evaluation of Aerodynamic Characteristics

The lift and drag characteristics of the vehicle were estimated by resolving normal and axial forces into components perpendicular and parallel to the flight path. Emperical or theoretical methods were used, supplemented by experimental data when available. ARDC Model Atmosphere, 1956, was used throughout. All coefficients are based on theoretical wing planform area.

2.1 Buildup of the Lift Curve Slope

Within the normally small angle of attack range used for level flight (the maximum cruise angle of attack is approximately 6 degrees at a Mach number of M = 1.5 at 70,000 feet altitude) the lift curve was assumed to be linear and equal to the normal force curve estimated for small angles of attack. The normal force was determined by obtaining the lift of the body and wing separately and the interference lift of the combination and adding these terms linearly.

Body Lift

An ogive-cylinder body having a nose fineness ratio of 3 and an overall fineness ratio of 12.5 was used to represent the target body. The actual configuration is slightly longer than the one used for determining aerodynamic characteristics but the additional contribution of the longer body is negligible. Reference 1 presents a second-order shock-expansion theory for obtaining normal force curve slopes at supersonic speeds near zero angle of attack. Experimental correlation is also presented in this reference for a similar configuration (e.g. for an overall fineness ratio of 13) at Mach numbers above 3 to illustrate the applicability of this theory. Reference 2 presents experimental data for this body configuration at M = 1.98.

At subsonic speeds, the slender body value of

$$C_{N_\alpha} = 2 \text{ per radian}$$

was used. A smooth transition from the subsonic to supersonic values was assumed through the transonic region.

Wing Lift

Wing alone lift was obtained from experimental data whenever possible. Reference 3 presents wing data in the supersonic region which required very little interpolation to correct from the experimentally tested aspect ratio's to the aspect ratio of 1.19 used on the target. References 4 and 5 were used to establish subsonic and transonic lift curve slopes for the wing. Linear theory results presented in the latter reference were used to aid in establishing the variation of lift with Mach number.

Interference Lift

The method presented in Reference 6 was used to estimate the carryover effects of the body and wing. These carryover terms are defined from slender body theory and presented in the form

$$C_{L_{\alpha_{BW}}} = K_{BW} \, C_{L_{\alpha_W}}$$

$$C_{L_{\alpha_{WB}}} = K_{WB} \, C_{L_{\alpha_W}}$$

where the interference lift is proportional to the lift on the exposed wing panels joined together. These proportionality factors are given as a function of diameter to span ratios. The theoretical span was used in establishing this ratio.

Configuration Lift

Total lift on the configuration is defined by the sum of the component lifts and the interference effects as

$$C_{L_\alpha} = C_{L_{\alpha_B}} + (K_{WB} + K_{BW}) \, C_{L_{\alpha_W}}$$

It should be noted that the wing lift, in the above expression is based on exposed wing area while the aerodynamic characteristics presented are based on theoretical wing area.

A check was made of the results by applying this technique to a configuration presented in Reference 7 which was tested at M= 2.01. This test configuration was almost identical to the target configuration, having the same nose shape and wing aspect ratio as the target. The major difference was in the ratio of wing to body size. There was excellent agreement between the estimated result and the experiment lift curve slope up to an angle of attack of 6 degrees.

Final results are presented in Figure 2 as C_{L_α} versus M, untrimmed for a rigid airframe.

2.2. Drag at Zero Lift

The drag at zero lift of each component was determined, using the methods described below, and summed to obtain the total minimum drag coefficient, C_{D_0}, presented in Figure 3 as a function of Mach number and altitude. Table III shows the zero lift drag breakdown by component at Mach numbers of 1.5 and 2.0 and at an altitude of 70,000 ft.

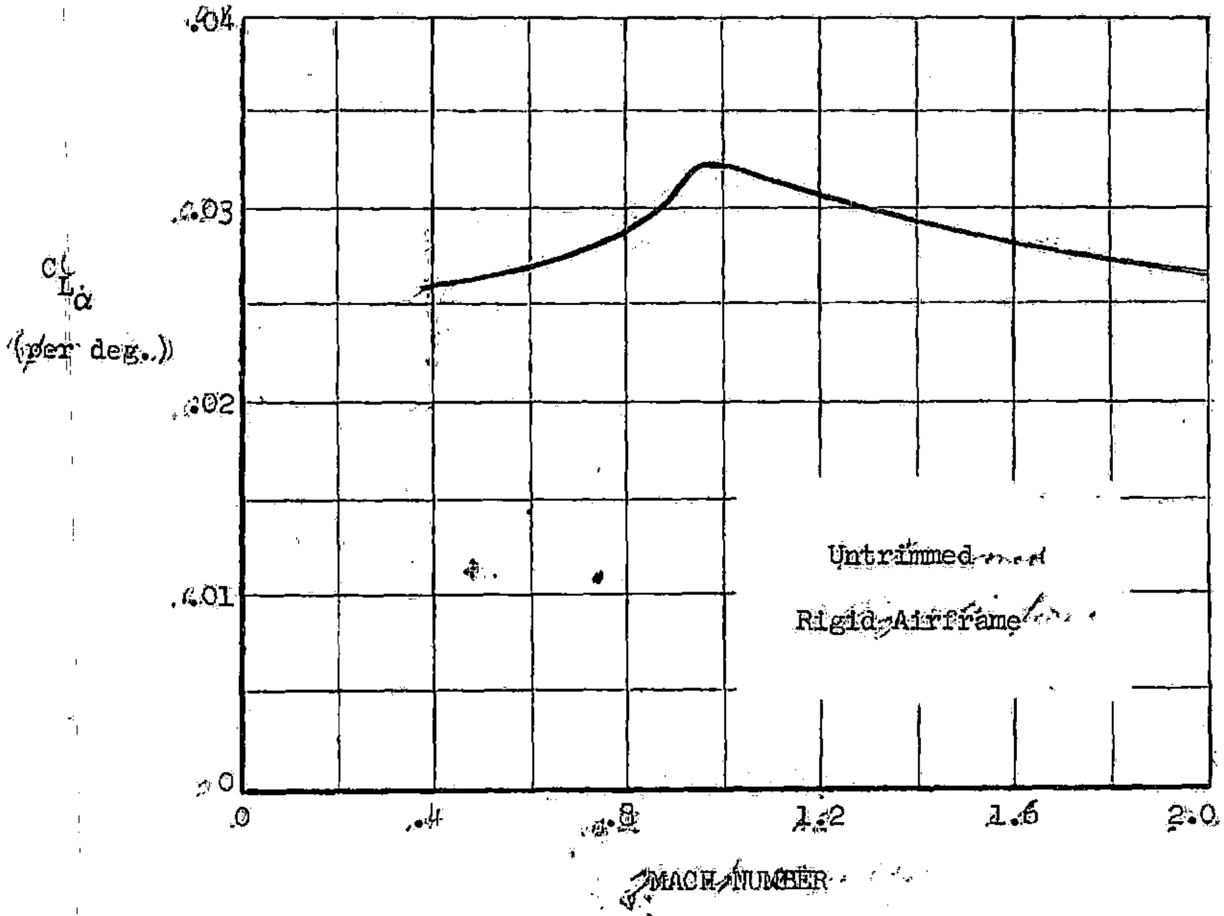


Figure 2 - Lift Curve Slope versus Mach Number

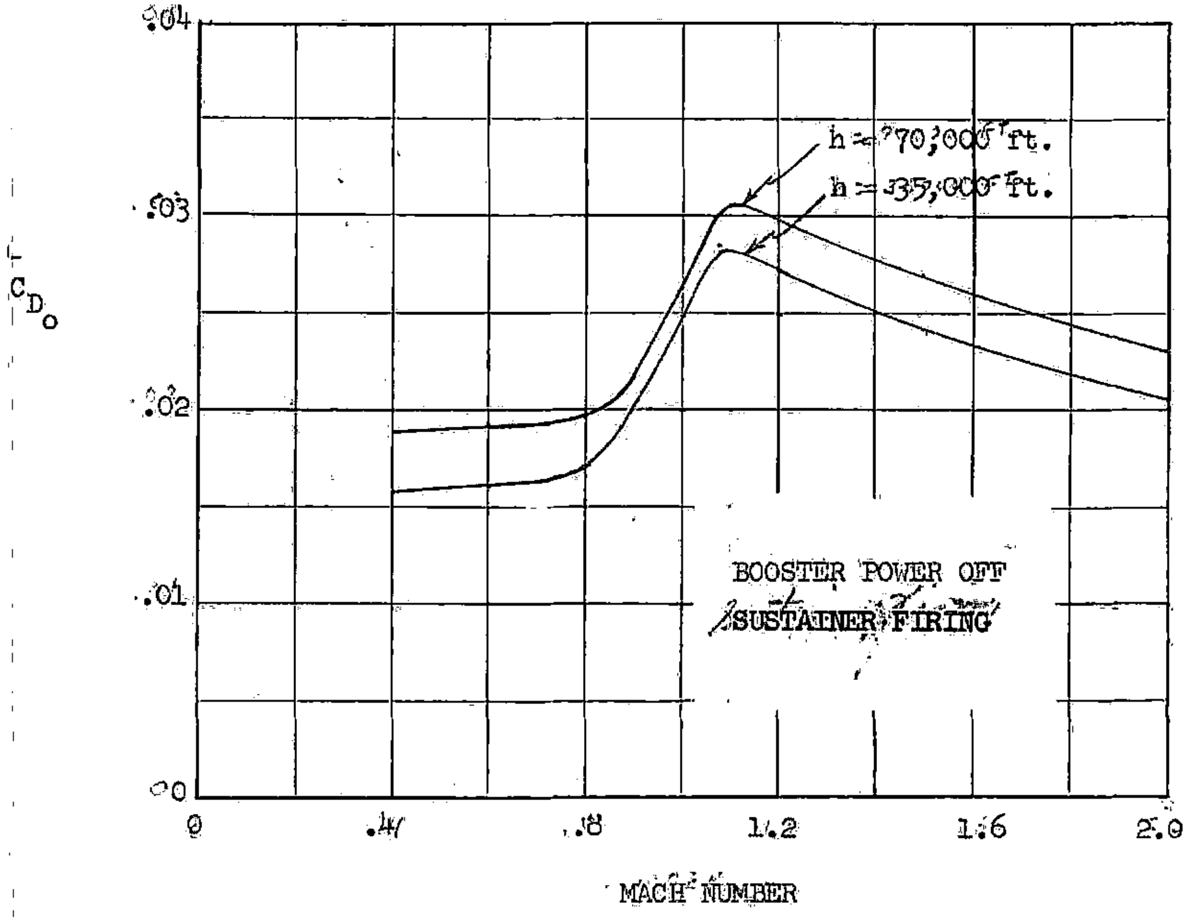


Figure 3 - Drag Coefficient at Zero Lift versus Mach Number

TABLE III

ZERO-LIFT DRAG BREAKDOWN

$S_{Ref} = 20.3\ ft^2$ = Theoretical wing area

Altitude, h = 70,000 ft.

Pressure Drag	M = 1.5	M = 2.0
Nose	.004640	.004150
Wing	.001140	.000733
Wing leading edge	.000244	.000274
Tail	.000733	.000474
Tail leading edge	.001060	.000850
Wing & tail base	.001878	.001437
Body base	.004100	.003590
Booster base	.001110	.000973
Body boattail	.000782	.000611
Burner fairings	.000298	.000259
Total pressure	.015985	.013451
Friction Drag		
Body	.006140	.005500
Wing	.003500	.003008
Tail	.000845	.000733
Burner fairings	.000415	.000357
Total friction	.010900	.009598
Total C_{D_0}	.026885	.022949

Wave Drag

Wave drag on the nose was obtained for a profile representing a compromise between minimum drag and acceptable radome performance. This profile is a tangent ogive with fineness ratio of 3.5. The wave drag was obtained from experimental data on the Tartar missile nose with a fineness ratio of 2.5, and corrected to a fineness ratio of 3.5 by use of data from Reference 8.

Wave drag on the wing was determined from two-dimensional second order theory for double wedge airfoils. Thus, $C_D = \frac{4(t/_c)^2}{\beta}$ where $\bar{c} = \overline{c}$ and t = maximum thickness of $\overline{c}$, minus leading edge thickness. Wing and tail leading edge drag was obtained from data presented in Reference 9 for circular cylinders at angle of attack. The wing leading edge is swept back 72.9 degrees and the cross section normal to the leading edge was assumed circular with a diameter of 1/4 inch. The vertical tail leading edge is swept back 35 degrees with a cross section diameter of 1/16 inch.

Wave drag on the rocket motor fairings was determined using wave drag coefficients found in Reference 10 for a cone of half angle 5 degrees.

Base Drag

Base drag for the body, wing trailing edge, and tail trailing edge was determined from experimental data presented in Reference 11, assuming three-dimensional flow at the body base and two-dimensional flow at the wing and tail trailing edges.

Friction Drag

Skin friction drag was determined for each component of the vehicle for Mach numbers to 2.0 and altitudes to 70,000 ft. using van Driest's equation for turbulent flow on a flat plate assuming Prandtl's mixing length and $T_w/T_\infty = 1.0$ as found in Reference 12.

Drag at Subsonic Speeds and Transonic Drag Rise

Drag at subsonic and transonic speeds are estimated from test data on configurations of similar body size, shape, and surface areas, including unpublished wind tunnel test data on the Convair Tartar Missile.

2.3 Induced Drag

Induced drag was calculated assuming a 10 percent longitudinal static stability margin and the corresponding trim drag penalty found from flight test data on the F-106A interceptor. Thus, the trim drag used amounted to a 36 percent increase in induced drag. The induced drag was defined as the component of the normal force in the drag direction at supersonic speeds and $C_L^2/\pi(\ AR)$ at subsonic speeds

2.4 Lift/Drag Ratio

Lift to drag ratio was computed as $L/D = C_L/C_D = \frac{C_L}{C_{D_0} + C_{D_i}}$, trimmed for 1 g flight at a 10 percent static stability margin.

Figure 4 shows the variation in lift to drag ratio vs. altitude for two weights representing nominal start of cruise weight and burnout weight at the end of cruise.

The actual static stability margin will be less than 10 percent for most flight and weight conditions. Thus, the lift to drag ratios presented are conservative.

2.5 Drag Due to Externally Mounted Target

The drag added to a basic airplane configuration when an external store (in this case, the target) is mounted on it is strongly dependent upon the interference drag induced in the combined configuration. Two items are immediately observed from a cursory look at data involving this drag. First, the interference drag is of a highly unpredictable nature. Second, the magnitude of the interference drag can be varied appreciably by moderate changes in location of the external store on the wing. Reference 13 illustrates these two points quite well. A compilation of experimental data on external stores and nacelle drag through the transonic and supersonic Mach range up to M = 2.0 indicate that interference drag can be favorable at transonic speeds while the minimum interference appears to be zero at supersonic speeds. Throughout the Mach range, however, the maximum interference as shown in Reference 13 appears to be three to four times the drag of the isolated store. This variation is caused by size as well as location changes for the stores used for these experimental results. Due to location changes only, a two to one variation in interference drag is more reasonable.

This brief discussion on drag due to external stores serves to illustrate the possible error of basing performance of an airplane-store configuration upon an estimated drag increment due to the store.

In view of the nature of such a drag estimate, only gross results are presented of the drag increment in pounds versus Mach number caused by carrying two targets externally mounted on the wings of an airplane.

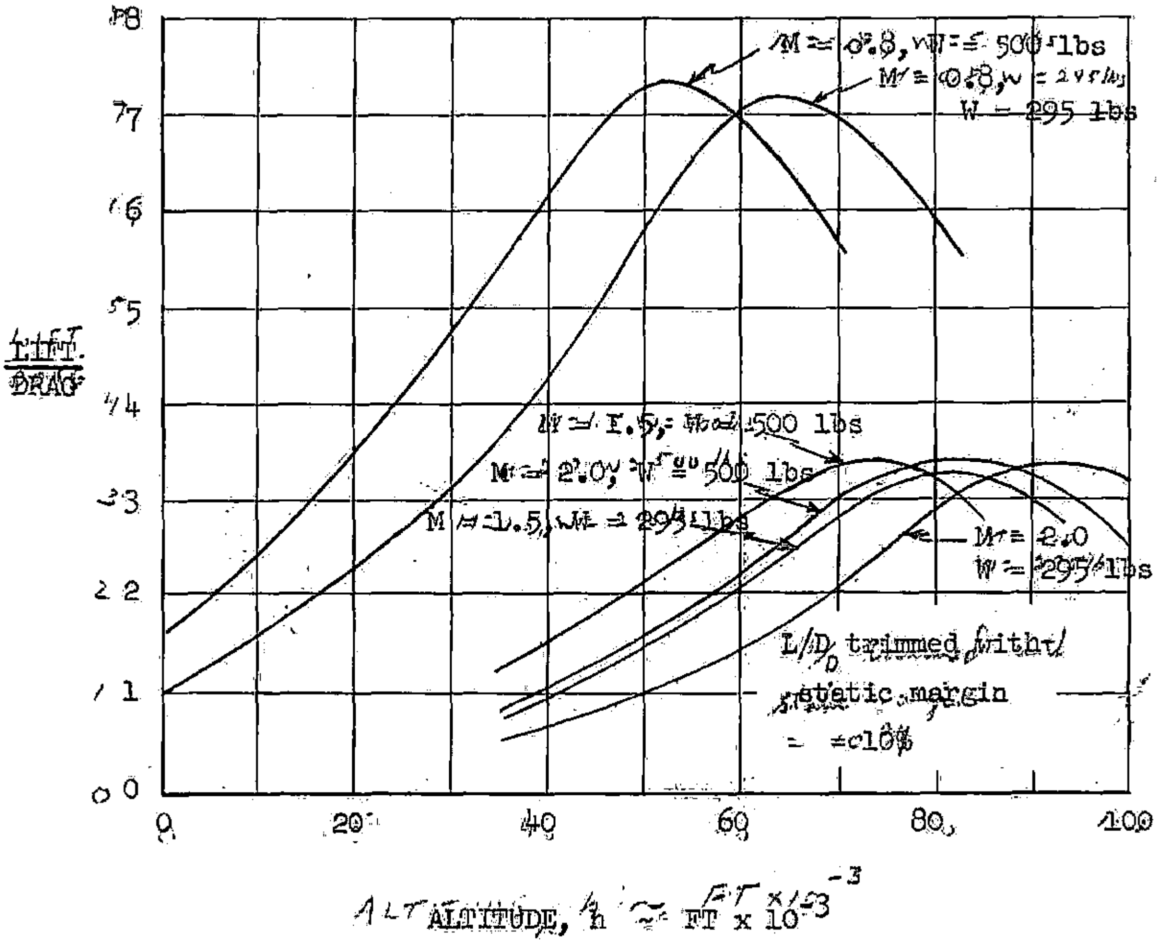


FIGURE 4 LIFT TO DRAG RATIO vs ALTITUDE

Figure 4 - Lift to Drag Ratio versus Altitude

Data used to establish these results were based on unpublished, preliminary flight test results of the effect of carrying external stores on the F-106 interceptor and on NACA data in Reference 13. The flight test data indicated that the drag increment due to the external stores was approximately four times the drag of the isolated store throughout the supersonic Mach range up to M = 2. This result agrees with the maximum value due to externally mounted store drag shown in Reference 13. Therefore, the maximum ratio of mounted store drag to isolated store drag presented in this Reference was used to compute the drag of externally mounted targets. The results are presented in Figure 5 for sea level and 35,000 feet altitudes. No distinction was made as to the type of parent airplane used in that the results represent the maximum drag to be expected. Experimental studies of optimum mounting positions on specific aircraft may reduce the maximum values for that aircraft.

3. Performance

3.1 Power Required versus Power Available

Power required to maintain constant Mach number and altitude at a given vehicle weight is assumed equal to the total aerodynamic drag.

Power available is assumed as a nominal 1,100 pounds total thrust with booster and sustainer operating, and 90 lbs. or 180 lbs. for operation of the sustainer only. The sustainer burns continously at the lower thrust level with intermittent operation at the higher thrust level as required to maintain the desired cruise Mach number. For conditions such as Mach number = 2.0 at 35,000 feet, where drag exceeds maximum sustainer thrust, the booster operates intermittently to maintain the desired average speed.

Figure 6 shows the drag variations with Mach number for altitudes of 35,000 ft. and 70,000 ft.

3.2 Rate of Climb

Rate of climb for the vehicle is quite high since the maximum thrust-to-weight ratio during boost is always greater than one. Maximum rate of climb is limited to the velocity corresponding to design Mach number, which is 2.0. Thus vertical climb at Mach number = 2.0 is possible at altitudes above approximately 45,000 feet at full gross weight. Figure 7 shows unaccelerated rate of climb versus altitude for a full gross weight vehicle at Mach numbers of 0.8, 1.5 and 2.0.

3.3. Speed-Altitude Limits

Maximum speed and minimum speed (defined by a maximum angle of attack of 20°)versus altitude is shown in Figure 8 for the vehicle at full gross weight. Sufficient thrust is available to exceed Mach number 2.0 in level flight at altitudes above approximately 20,000 ft. Also shown on this figure is the minimum speed-altitude relationship at which a 2g maneuver could be made considering the $\alpha = 20°$ limit.

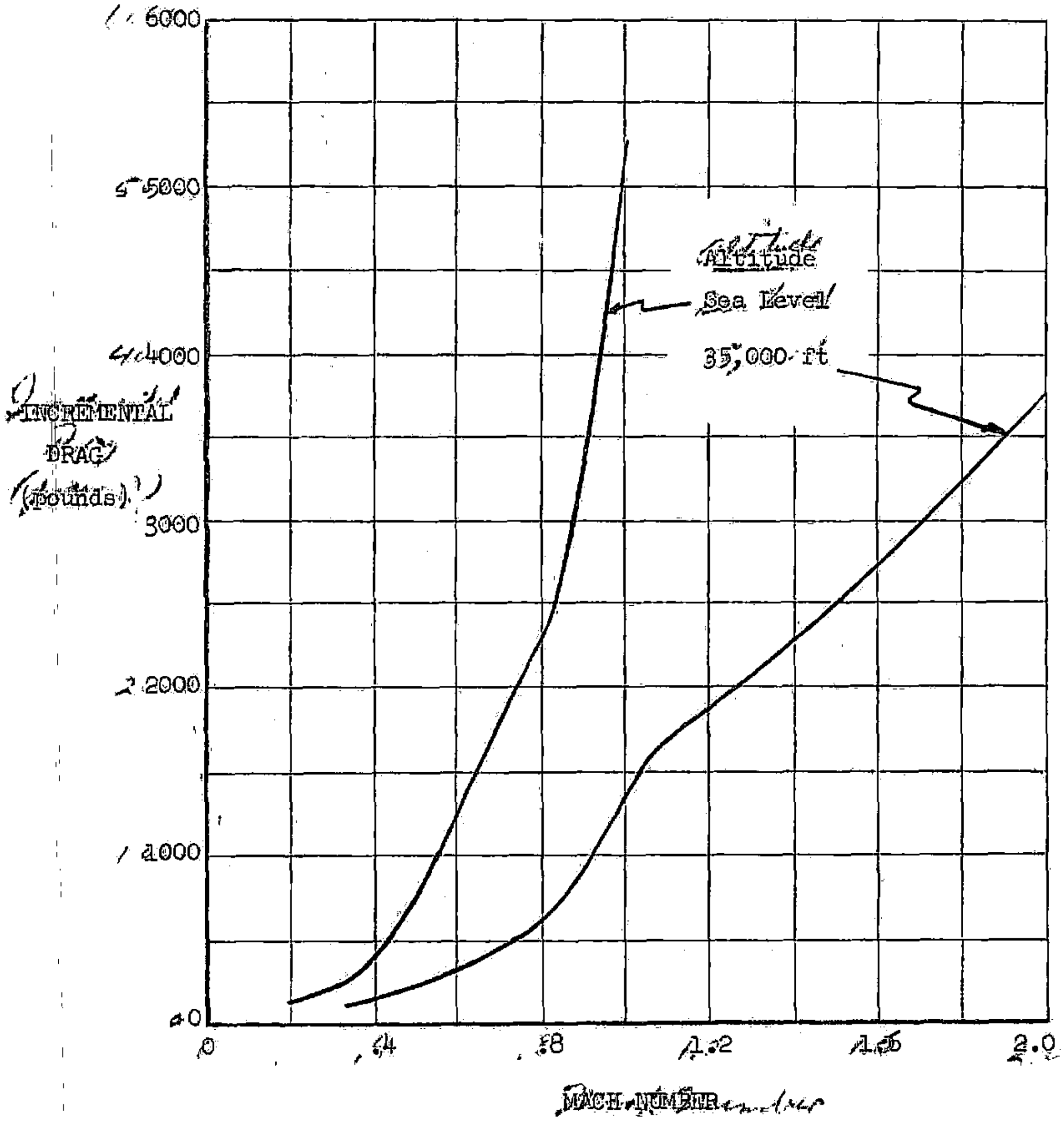


Figure 5 Incremental Drag versus Mach Number Due to Two Externally Mounted Targets.

Figure 5 - Incremental Drag versus Mach Number Due to Two Externally Mounted Targets

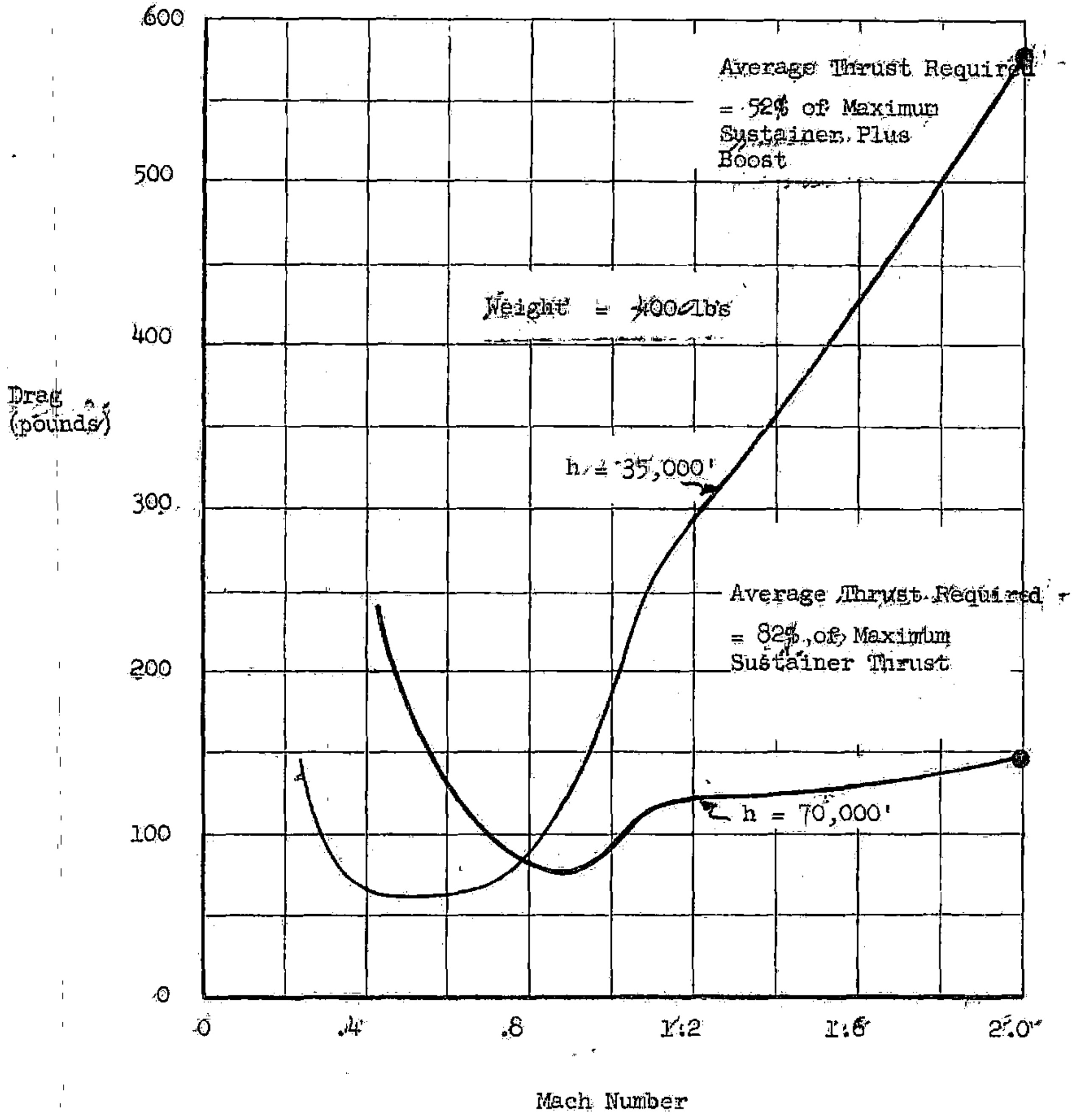


Figure 6 - Variation of Drag with Mach Number

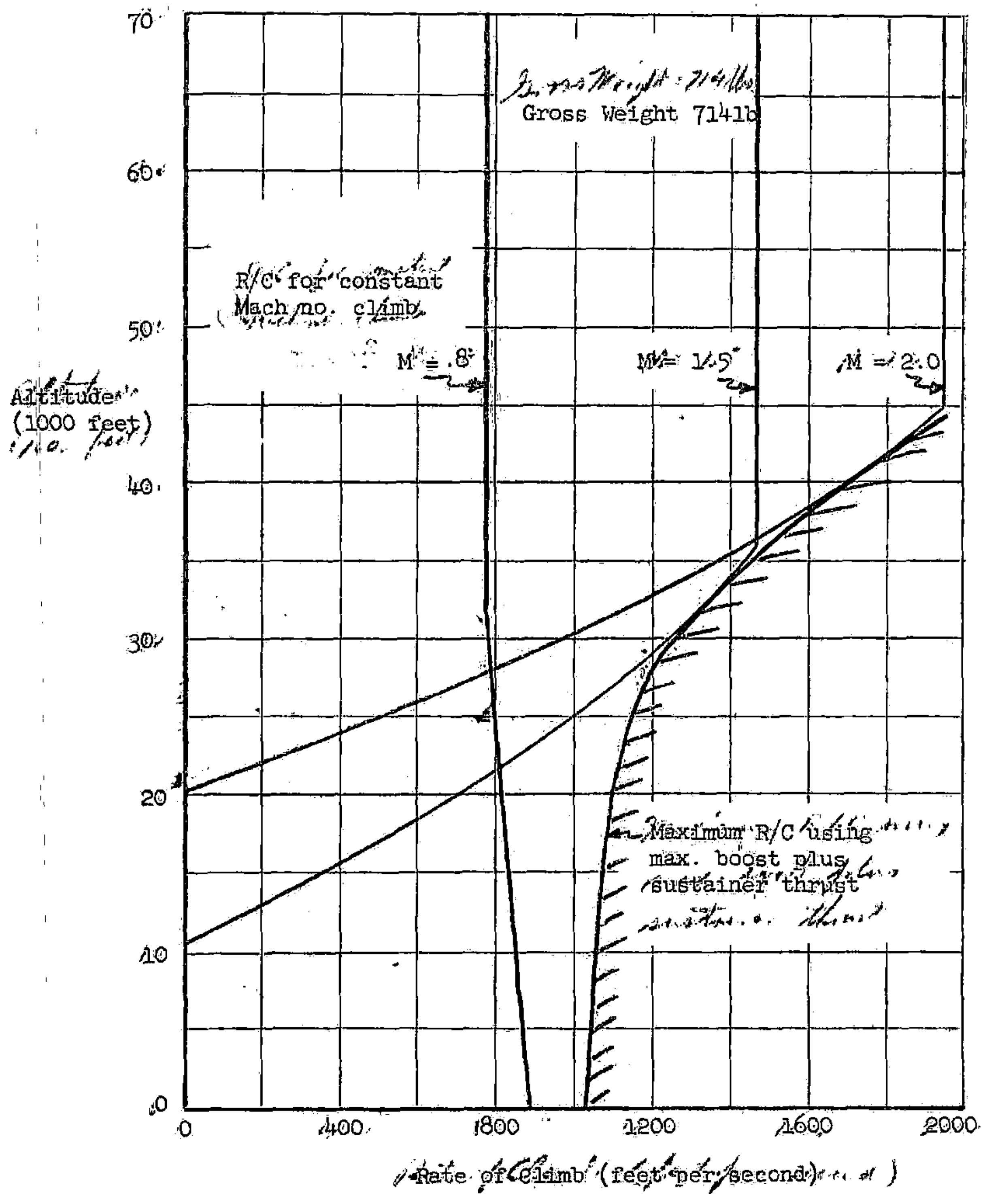
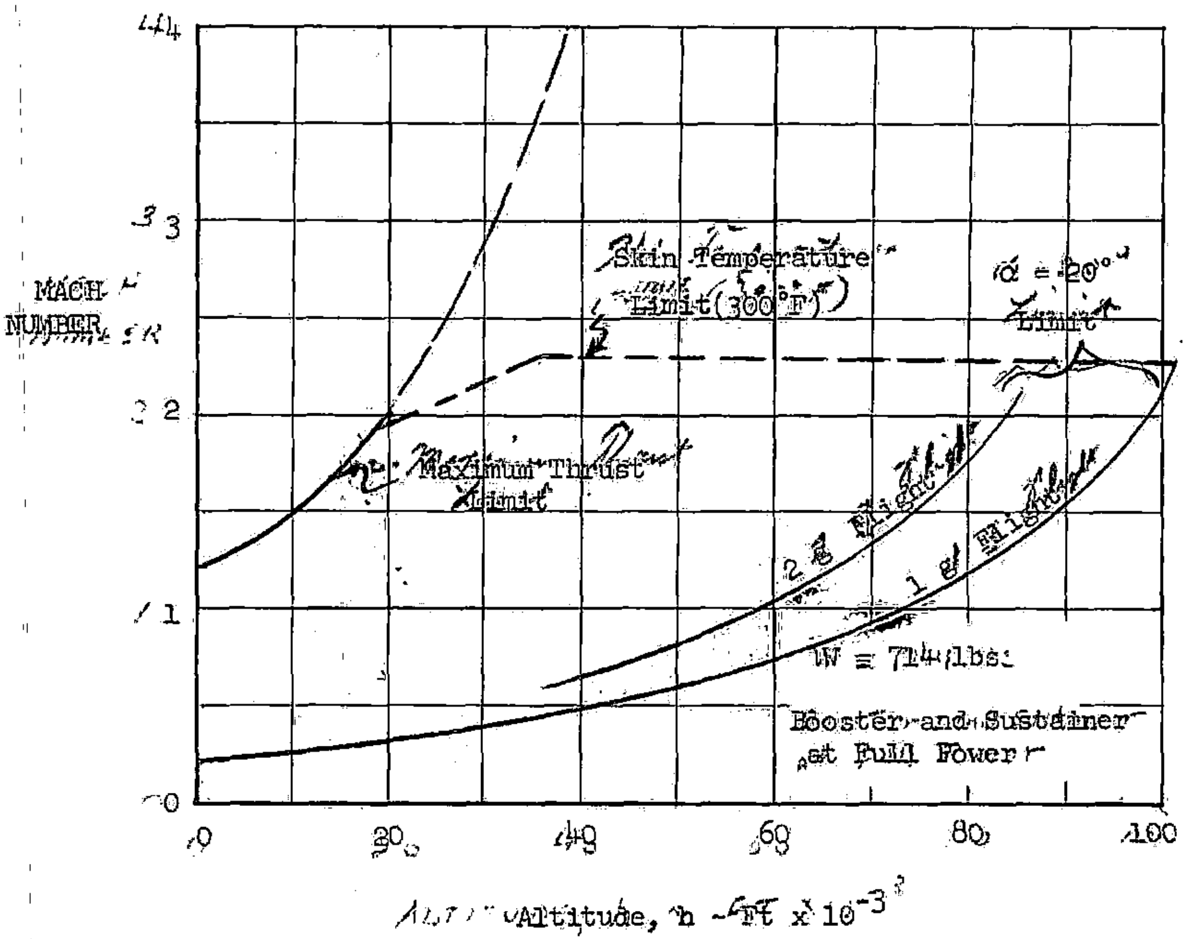


Figure 7 - Unaccelerated Rate of Climb versus Altitude



[illegible] 8 - Speed Altitude [illegible]

Figure 8 - Speed Altitude Limits

3.4 Maximum Endurance

Endurance was calculated for several cruise conditions. For the specified condition of launch at Mach number = 0.80 at 35,000 ft., climb to 70,000 ft. and cruise at Mach number = 1.5 or 2.0 the endurance is:

Climb to 70,000 ft. and Mach number = 1.5: 0 min. 58 sec.

Cruise at 70,000 ft. at Mach number = 1.5: 8 min. 9 sec.

Climb to 70,000 ft. and Mach number 2.0: 1 min. 16 sec.

Cruise at 70,000 ft. and Mach number 2.0: 5 min. 40 sec.

Endurance at maximum power ($T = 1{,}100$ lbs. @ $I_{s_p} = 245$ sec.) = 1 minute 33.5 sec.

Launch and climb were calculated by a step solution of the equations of motion, assuming launch in level flight and pullup to climb attitude when well clear of the parent airplane. Solutions were found by hand and by IBM 704 computer.

Cruise endurance was calculated as:

$$t = (L/D)_{avg} \, I_{s_p} \ln \mu \quad \text{(seconds)}$$

3.5 Launching Speed

The vehicle may be launched at any speed and altitude within the launch envelope specified in Reference 14.

At the launch condition representing highest drag (Mach number 1.35 at 15,000 ft.) the thrust to drag ratio of the vehicle is 1.165, when beyond the influence of the mother airplane. Acceleration and climb capability is thus assured.

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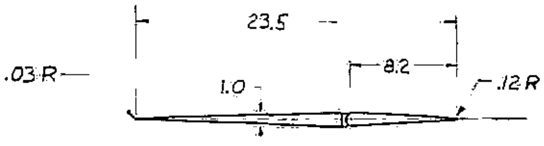
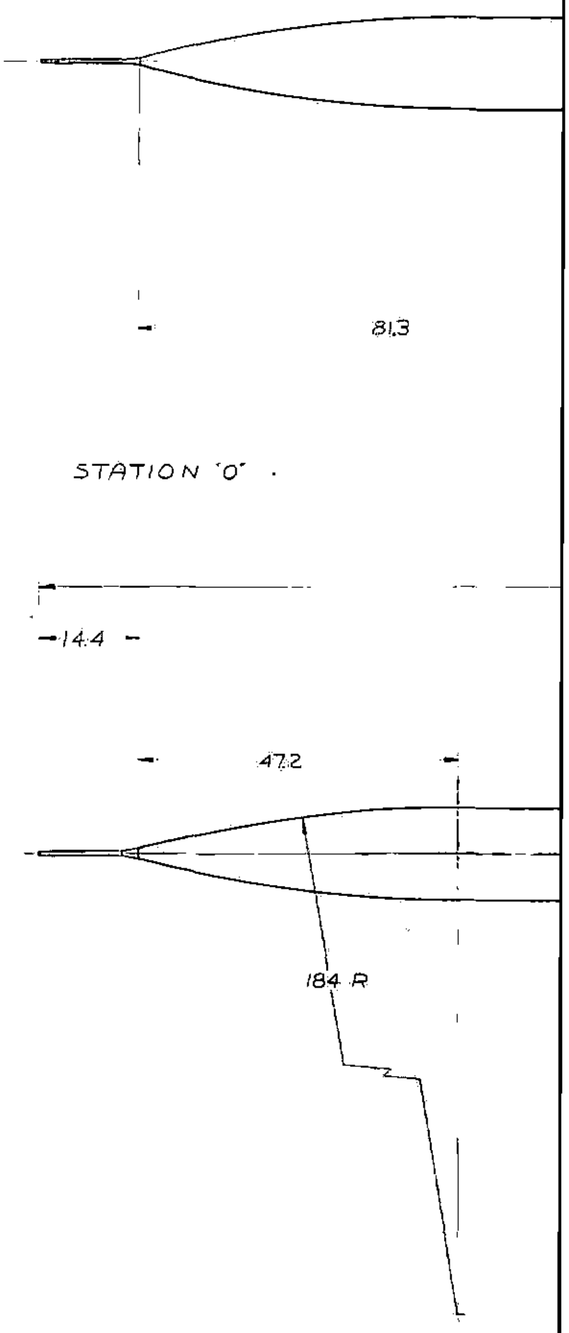
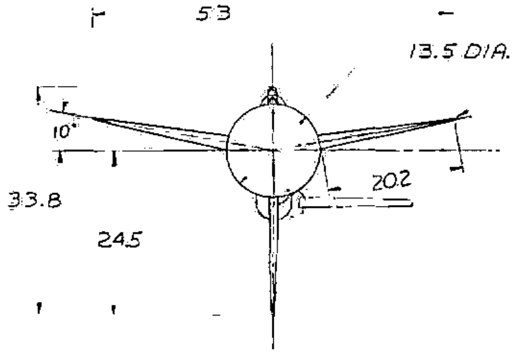
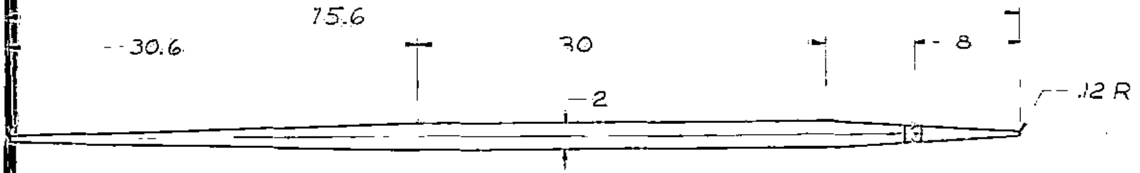


TAIL ROOT AIRFOIL
1/5 SCALE

EXPOSED WING AREA (WING + ELEVON)	11.48 FT.²
ELEVON AREA	1.72 FT.²
VERTICAL FIN AREA (FIN + RUDDER)	2.50 FT²
RUDDER AREA	.78 FT²
ELEVON ANGULAR MOVEMENT	±20°
RUDDER ANGULAR MOVEMENT	±20°
WING INCIDENCE	0°
ELEVON INCIDENCE	0°
AIRFOIL SECTION — MODIFIED DOUBLE WEDGE	







WING ROOT AIRFOIL
1/5 SCALE

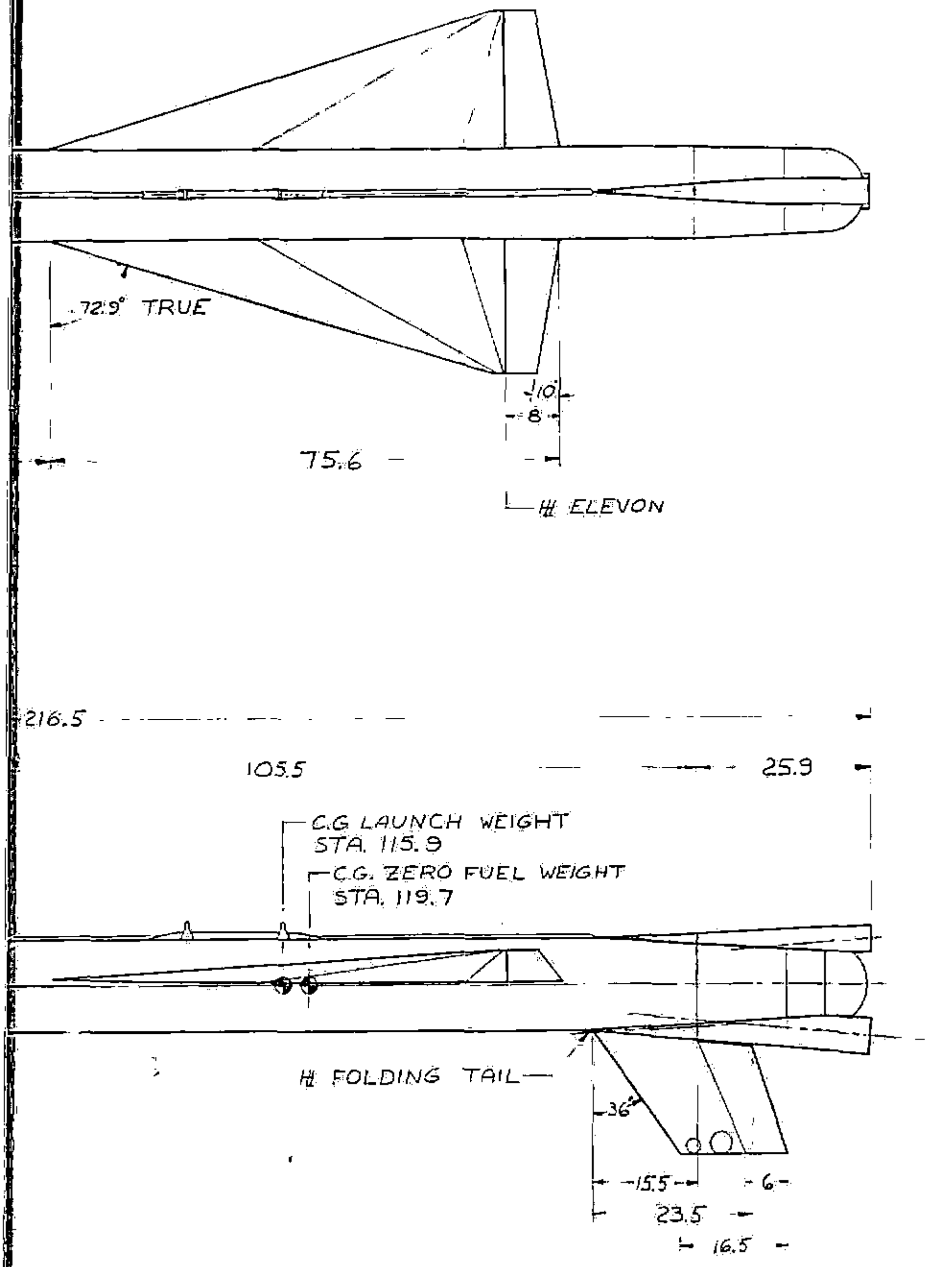


FIGURE I

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CONVAIR
MISSILE PRELIMINARY DESIGN

GENERAL ARRANGEMENT DRAWING
NAVY/AIRFORCE
SUPERSONIC TARGET

BY R. ELMS	CHECKED	APPROVED	SCALE 1/10	DATE 12/29/58

CONVAIR - SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION

SD-58-60014

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ZP-8-033

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CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION

SAN DIEGO

REPORT ZP-8-033

DATE 1 Feb. 1956

MODEL 8

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~~CONFIDENTIAL~~

TITLE

STUDY

REVISED NOSE SECTION

SIDE-BY-SIDE SEATING

TF-102 AIRPLANE

ZP-8-033

PREPARED BY R. C. Lingley
R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert

NO. OF PAGES

NO. OF DIAGRAMS

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INTRODUCTION

This study has been prepared to determine the necessary rearrangement of the TF-102A side-by-side trainer in order to obtain a minimum cross-sectional area.

Pulling in of the external lines has been accomplished by the elimination of the center console and by positioning of the seats close together. The center console had been used to exactly simulate the tactical F-102 right hand console. Console panels which had been located in this area, are now located on the instrument panel.

In order to improve the lines in the canopy area, a flat center panel has been added to the windshield.

A long nose, similar to the F-102A, has been added to further improve the drag.

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SUMMARY

The reduction in cross-sectional area, improved windshield and canopy lines and extended nose, should contribute to a reduction in drag which should result in an improvement in ceiling and range.

The gross weight change, compared to the existing TF-102A, is negligible; however, the average center of gravity moves forward approximately one-half percent, due to the addition of the long nose and subsequent forward shift of the nose radar equipment.

PERFORMANCE SUMMARY

		Existing TF-102A	TF-102A With Revised Nose Section
High Speed	Mach No.	.98	1.03
Combat Ceiling	Feet	49,300	49,900
Combat Radius	N. Mi.	316	321

Discussion:

The following outline shows the major changes from the existing TF-102A airplane required to accomplish this change:

General:(Ref. Dwg. No. SD-56-08002)

1. Revise entire contour forward of Sta. 310.25 while maintaining the forward missile bay and wing intersection. (See lines drawing SD-56-08004).
2. Revise engine air inlet duct from lip to Sta. 217 only.
3. Revise boundary layer plows and bleed duct to refrigeration unit.
4. Incorporate long nose (use F-102A radome).
5. Change "V" type windshield to a 12" wide, flat center panel, flat diagonal panels and curved corner panels.
6. Revise canopy contour to provide minimum "HIAD" clearance at pilots' heads.
7. Remove center console.
8. Move seats inboard 2.88" and down 2".

The following breakdown shows detail changes required:

A. Fuselage (Ref. Dwg. No. SD-56-08003)

   1. Redesign entire structure forward of Sta. 310.25 except missile bay doors, beams and missile launchers.
   2. Design new windshield and canopy.
   3. Use F-102A radome.
   4. Provide nose access doors similar to F-102A.

5. Design new ATM door and frame.

6. Design new nose gear door.

B. Cockpit (Ref. Dwg. No. SD-56-08005)

1. Remove center console.

2. Relocate console panels to instrument panel and outboard consoles.

3. Design new instrument panel.

4. Move seats inboard 2.88" and down 2".

5. Move outboard consoles inboard 2.88".

6. Design new glare shields and glare curtains.

7. Design new blind flying enclosures.

8. Cut off top of seat rails.

9. Revise installation of optical sight.

C. Controls System (Ref. Dwg. SD-56-08006)

1. Reduce spread between rudder pedals by 2" and move inboard 2.88"

2. Relocate control stick pivot points and move inboard 2.88".

3. Move throttles inboard 2.88".

4. Revise entire control system routing forward of Sta. 310.25.

D. Landing Gear

1. Use existing TF-102A nose gear and mechanism.

E. Miscellaneous

Since most of the contour is changed forward of Sta. 310.25, it will be necessary to revise all systems consistent with the revised structure; although most items are located in the same general area as in the existing TF-102A airplane.

See following pages for weight and balance information and performance summary.

ANALYSIS
PREPARED BY
CHECKED BY
REVISED BY

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

PAGE
REPORT NO. ZP-8-033
MODEL 8
DATE 1 Feb. 1956

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WEIGHT SUMMARY - TF-102A SIDE-BY-SIDE TRAINER

REVISED NOSE SECTION

Wing		3,044	
Tail		315	
Fuselage		3,902	
Landing Gear		1,047	
Surface Controls		466	
Nacelle Group		38	
Propulsion Group		6,317	
Engine	4,980		
Air Induction	612		
Exhaust System	129		
Cooling System	18		
Lubricating System	28		
Fuel System	427		
Engine Controls	36		
Starting System	87		
Instruments and Navigation		164	
Hydraulic and Pneumatic		291	
Electrical		563	
Electronics		2,316	
Armament		592	
Furnishings and Equipment		380	
Air Conditioning and Anti-Icing		242	
Auxiliary Gear		61	
Unaccountable Weight		- 45	
Weight Empty			19,693
Pilots (2)		460	
Fuel (JP-4)		6,979	
Unusable	154		
Wing	6,825		
Oil		46	
Trapped	21		
Engine	25		
Armament		1,366	
Missiles (6)	762		
Rockets (36-2.00")	382		
Launchers	222		
Equipment		36	
Useful Load			8,887
Gross Weight			28,580

Note: Weights are based on Actual Weight and Balance Report ZW-8-021 for the TF-102A Airplane. (28,484 lbs.)

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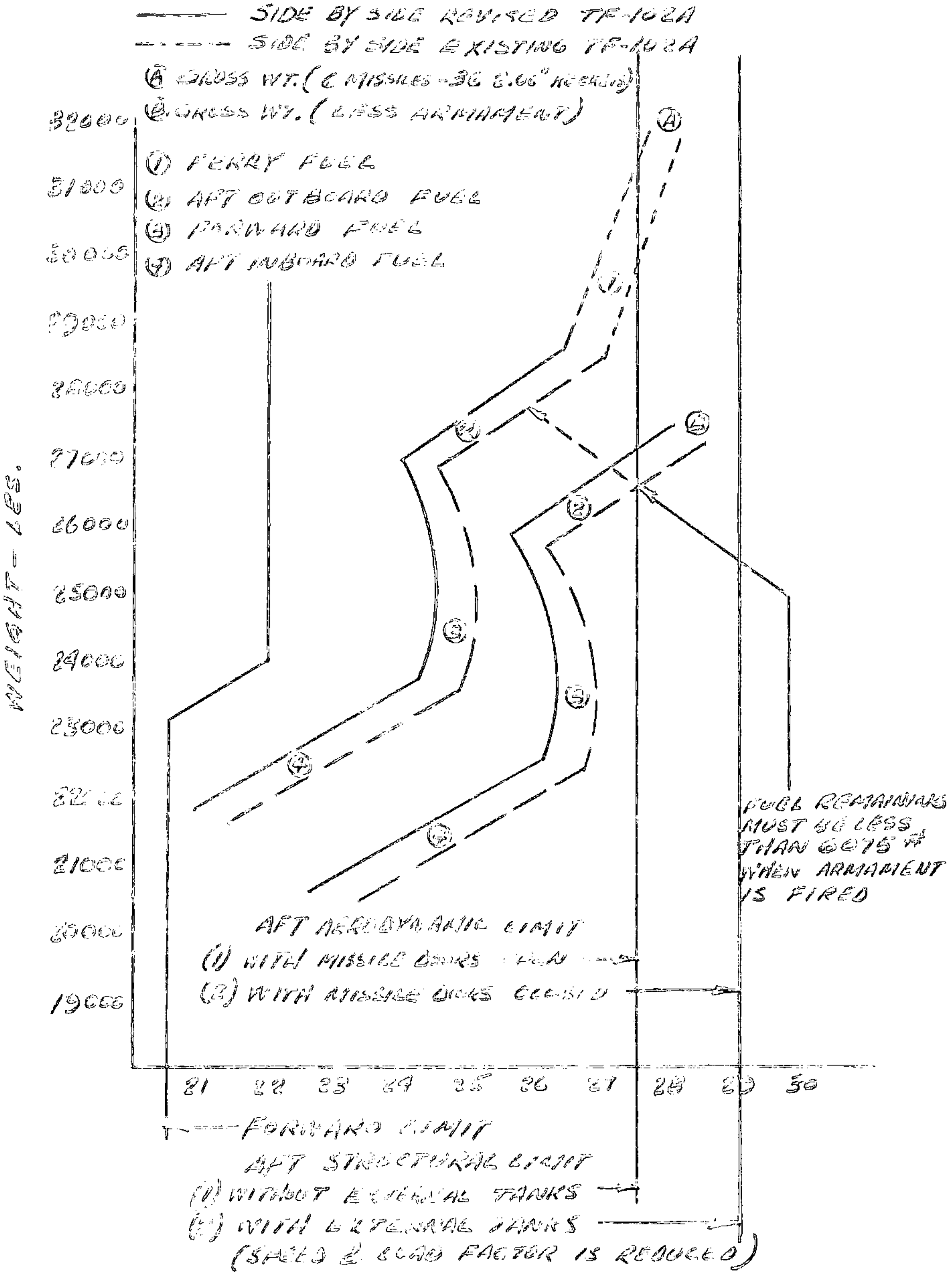
REVISED TF-102A
CENTER OF GRAVITY - % M.A.C.
(JP-4 FUEL)



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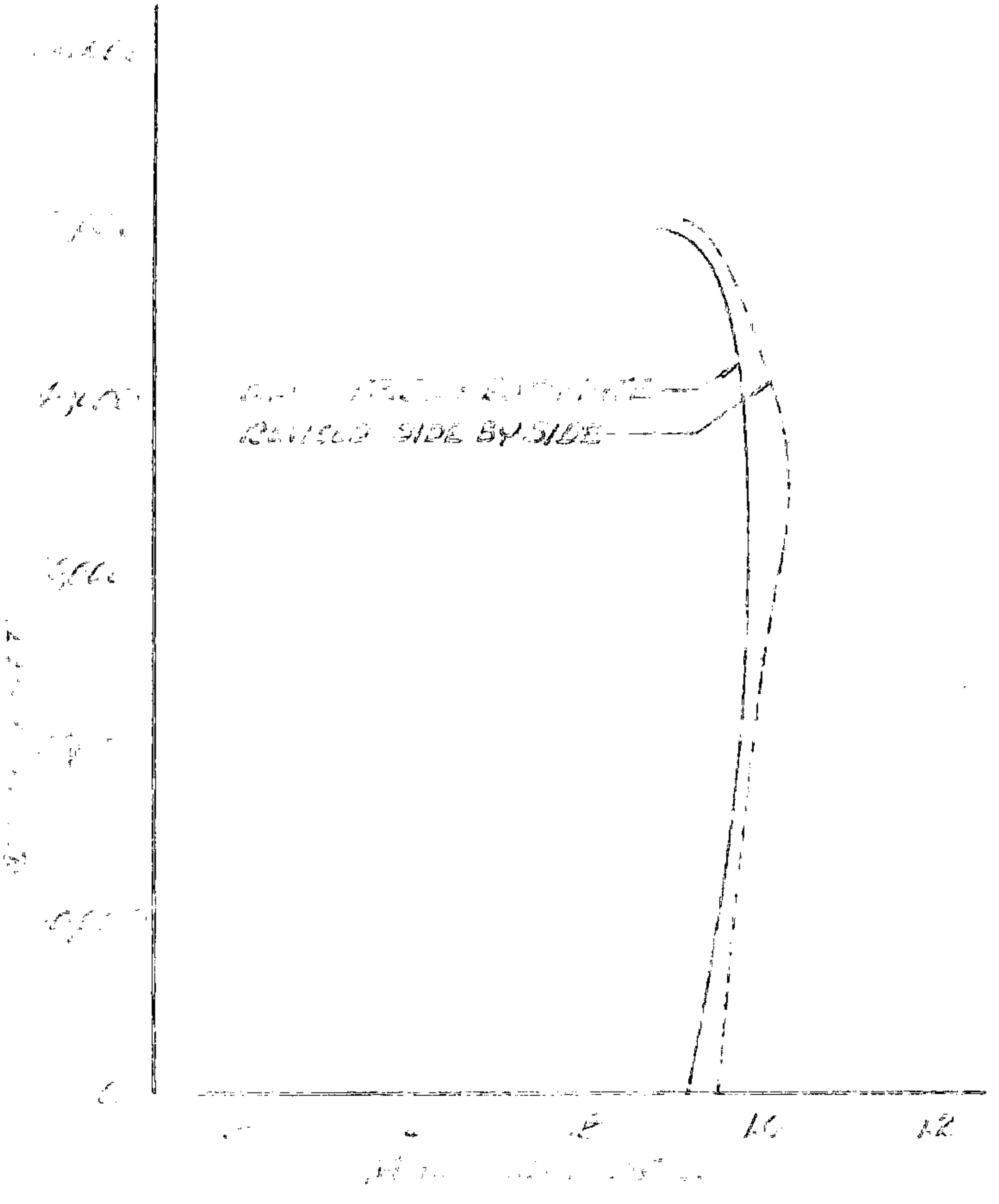
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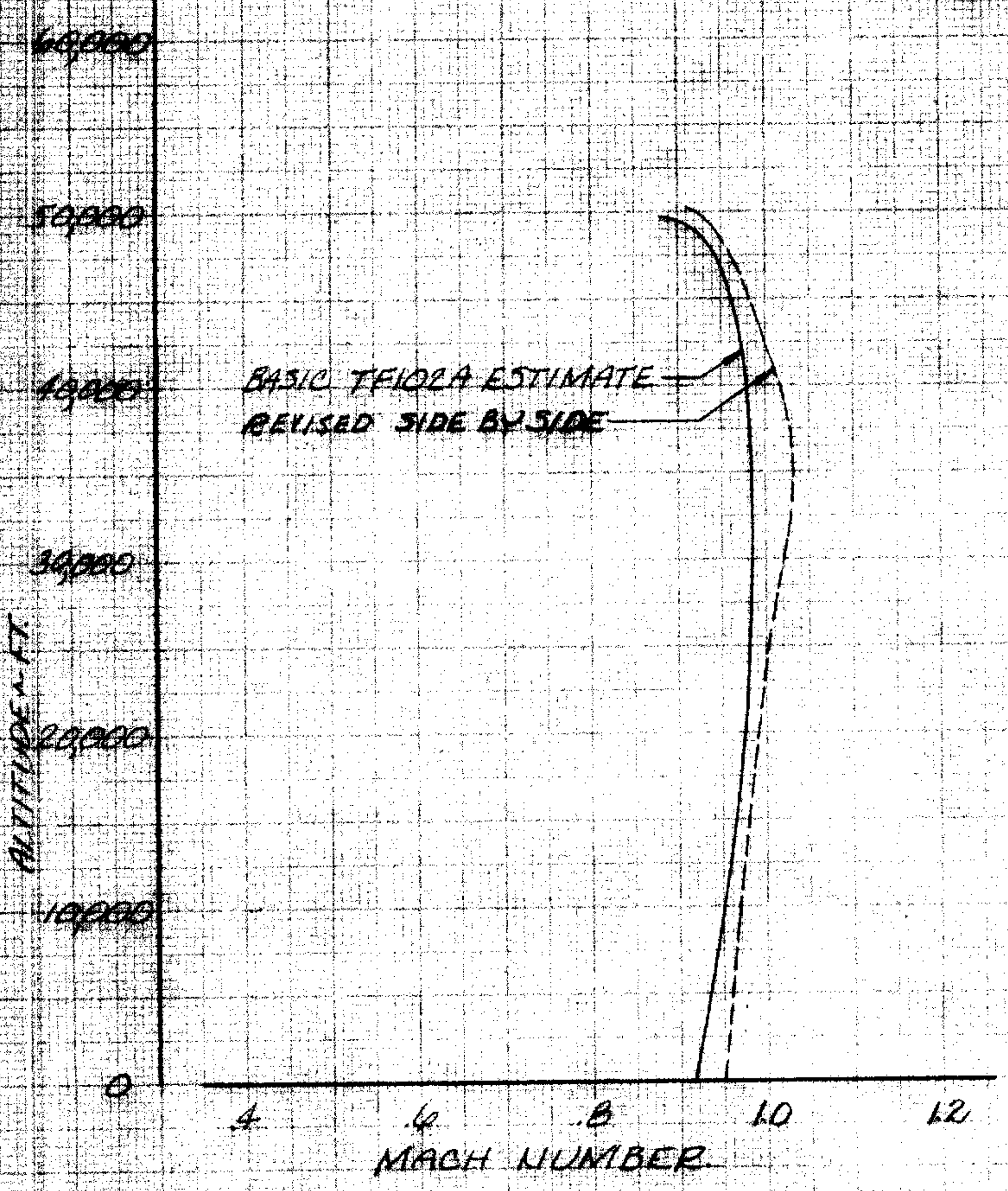
SIDE BY SIDE

K+E 10 X 10 TO THE 1/2 INCH 359T-11G
KEUFFEL & ESSER CO.

TF-102A—REVISED NOSE SECTION

HIGH SPEED & ALTITUDE

J57-P-23 MILITARY + AB POWER



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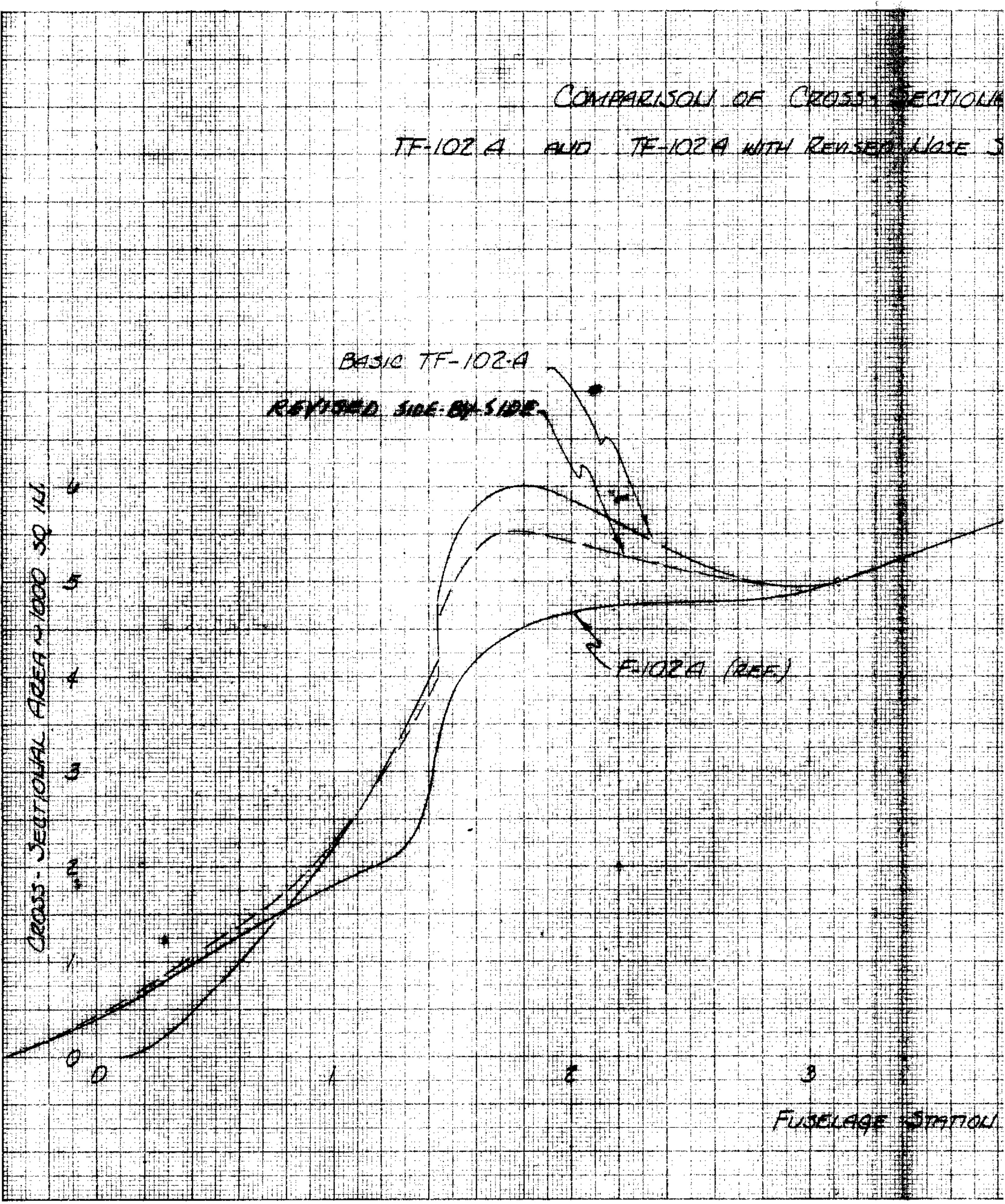

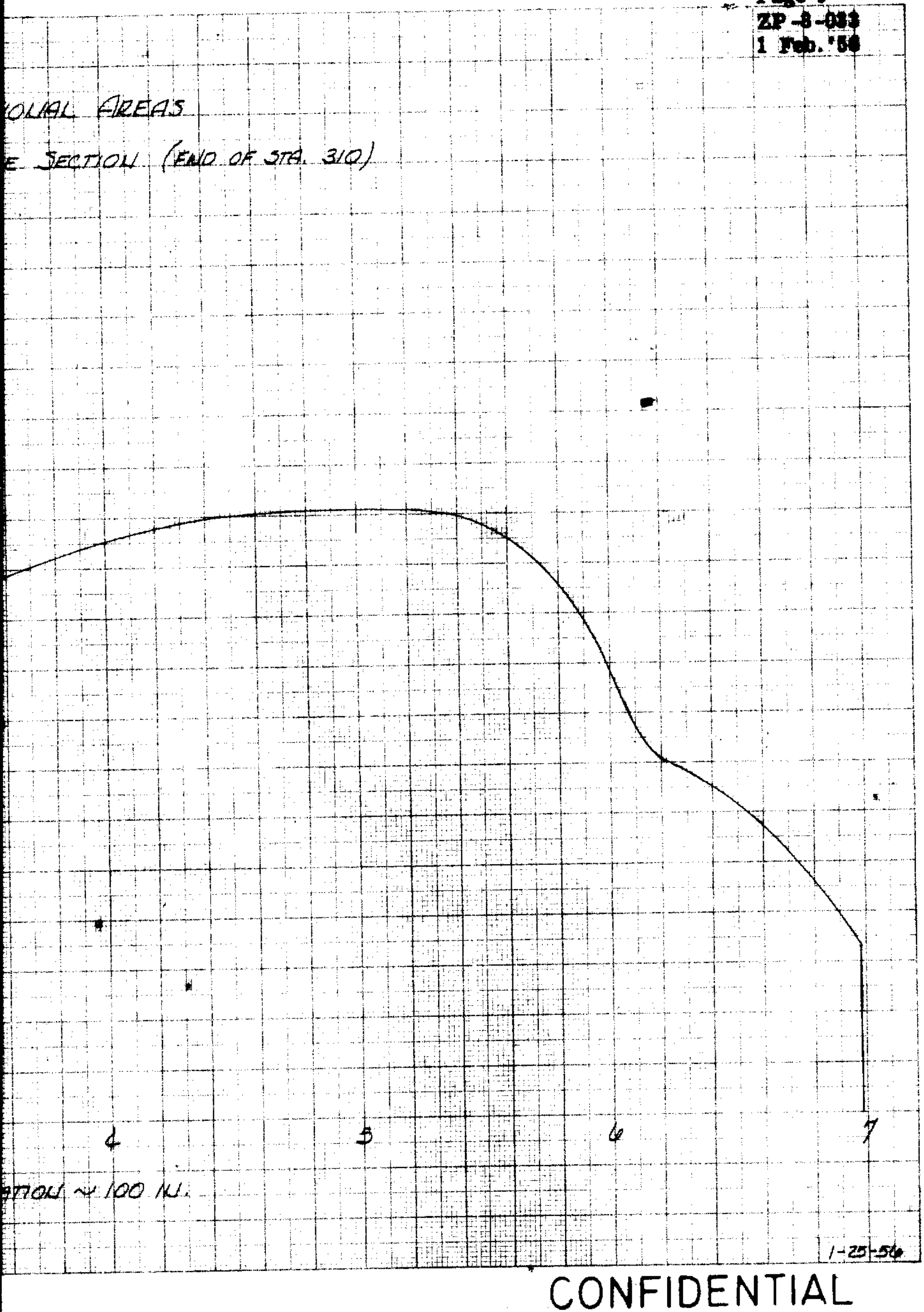

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1-25-56

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ZP-8-052

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REPORT ZP-8-052
DATE 18 June 1957
MODEL F-102A

Classification Changed to:
UNCLASSIFIED
Author [illegible] 12-12-66
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Reclassif[illegible] Dept. Date
[illegible] 13[illegible] 4-4-67

TITLE

Alternate Armament for the F-102A Airplane

1. (2) External MB-1, (5) Falcons Stored Internally.

2. (1) MB-1 + (4) Falcons Stored Internally.

ZP-8-052

PREPARED BY F.L. Tripp

GROUP A. R. Lambert
Preliminary Design

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CHECKED BY ______

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INTRODUCTION

This study represents a comparison of two additional armament installations on the F-102A. The armament selections are: two MB-1 rockets externally mounted with five internal Falcon missiles retained; and four Falcon missiles and one MB-1 rocket carried internally. (Hereafter in this report these configurations will be referred to as 2+5 and 1+4 respectively.) The basic armament consists of six Falcon missiles and (24) 2.75" rockets. No 2.75" rockets or provisions for rockets are considered in either of the proposed configurations.

Factors considered in the study are changes in aircraft performance, weight and ballast requirements, fire control system revisions, structural modification and operational aspects.

Convair has installed an MB-1 ejection system in the center bay of an F-102A airplane. Development tests have been conducted on the system.

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SUMMARY

Alternate armament configurations are considered for the F-102A consisting of 2 MB-1 rockets mounted externally with five internal Falcon missiles retained (no 2.75 rockets) and 1 MB-1 rocket with 4 Falcon missiles carried internally (no rockets).

Results are as follows:

Incorporation of the external storage configuration in relation to the internal configuration will necessitate a longer flight test program. Ballistic data must be obtained from a development flight test program and flight miss distance must be determined from the evaluation program. Since a quantity of ballistic information is available on the (1+4) configuration, it may be possible to include this data into the (1+4) computer within the schedule shown herein.

Weapon system effectiveness data, also included herein, gives a comparison of the effectiveness due to the decrease in Vmax with external stowage. For the reduction in interceptor Vmax from about M 1.2 to M 1.0, with the external stowage the probability of detection and conversion for rear hemisphere attacks against a M 0.9 threat would be essentially the same. However, there is a loss of effectiveness in the area intercept radius due to the reduced closing rate requiring a longer combat time. No tail attack effectiveness is realized when the M 1.0 interceptor is used against the M 1.0 threat. The snap-up capability of the M 1.0 system is degraded. For a M 0.9 target at 60,000 feet, beam and nose attacks would be marginal and tail attacks would probably be unsuccessful.

With external configuration it is necessary to carry 550 lbs. of ballast to maintain the C.G. within available limits to permit any armament selection. It is also necessary to limit the load factor to six in confining structural rework to local modifications only. Spar and rib rework is necessary to maintain the present design load factor of 7g.

There is no loss in performance or decrease in load factor over the basic airplane with the 1+4 system. System effectiveness is increased over the 2+5 because of greater radius capability against a higher speed threat, and greater snap-up capability. Both systems (1+4 and 2+5) effectiveness are increased over the basic airplane under a wide variety of attack conditions due to the mixed load. Flight test time for the 1+4 system is limited to the

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evaluation program only as the development program is already partially completed. A limited number of test rounds must be fired to complete the required results for the development program.

Take-off weights for the two configurations and basic airplane are as follows:

	Basic A/C	1 + 4	2 + 5
Armament	1194	1323	2265
Ballast	0	0	550
Take-off Weight	27,943	27,920	29,566

Performance of the two configurations and basic airplane is as follows:

	Basic A/C	1 + 4	2 + 5
Max Speed	1.18	1.18	1.0
Area Intercept Radius (Equal Combat Time)	375 N. Mi.	375	310

An inherent advantage of the (1+4) configuration is the extended range mission capability.

Based on these results, the (1+4) configuration is recommended.

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I Performance

Since no external changes are made and only a small difference in weight (23#) occurs between the (1 + 4) and the basic airplane, performance values are not effected. A comparison of performance with the (2 + 5) is as follows:

Performance Factor	Basic A/C & (1+4)	(2+5)
Max Speed	1.18	1.0
Combat Ceiling	51,700	50,000
Time to Climb to 40,000'	4.3 Min.	4.9
Area Intercept Radius	375 N. Mi.	310
Take-off Distance	2,900 Ft.	3,350

The amount of aileron and rudder control deflections required for trim when one externally stored missile is expended is, at 150 knots, ± 1.6° aileron, ± 1° rudder and at $M = 0.9$, ± 0.5° aileron and ± 0.3° rudder. These values are acceptable.

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II Aircraft and System Modification

A. 2+5 Configuration (Fig. 1, Dwg. No. 8-05078)

a. Structural Requirements

The external configuration of the MB-1's must be limited to a load factor of approximately six in the absence of detailed analysis to assure that the modifications are confined to local areas.

This reduction in maximum load factor is due to an increase in wing load outboard of a store location. The increased inertia relief is, of course, in-effective in this area. A detailed load and stress analysis would be required to accurately determine the maximum load factors. However, areas effected would be spars and ribs outboard of the attachments and the spars and fittings at the wing root. Rework in these areas would be costly due to difficulty in obtaining access.

b. General

1. Replace center forward Falcon with electronics compartments.
2. Remove provisions for (24) 2.75" rockets.
3. Add shorter center bay doors.
4. Fair over forward center bay and provide access into this compartment.
5. Add (2) MB-1 auxiliarys and launching provisions.
6. Revise pneumatic, electronic and electrical systems to accommodate the above changes.

c. Armament Group

1. Remove center-forward Falcon missile displacing gear.
2. Remove rocket provisions. (Outboard doors may be retained by removing rocket tubes and blast pans.)
3. Add pylons and launching mechanism to wing external tank location.
4. Add shorter center bay doors to accommodate one aft Falcon.
5. Modify missile bay beams to accommodate relocation of center door actuators.

d. Wing Group

1. Redesign fuel tank pylon attachment fittings.

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(Fittings can be removed and installed through a wing lower surface access door attached with screws.)

2. Revise structure adjacent to fittings to provide for higher loads. (minor modifications can be realized through the access door.)

e. Fuselage Group

1. Add structure at forward-center missile bay. Add partition between missile bay beams between forward and aft bay. Provide access door into this compartment and provide for marker beacon antenna in this area.

f. Mechanism Group

1. Revise center bay door actuator installation to provide for shorter center bay doors.

g. Pneumatic Group

1. Remove forward-center missile displacement gear actuator and revise pneumatic system accordingly.

2. Revise pneumatic system to provide for revision of door actuators on shorter center bay door.

h. Electronics Group

1. Remove provisions for forward center missile and (24) 2.75" rockets.
2. Modify electronics and add provisions for (2) MB-1.
3. Add rocket umbilical plugs for (2) MB-1.
4. Modify racks as required and add racks in new compartment.
5. Relocate marker beacon antenna from center bay door to forward bay.

i. Electrical Group

1. Delete rocket wiring and provisions.
2. Delete harnesses and provisions for forward center missile.
3. Provide harnesses and provisions for two externally mounted MB-1 rockets.

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4. Revise cockpit armament control panel.
5. Provide for MB-1 heating blanket.

j. Heating and ventilating Group

1. Supply cooling air to electronic compartment in forward center bay area.

B. 1+4 Configuration (Fig.2, Dwg. No. 8-08036; Ref. ZP-8-040).

a. General

1. Replace (2) center Falcons with (1) MB-1 rocket.
2. Remove provisions for (24) 2.75" rockets.
3. Add shorter center bay doors.
4. Fair over forward end of center bay and provide access into this compartment.
5. Revise structure to provide for ejection loads.
6. Revise pneumatic, electronic and electrical systems to accommodate the above changes.

b. Armament Group

1. Remove (2) Falcon missile displacing gear from center missile bay.
2. Remove provisions for rockets, (Existing outboard doors may be retained by removing rocket tubes and blast pans).
3. Add ejection mechanisms and provisions for (1) MB-1 rocket in the center missile bay.
4. Add shorter center bay doors to accommodate the MB-1 rocket.
5. Modify the missile bay beams to accommodate relocation of center door actuators and removal of trunnion supports for the center missiles.
6. Design and add ejection beam.

c. Fuselage Group

1. Revise structure to provide for ejection loads of the MB-1 rocket. The following frames are effected: Stations 273, 310.25, 349.75 and 405.
2. Add structure at forward end of center missile bay between stations 217 and 273. Add partition between missile bay beams at station 273. Provide access door into this compartment and provide for marker beacon antenna in this area.

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d. Mechanism Group

1. Revise center bay door actuator installation to provide for shorter center bay doors.

e. Pneumatic Group

1. Remove two center missile displacement gear actuators and revise pneumatic system accordingly.

2. Revise pneumatic system to provide for revision of door actuators on shorter center bay door.

3. Relocate (4) air flasks to center bay compartment.

NOTE: Assume all six doors will operate simultaneously.

f. Electronics Group

1. Remove provisions for two center missiles and (24) 2.75" rockets.

2. Add provisions for (1) MB-1 rocket.

3. Add rocket umbilical plug for the MB-1.

4. Modify electronics for removal of (2) Falcons and addition of (1) MB-1.

5. Relocate marker beacon antenna from center bay door to forward end of bay.

g. Electrical Group

1. Delete rocket wiring and provisions.

2. Delete harnesses and provisions for two center Falcon missiles.

3. Provide harnesses and provisions for one MB-1 rocket.

4. Revise armament control panel in cockpit.

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h. Controls Group

1. Revise elevon control cable and relocate turnbuckles in center missile bay.

i. Heating and Ventilating Group

1. Supply cooling air to electronic compartment in forward center bay area.

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III Weight and Balance

Weight changes for each configuration are shown in Table I. The 1+4 configuration results in an airplane that is 23# lighter than the current production airplane. In order to maintain the c.g. within the limits, for take-off and utilizing any armament configuration (2 MB-1 only, 5 Falcons only or 2+5), the 2+5 airplane requires 550# of ballast. Overall weight increase of the 2+5 over the 1+4 configuration is 1646# of which 942# is armament.

C.G. envelopes are shown for each configuration (Figs. 3&4). The (2+5) configuration with full armament and fuel expended exceeds the forward C.G. limit at landing. Additional caution should be exerted with this configuration until further analysis removes any restriction at landing.

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IV Operational Aspects

A. (2+5) vs (1+4)

1. On the basis of a single pass attack, considering the MB-1 only in each case, the following facts are of interest: In a co-altitude attack mode the kill capabilities of either configuration are about the same against a Bison (B-52 Type) target traveling at M = 0.9. A bomber traveling at M = 0.9 at 40,000 feet elevation can be successfully attacked from any point of the compass by either airplane. However, if the bomber speed should be increased to 1.0, the 1.2 interceptor would have "around the clock" capabilities, but a 1.0 interceptor would have reduced capability in a tail attack. See Figure 5.

When 2 MB-1's are fired, the escape maneuver must be modified to limit the pilot radiation dosage. Considering an attack against a M 0.9 threat at 35,000 ft, and 2g escape maneuver and using half the dosage for the (2+5) configuration, the kill probability for (1) MB-1 fired from either configuration is the same. The decreased speed of the (2+5) configuration permits a smaller radius turn therby providing escape at a greater distance from the burst although launching occurs at approximately the same separation.

2. The decreased maximum velocity resulting from external mounting will decrease the closing rate for tail attacks and increase the combat time and fuel used at afterburner power. Providing for this increased combat fuel allowance will decrease the overall mission radius approximately 12 miles for each additional minute of combat. For the area intercept mission, combat time is (5) minutes. Each minute of tail chase of this combat time for the (1+4) configuration requires (3) minutes for the (2+5) configuration to produce the same closure. An interceptor must be vectored to within eight miles of the target at the start of the combat tail attack to make one pass only and utilize the full area intercept radius of the (2+5) configuration (310 N. Mi.)

3. In a snap-up attack, the interceptor's effectiveness is impaired by external weapon stowage due to the reduced initial velocity of the interceptor. Tail attacks will be particularly degraded since closing velocity is at a minimum for these conditions.

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B. Mixed Load Advantages

1. Counter measure denying range information degrade the effectiveness of the MB-1 rocket. Guided missiles such as the Gar 1+2 need only be fired within their aerodynamic range.

2. Under conditions where an attack using the optical sight is necessary, the effectiveness of MB-1 can be degraded due to ranging errors. Under such conditions the employment of a passive IR homing missile (Gar 2) could provide enhanced combat effectiveness.

3. One of the most complex employment problems arises in connection with multiple interceptors attacking multiple targets with nuclear warhead missiles. The capability of using small high explosive missile in this situation may be of significant advantage.

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V Schedule

A. (1+4) Configuration

1. Ejection Launching. Ejection launching of MB-1's during the development flight tests have been sufficient to verify analytical predictions. No problem areas are expected to be encountered.

2. Ejection Mechanism. Qualification testing has been proceeding. Production equipment will require packaging and some additional qualification tests.

3. Flight Test. Development flight tests are nearing completion. Evaluation flight tests are required.

B. (2+5) Configuration

1. Launching Mechansim. Launching rails, which must be procured, can be mounted on modified external fuel tank pylons.

2. Flight Test. Development and evaluation flight tests are required for the external stowage configuration. Some schedule overlap is possible.

C. Program Schedule

Schedules for both configurations are shown by Figure 6.

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APPENDIX A

The installation study (Convair Report ZP-8-052) of adding MB-1 capability to the F-102A has been extended to provide a minimum aircraft modification program. This retains the 2.75" rocket capability. In each case, the 2 + 5 configuration (2 external MB-1's, 5 internal Falcons) or the 1 + 4 configuration (one MB-1 and 4 Falcons stowed internally), the existing center missile bay doors are retained. The additional electronics space required is provided above the center missile bay doors in the area from Stations 217 to 273.

(2 + 5) Configuration

Retention of the rocket capabilities (24 - 2.75" rockets) and retaining the capability of taking off with MB-1 armament only requires ballast of 395#. Take off weight with full armament is 30,004.

Performance changes due to the addition of the 2.75" rocket capability are: Combat Ceiling from 50,000' to 49,700' and time to climb to 40,000' from 4.9 to 5.0 min.

(1 + 4) Configuration

Retention of the rocket capabilities (18[*] - 2.75" rockets) gives a take-off gross weight of 28,364#. The resultant CG curve will lie within acceptable limits.

Performance changes are: Combat ceiling from 51,700' to 51,400' and time to climb to 40,000' from 4.3 to 4.4 min.

The missile bay doors must be reworked to provide clearances in the aft end for the MB-1 rocket.

---

[*] In this modification 6 rockets are deleted due to interference with the MB-1 installation.

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TABLE #I [redacted]

Configuration Weight and Balance

Modification	(2) Ext. MB-1 (5) Int. Falcons				(1) Int. MB-1 (4) Int. Falcons			
	Wt.	Arm	C.G.	Mom.	Wt.	Arm	C.G.	Mom.
Basic Aircraft	27,943			11,731,736	27,943			11,731,736
Remove Falcons	-762			-244,221	-254			-81,407
Remove Falcon Provisions	-110	264		29,040	-225			-69,905
Remove Rockets	-432			-114,048	-432			114,048
Remove Rocket Provisions	-106			-28,876	-106			-28,876
[redacted] Add MB-1	1630	440		717,220	815	342		278,730
Add MB-1 Provisions	273			117,190	110			36,709
Fuselage Str. Increase					+125			44,491
Missile Bay Door	-60			-17,322	-41			-11,838
Misc.	+30/-30			-904	-15			-4,350
Total	28,376			12,131,715	27,920			11,780,742
Ballast	400	217		86,800				
	155	274		42,470				
TOTAL 2MB-1/(1+4)	28,931	423.8	30.8%	12,260,985	27,920	421.9	30.2%	11,780,742
2 MB-1	28,931	423.8	30.8%	12,260,985				
2 MB-1+5 Falcons	29,566	421.8	30.2%	12,470,408				
5 Falcons Only	27,936	420.7	29.9%	11,753,206				
Armament Out	27,301	422.8	30.5%	11,543,785	26,597	426.4	31.8%	11,339,198

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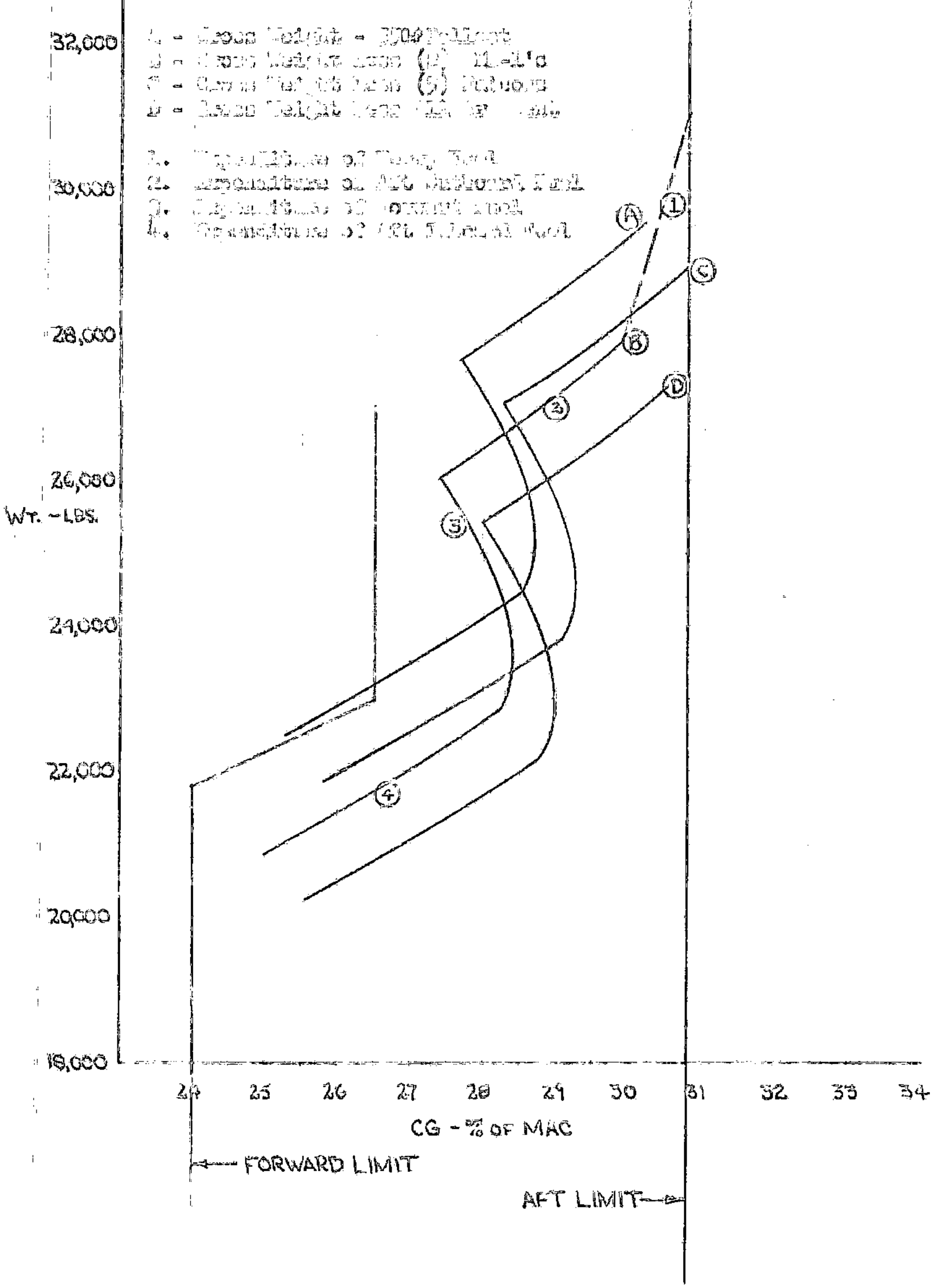
Fig. 5

[illegible] = (2) [illegible] Falcons [illegible] Loading [illegible] Chart ([illegible] Fuel).



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[illegible]

[illegible]

[illegible]

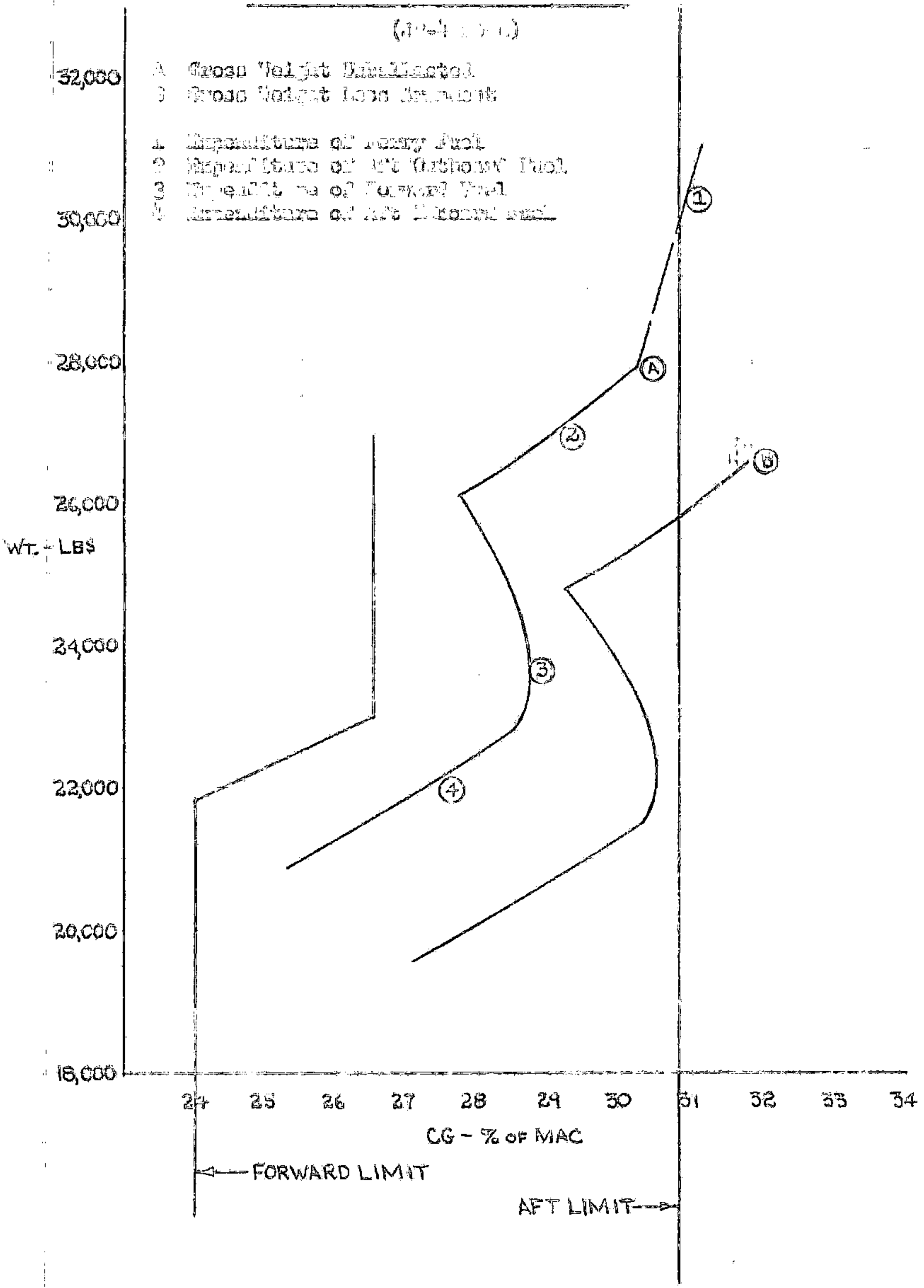

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Fig. 5

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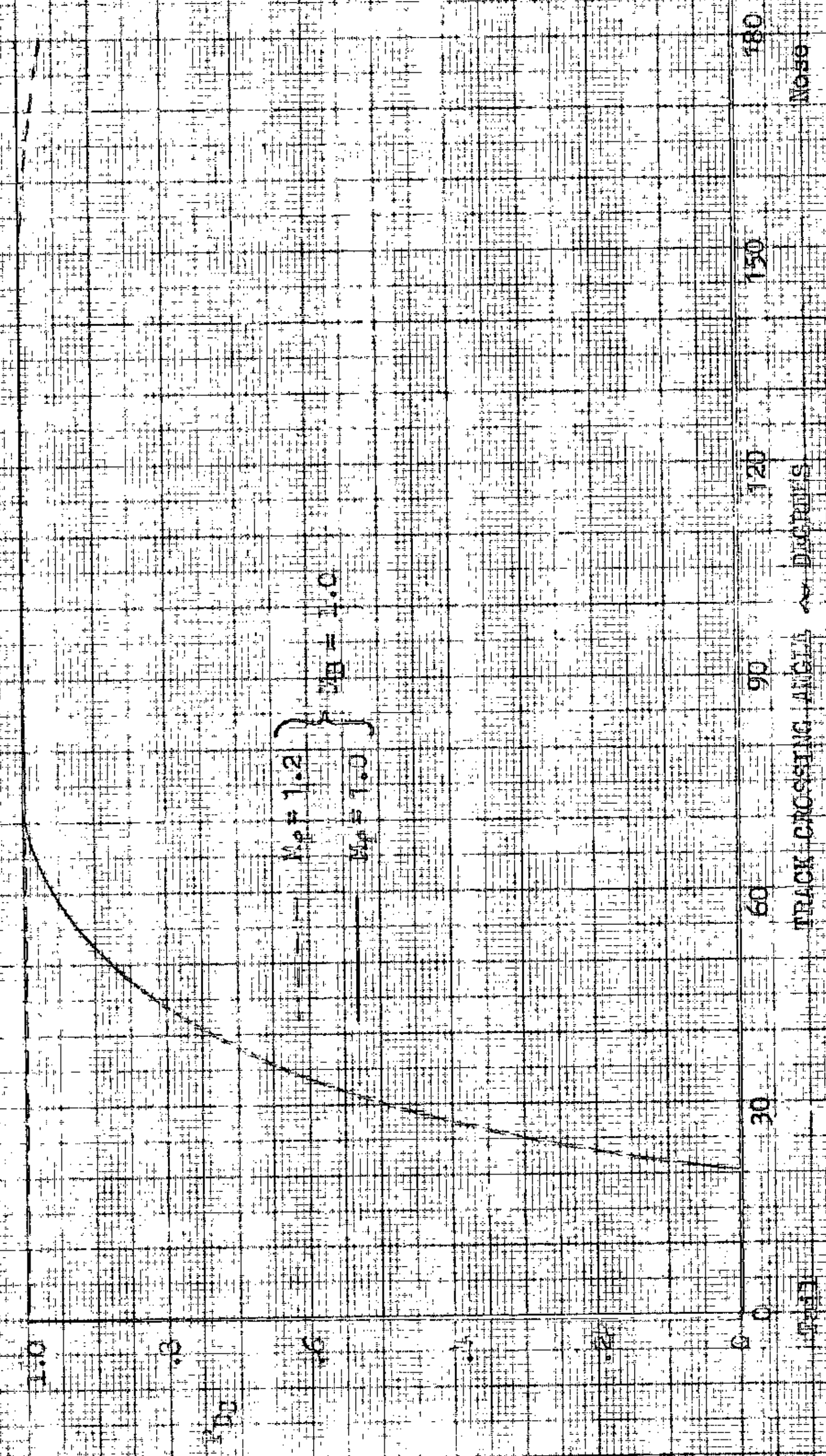
PROBABILITY OF DETECTION AND CONVERSION
D VS TRACK CROSSING ANGLE



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K&E 10 X 10 TO THE CM. 359-14

K&E 5 YEARS BY MONTHS X 100 DIVISIONS KEUFFEL & ESSER CO. 359-192 MADE IN U.S.A.

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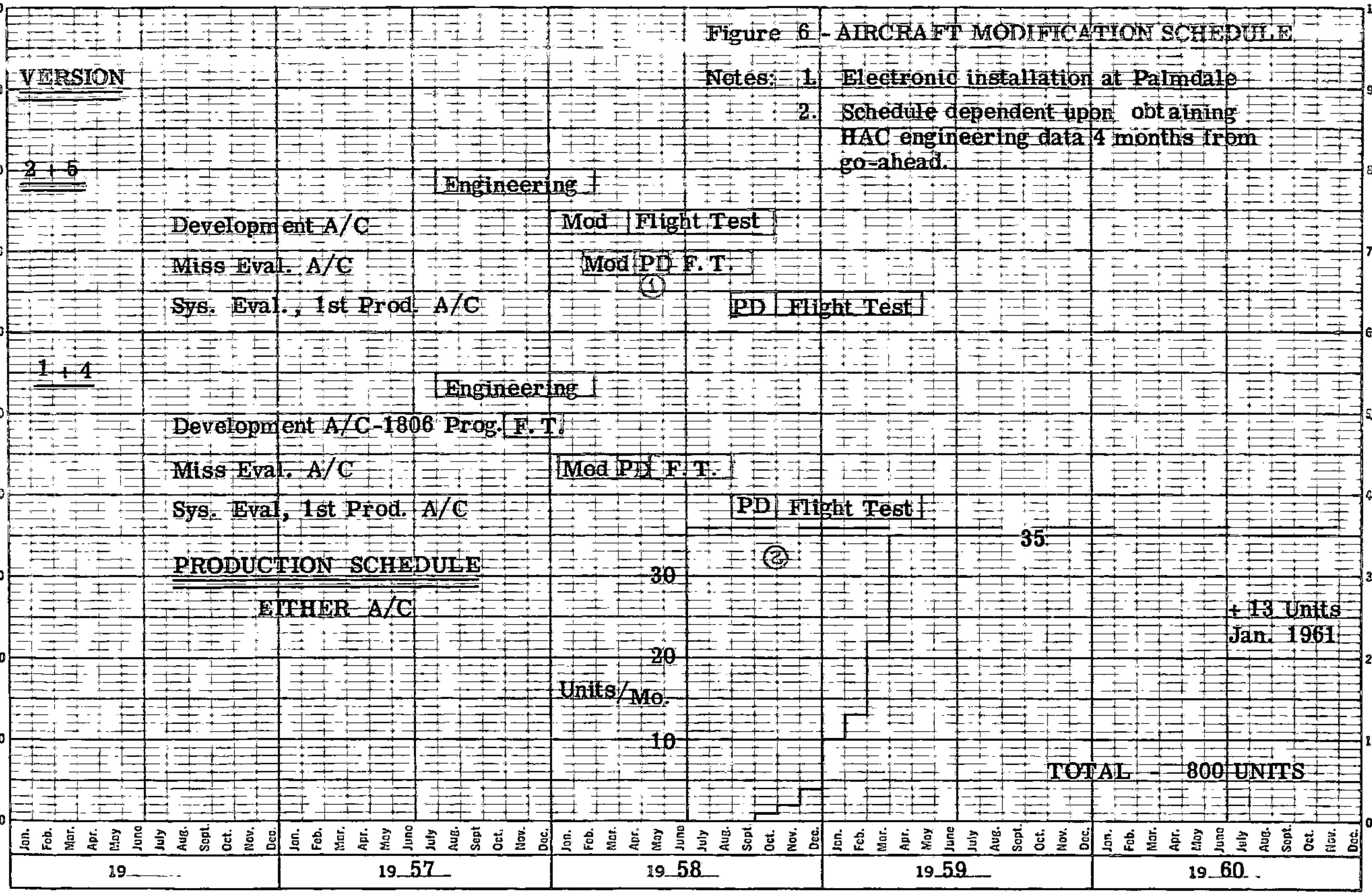

Figure 6 - AIRCRAFT MODIFICATION SCHEDULE

Notes:
1. Electronic installation at Palmdale
2. Schedule dependent upon obtaining HAC engineering data 4 months from go-ahead.

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ZP-8.058
ZP.8.058

C O N V A I R
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-058
DATE 10 March 1958
MODEL F-102A

[redacted]

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TITLE

A STUDY OF THE FERRY CAPABILITIES OF THE F-102A AIRPLANE WITH MISSILE BAY FUEL TANKS

PREPARED BY W. Ravenscroft
W. Ravenscroft
J. Callahan
J. Callahan

GROUP Preliminary Design

REFERENCE

CHECKED BY

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Preliminary Design

Propulsion Fluids

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SUMMARY

1. A gross range of 2490 Nautical miles can be obtained with the F-102A by the addition of 565 gallons of fuel in the missile bays in additon to 430 gallons in the external tanks. This range is sufficient for a 21% distance margin for the 2050 Nautical mile, San Francisco to Hawaii ferry mission. This range assumes Military power operation through take off and climb to cruise altitude. The use of afterburner in take off reduces range by approximately 40 Nautical miles.

2. 565 gallons of fuel in the missile bay is considered the maximum volume available consistant with concept of minimum airplane modification and fuel tank complexity.

3. The permanent weight increase of approximately 100 pounds in the area of the missile bay to the basic F-102A has a negligable effect on C.G. balance.

4. Failure of the missile bay tank transfer system will not cause airplane to become unmanageable during flight or landing.

5. To install the 565 gallon missile bay tanks, the following armament gear must be removed from each bay:

    a. Missile launcher displacement gear.
    b. Displacement gear up latch mechanisms.
    c. Displacement gear dampers.
    d. Displacement gear actuating cylinders.
    e. Missile launcher electrical disconnect panels.

6. Required manhours to remove armament gear and install missile bay tanks and fuel system is estimated at approximately 26 manhours with an elapsed time of 8 hours. (This assumes provisions for installing missile bay tanks and fuel system has been previously provided).

7. Permanent revisions to the Basic F-102A airplane to provide for the missile bay tank installation consist in general of the following:

    a. Local reinforcement of the missile bay beams to provide for the tank supports
    b. Holes in missile bay beams, fuselage skin and bulkhead, and forward wing tank rib for routing of fuel lines.
    c. Support provisions for the missile bay fuel system lines.
    d. Fuel vent lines from the forward wing tank into missile bay.
    e. Fuel transfer line from refueling line into missile bay.
    f. Missile bay boost pump switch and tank low level light installation in cockpit.
    g. Electrical wiring, relays, fuse panel.
    h. It is estimated 400 manhours and 10 days lay up will be required to accomplish the above tasks.

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INTRODUCTION

A study has been made to determine the feasibility of adding fuel tanks in the missile bays of the F-102A airplane to increase range for ferry purposes. This study includes range comparisons for the following configurations:

1. Basic F-102A
2. Basic F-102A with external tanks retained.
3. Basic F-102A with external tanks dropped.
4. F-102A with missile bay fuel.
5. F-102A with missile bay fuel and external tanks retained.
6. F-102A with missile bay fuel and external tanks dropped.
7. F-102A with missile bay fuel and enlarged external tanks retained.
8. F-102A with missile bay fuel and enlarged external tanks dropped.

The study considers various ferry missions, which might be accomplished by the above configurations.

The effect of adding additional fuel in the missile bays is investigated for the following items:

1. Airplane operational and structual limitations.
2. Airplane modification.
3. Additional equipment required.
4. Performance.
5. Safety.
6. Weight and balance.

This study is based on the assumption that airplanes, to receive missile bay tanks, will be modified on a retrofit program to install the provisions for the missile bay tanks and the fuel system, but tanks and plumbing in the missile bay will be installed at squadron level.

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DISCUSSION

The ferry missions considered in this study are based on information contained in a report by the Weapon Systems Integration Group on ferry routes to the NATO and SEATO countries.

NATO: North Atlantic Treaty Organization

There are two routes to the NATO countries: The northern via Greenland and the southern via the Azores.

Northern:

Boston to Goose Bay, Labrador	691 Nautical Miles
Goose Bay to BW8 (Greenland)	701 Nautical Miles
BW8 to Iceland	750 Nautical Miles
Iceland to Ireland	796 Nautical Miles

Southern

U.S. to Bermuda	763 Nautical Miles
Bermuda to Azores	1800 Nautical Miles
Azores to Port Lyautey, Morocco	1050 Nautical Miles

SEATO: South Eastern Asia Treaty Organization

Listed below is a route possibility:

San Francisco to Hawaii	2050 Nautical Miles
Hawaii to Johnson Island	850 Nautical Miles
Johnson Island to Kwajalein	1300 Nautical Miles
Kwajalein to Guam	1600 Nautical Miles
Guam to Okinawa	1300 Nautical Miles
Okinawa to Tokyo	850 Nautical Miles

An examination of these routes shows that the longest ferry mission is the one from San Francisco to Hawaii, a distance of 2050 Nautical miles. For a long ferry over water and against the prevailing winds, a minimum reserve of 20% of the ferry distance is considered a minimum requirement. From Table II of range comparisons configuration Number 6 shows a gross range of 2490 Nautical miles and gives a 21% distance reserve provided a Military power acceleration is used in take off and climb to initial cruise altitude. Afterburner take off reduces the distance reserve to approximately 19%.

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DISCUSSION (Cont'd)

The ferry mission from San Francisco to Hawaii requires that the 230 gallon external tanks be dropped, as soon as they are emptied.

Range comparison Number 7 was made to determine the amount of fuel necessary to carry the external tanks for the entire trip. It was found that the external tanks would have to be modified by the addition of 110 gallons. (See Tank Installation Drawing SD-58-08010). The practicability of adding larger external tanks has not been studied in this report and would require further investigation.* Configuration Number 8 shows potential range using larger external tanks and dropping them when empty.

The remaining SEATO ferries, including the long hop, of 1600 Nautical miles from Kwajalein to Guam can be accomplished with configuration Number 4, which includes missile bay fuel, but no external tanks. Configuration Number 4 gives a 29% distance margin on the Kwajalein to Guam ferry.

The northern NATO ferry missions can be accomplished by the basic F-102A with external tanks. Missile bay fuel is not required.

The southern NATO ferry route mission from Bermuda to the Azores of 1800 Nautical miles requires missile bay fuel, in addition to external tanks. By retaining the external tanks on the short ferry from the U.S. to Bermuda, they will be available for the long ferry hop. Configuration Number 6 gives a distance margin of approximatley 35% on the Bermuda to Azores ferry.

Table II summarizes the above discussion.

* Investigation would probably require ejection and flutter tests.

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TABLE I

MISSION	DIST. N. MI.	AIRPLANE CONFIG.	GROSS RANGE MIL.ACCEL. MIL CLIMB N. MI.	DIST. RESERVE %	GROSS RANGE A/B ACCEL MIL.CLIMB N. MI.	DIST. RESERVE %
Northern NATO						
Boston to Goose Bay	691	#2	1600	132	1560	126
Goose Bay to BW8	701	#2	1600	128	1560	122
BW8 to Iceland	750	#2	1600	114	1560	108
Iceland to Ireland	796	#2	1600	100	1560	96
Southern NATO						
U. S. to Bermuda	763	#5	2220	190	2180	186
Bermuda to Azores	1800	#6	2490	38	2450	36
Azores to Morocco	1050	#4	2070	97	2030	95
SEATO						
San Francisco to Hawaii	2050	#6	2490	21	2450	19
Hawaii to Johnson Is.	850	#4	2070	144	2030	139
Johnson Is. to Kwajalein	1300	#4	2070	59	2030	56
Kwajalein to Guam	1600	#4	2070	29	2030	27
Guam to Okinawa	1300	#4	2070	59	2030	56
Okinawa to Tokyp	850	#4	2070	144	2030	139

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RANGE COMPARISON

Table II summarizes the gross ranges obtainable with the F-102A airplane for the following fuel load configurations.

#1 Basic F-102A with internal wing fuel.
#2 Basic F-102A with internal wing fuel plus 2 - 230 gallon external tanks retained during entire flight.
#3 Basic F-102A with internal wing fuel plus 2 - 230 gallon external tanks dropped after emptied (External tank fuel used first).
#4 F-102A with internal wing fuel and 565 gallons of fuel in missile bay.
#5 F-102A with internal wing fuel and 565 gallons of fuel in missile bay plus 2 - 230 gallon external tanks retained during entire flight.
#6 F-102 with internal wing fuel and 565 gallons of fuel in missile bay plus 2 - 230 gallon external tanks dropped after emptied (External tank fuel used first).
#7 F-102A with internal wing fuel and 565 gallons of fuel in missile bay plus 2 - 340 gallon external tanks retained during entire flight.
#8 F-102A with internal wing fuel and 565 gallons of fuel in missile bay plus 2 - 340 gallon external tanks dropped after emptied (External tank fuel used first).

The range calculation were based on the following assumptions:

1. F-102A with case X wing.
2. Pratt and Whitney J57-P23 engine.
3. Fuel flow increased by 5%.
4. Standard day.
5. No wind allowance.
6. Gross range (no reserve).
7. Military power acceleration from start of take off run to climb speed and during climb to initial cruise altitude.

   After burner acceleration to climb speed uses approximately 285 pounds more fuel, which reduces the range by approximately 40 Nautical miles.

8. Optimum cruise-climb condition.

[redacted]

TABLE II

Reserve fuel = 0
Internal wing fuel = 1,070 gal.

Take-off allowance = 575 lbs.

AIRPLANE CONFIG.	TAKE OFF WEIGHT LBS:	MISSILE BAY FUEL GAL.	EXT: TANK FUEL GAL.	TAKE OFF GROUND RUN FT.	GROSS RANGE MIL.ACCEL. N. MI.
1	28000	0	0	3800	1300
2 *	31100	0	430	4800	1600
3 ‡	31100	0	430	4800	1780
4	30500	565	0	4600	2070
5 *	33700	565	430	5700	2220
6 ‡	33700	565	430	5700	2490
7 *	35150	565	652	6200	2430
8 ‡	35150	565	652	6200	2700

* External Tanks Retained During Entire Flight
‡ External Tanks Emptied First and Dropped When Empty.

Note: 1. Above data based on no wind and standard day temperatures.
2. Mil. power used for T.O., acceleration to climb spd., and climb to initial cruise altitude.
3. Use of afterburner for acceleration to climb speed would reduce each of the above range figures by 40 n. mi.
4. The smaller external tanks each have a capacity of 230 gallons but only 215 gallons are useable.

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FUEL SYSTEM

The fuel system has been considered on the basis of the following points:

1. Existing airplane structure modification to be a minimum.
2. Revision to existing fuel system be a minimum.
3. Permanent changes to airplane be a minimum.
4. Ease of installation and maintenance
5. Reliability
6. Safety
7. Airplane weight and balance
8. Single point refueling
9. Pilot management of fuel sequencing of the missile bay tanks.

The missile bay tanks are of welded aluminum sheet metal construction consisting of skin and internal stiffining baffles. The baffles also contribute to fuel slosh control. The tanks are supported from the missile bay beam by means of form fitting cradles and straps. This mounting provides for easy tank installation, maintenance, and prevents introduction of fuselage structural loads into the tanks. Fore and aft loads will be taken out by drag links from the cradles to the missile bay beam.

Six (6) tanks are installed in the missile bays. The three (3) forward tanks are interconnected to function as one forward missile bay tank. The three (3) aft tanks are interconnected to function as one aft missile bay tank. To maintain proper airplane center of gravity control during flight the fuel sequencing is as follows:

1. External tanks
2. Forward missile bay tank
3. Aft outboard wing tanks
4. Forward wing tanks to total wing quantity of 3300 pounds
5. Aft missile bay tanks
6. Forward wing tanks
7. Aft inboard wing tanks

For C.G. curve see Figure I.

Fuel is transfered from the missile bay tanks to the wing tanks by means of a fuel transfer line that tee's into the existing wing refueling system. One (1) boost pump installed in each of the forward and aft missile bay tanks* pump the fuel into the wing tanks. The existing refueling shutoff valve in the aft inboard wing tank and float pilot valve in the aft outboard wing tank will prevent missile bay fuel from over filling the wing tanks.

A low fuel level switch in the missile bay tanks will indicate to the pilot by means of a light in the cockpit when they are empty. The transfer boost pumps

* Two (2) boost pumps total per airplane.

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FUEL SYSTEM (Cont'd)

are controlled manually by the pilot by means of a switch in the cockpit.

The pilot manages the missile bay transfer system as follows:

When the external tanks are depleted, the pilot turns on the forward missle bay transfer pump. The aft missile bay transfer pump remains off. The fuel flows into the wing tanks through the refueling lines and is controlled by the float pilot valve in the aft outboard wing tank. The fuel from the missile bay tank replaces fuel used from the aft inboard wing tank as it is burned by the engine. When the forward missile bay tanks is empty, as indicated by the low level light, the pilot turns off the forward missile bay pump. The airplane is flown using wing tank fuel until the total wing fuel quantity is approximately 3300 pounds. At this time, the pilot turns on the aft missile bay transfer pump and tranfers missile bay fuel into the wing tanks. The aft missile bay tank is emptied in a short time, refilling the forward wing tanks (the float valve in the aft outboard wing tank does not control this transfer). When the aft missile bay tank low level light indicates that the tank is empty, the pilot turns off the aft transfer pump. The airplane is flown the remainder of the mission using wing tank fuel.

A tank venting system provides proper pressure relief and vacuum relief for the missile bay tanks. The vent lines are routed through the fuselage skin and forward wing tank closing rib and tee's into the existing forward wing tank vent line. This vent exits through the lower surface of the wing outboard of the wheel wells and minimizes the hazard of fuel re-entry into wing or fuselage cavities in case of fuel spillage from the vent. This location also minimizes vent pressure variations.

The missile bay tank refueling is accomplished simultaneously with wing tank refueling through the missile bay transfer line. A regulator in the refueling line reduces the refueling pressure prior to entry into the missile bay tanks. The fuel enters the missile bay tanks through a shutoff valve controlled by a float pilot valve. When the proper quantity of fuel is in the tanks, the shutoff valve closes stopping refueling. A refueling pressure operated vent valve opens the missile bay tanks to atmospheric pressure through the vent line to prevent pressure build up during refueling. This valve closes after refueling pressure is disconnected.

Defueling may be accomplished by pumping the fuel out of the missile bay tanks into the wing and defueling via the existing defuel point or by draining through the low point drains in the bottom of each missile bay tank.

The missile bay is ventilated by continuous air flow from the cabin pressure system through the missile bay and exiting around the missile bay doors. This prevents possible fuel fumes from collecting in the bay.

Failure of fuel to transfer from the missile bay will affect range and cause C.G. of airplane to shift forward as fuel is used from wing tanks, but with the low load factors encountered during ferry operation, no problem is anticipated in flying and landing the airplane.

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AIRPLANE MODIFICATION

A. In order to obtain adequate fuel volume in the missile bays, it is necessary to remove the following equipment at time of tank installation:

1. Missile launcher displacement gear
2. Displacement gear up latch mechanisms
3. Displacement gear dampers
4. Displacement gear actuating cylinders
5. Missile launcher electrical disconnect panels

B. To provide for the tank and fuel system installation, the following modifications are necessary:

1. Fuselage

    a. Rework missile bay beam locally to provide for tank mounting provisions.
    b. Rework missile bay beam to allow passage of fuel lines between tanks.
    c. Rework bulkhead at STA. 405.75 to allow passage of missile bay tank fuel transfer line.
    d. Rework skin at W.L. - 16.00 and approximately STA. 380 to allow passage of fuel vent line into wing tank vent system.

2. Pneumatics

    a. Reroute line from R.H. air flasks to clear missile bay fuel tanks.
    b. Cap air supply to missile gear actuators.

3. Wing

    a. Rework forward wing tank closing ribs at approximately STA. 380 to provide for missile bay fuel tank vent line by adding unreinforced hole.

4. Fuel System

    a. Revise forward wing tank vent line to provide for "TEE" connection to missile bay tank vent line.
    b. Revise defueling line in the fuselage section to provide for "TEE" connection to the missile bay tank fuel transfer line.

5. Electrical

    a. Revise missile launcher electrical disconnect panels to allow removal at time of tank installation.

[redacted]

AIRPLANE MODIFICATION (Cont'd)

C. Equipment and parts added to the airplane:

1. Permanent additions to the basic F-102A

   a. Fuel System

      (1) Vent lines from forward wing tanks into missile bay - lines to be capped when not in use.
      (2) Bulkhead fitting through fuselage bulkhead at STA. 405.75 - to be capped when not in use.
      (3) Mounting provisions for the missile bay tanks and fuel lines.

   b. Electrical

      (1) Wiring for missile bay tank boost pumps and switches
      (2) Wiring for missile bay tank low level switch
      (3) Wiring for missile bay tank switch and lights in cockpit
      (4) Support provisions for boost pump relays
      (5) Fuse panel.

   c. Furnishings

      (1) Missile bay boost pump switch and low level light installation.

2. Additions to the airplane at time of missile bay tank installation:

   a. Six (6) tanks in forward and aft missile bays.
   b. Interconnectors between missile bay tanks.
   c. Fuel transfer system from forward and aft missile bay tanks to aft inboard wing tanks.
   d. Missile bay tank refueling system.
   e. Missile bay tank vent system.

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OPERATIONAL LIMITATIONS

The airplane will have the following limitations imposed:

1. Missile bay tanks full of fuel
    - a. Load factors are limited to +5 G's and -2.2 G's in flight, while the gross weight exceeds 25,000 pounds.
    - b. Missile bay doors to be closed during flight.
2. Missile bay tanks empty.
    - a. Load factors are limited to +7 G's and -3.0 G's in flight.
3. External tanks "ON" (230 Gal. Tanks)
    - a. Load factors are limited to +3.0 G's and -1.0 G's in flight.
    - b. Maximum speed is limited to M = .95.
4. 340 Gallon Ext.

    The pylon and attach bulkhead will probably require strengthening to operate at the restrictions of the 230 gallon tanks.

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WEIGHT AND BALANCE

A. Weight

The weight permanently added to the airplane modified to receive the missile bay tanks is due to:

1.	Missile bay beam rework to provide for fuel tank mounting and fuel line routing provisions.	60 pounds
2.	Fuselage rework for passage of fuel lines	10 pounds
3.	Missile bay tank fuel transfer provisions	4 pounds
4.	Missile bay tank vent provisions	8 pounds
5.	Electrical wiring, fuse panel, switches	12 pounds
	TOTAL	94 pounds

B. Weight and Balance

1. With missile bay tanks installed, no ballast will be required for the configuration discussed in this report. See Figure I for C.G. Curve.
2. With missile bay tanks removed and armament gear installed, the balance of the airplane is not appreciably changed from the basic F-102A airplane and should be ballasted in accordance with F-102A requirements. The additional permanent weight improves the take off C.G. condition with external tanks installed.
3. The airplane cannot be flown with missile bay tanks empty until ______ fuel has been used from the external tanks and the aft outboard wing tanks. If the airplane is to be flown with missile bay tanks and fuel system installed, but deactivated, a ballast of 1000 pounds (154 gallons) of fuel is required in the forward missile bay tank. (See C.G. Curve Figure II)

[redacted]

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TANK INSTALLATION AND REMOVAL TIME

The following estimated times are for squadron level installation of the missile bay tanks and fuel system. It is assumed that the airplane has been previously modified to receive the tank installation:

1.	Remove armament equipment	6 man hours
2.	Install tank supports	6 man hours
3.	Install tanks	4 man hours
4.	Connect fuel and electrical systems	8 man hours
5.	Inspection and systems checkout	2 man hours
	TOTAL	26 Man Hours

Assuming a 4 man crew for installation work and 2 men for inspection and checkout, the total elapsed time would be about 8 hours.

To remove the tank installation and reinstall the armament gear, it is estimated that approximately the same number of man hours will be required.

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FIGURE I

F-102A FERRY CONDITION

565 Gallons of Fuel in Missile Bays

LEMAC = 335.9
MAC = 285.1

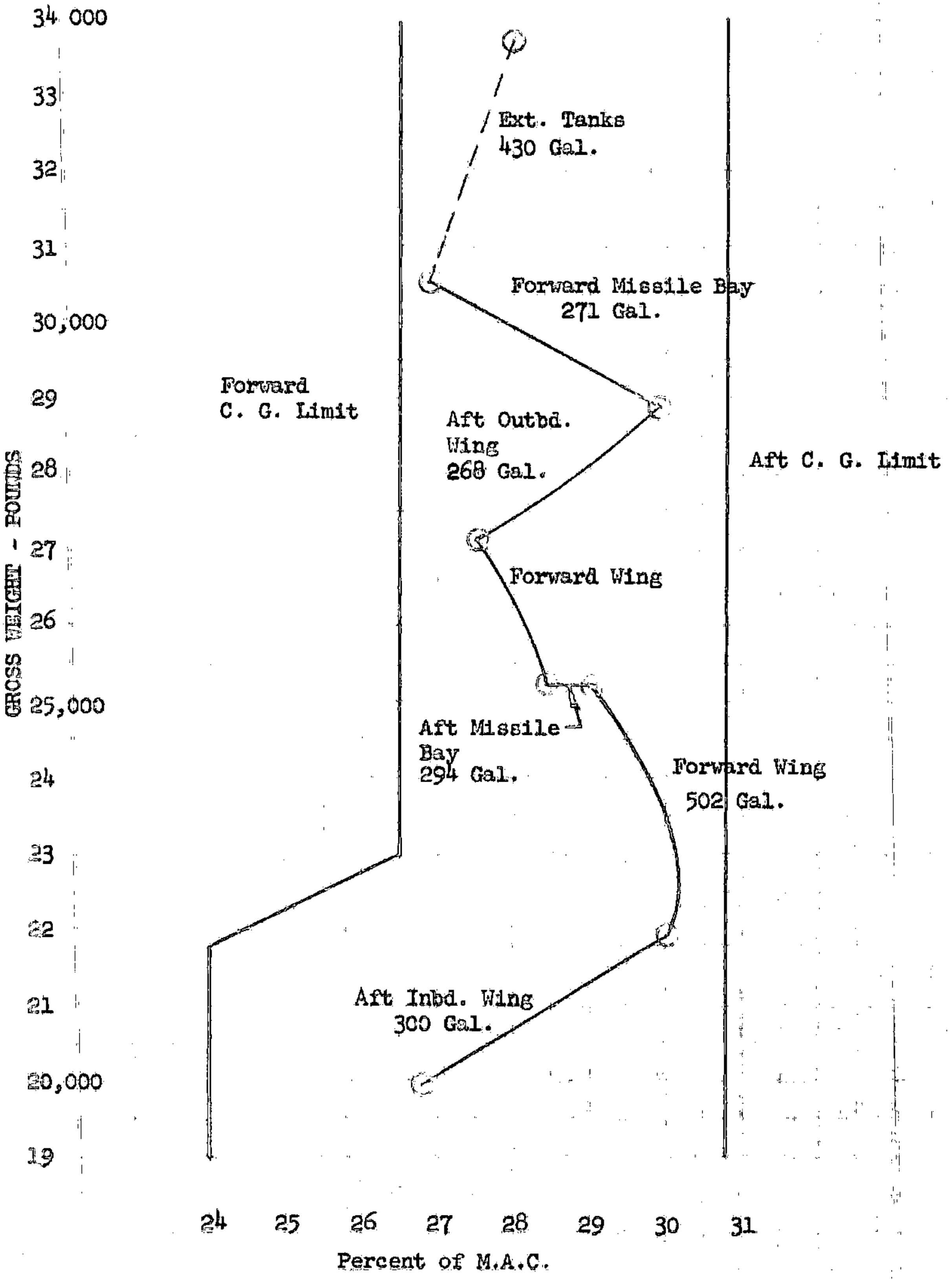

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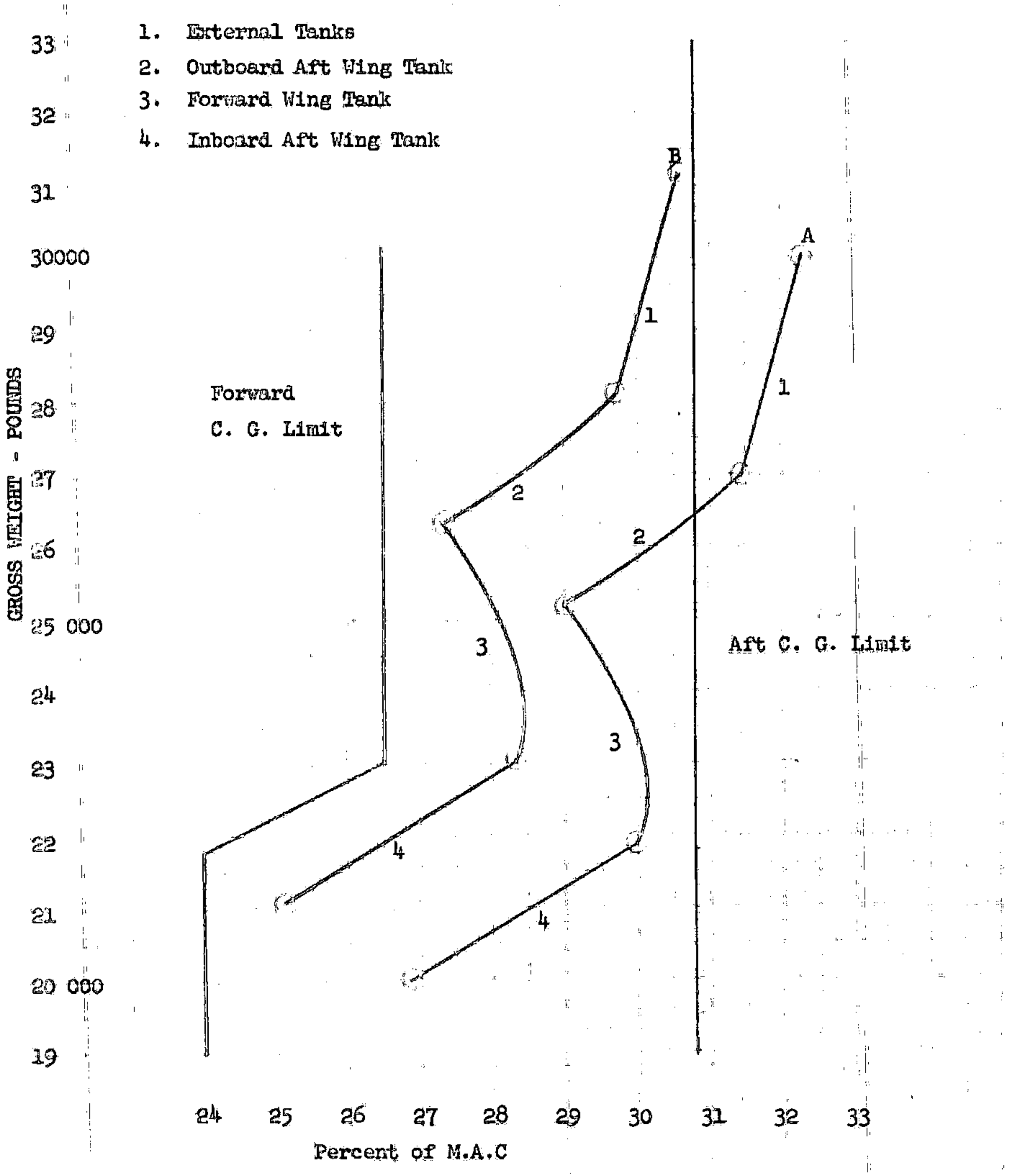
FIGURE II

F-102A FERRY STUDY

C. G. Envelope With Missile Bay Tanks Empty

Curve A: Missile Bay Tanks Empty LEMAC = 335.9

Curve B: 1000 Pounds of Fuel Ballast in Forward Missile Bay Tank MAC = 285.1



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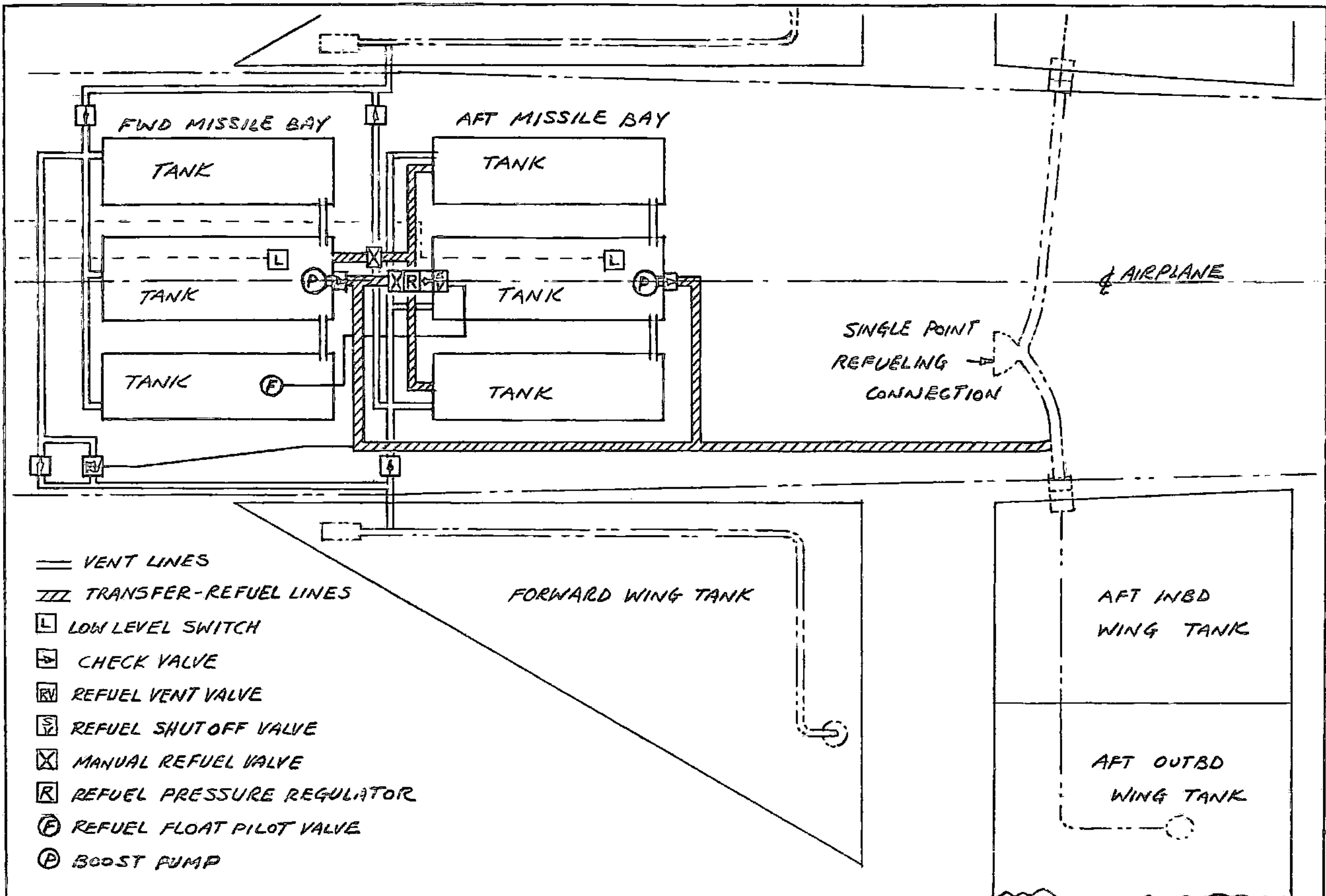

FIGURE III - FUEL SYSTEM DIAGRAM - F102A FERRY

[redacted]

ADDENDUM I

TO

ZP-8-058

MISSILE BAY TANKS FOR MISSION RADIUS EXTENSION

F. L. TRIPP

[redacted]

DISCUSSION

Alternate utilization of the missile bay ferry tanks permits an extension of the F-102A combat radius. For this configuration, the armament in the center bay is retained and fuel tanks are installed in the outboard bays. The (24) 2.75" FFAR rockets may also be retained with the addition of rocket blast protection over the fuel tanks.

The armament loaded in the center missile bay can be two GAR-1/2 or improved versions of the GAR-1 (Reference Convair Report ZP-8-057) when available.

Fuel capacity of the F-102A is increased by 298 gallons with the addition of the outboard missile bay tanks. This yields a gross range of 1710 n. miles (Reference Page 8). For a subsonic cruise out and back mission with two minutes combat and 2000 lb. landing reserve, this fuel loading results in a combat radius of 565 n. miles. The basic airplane without external tanks has a radius of 366 n. miles for the same mission.

The fuel system installation in the airplane is essentially the same as for the ferry extension installation. However, the tanks will require added provisions for their use in combat range extension. The system requires the same tie-in to the existing A/C system as before as may be seen in comparing Figure 1.A with Figure III. The 4-tank system requires the addition of boost pumps and low level switches to each of the tanks plus a refuel float pilot valve to the right-hand forward tank.

The provisions for these items can be added to the tanks of the ferry fuel installation.

Travel of the airplane C.G. and fuel sequence is shown in Figure 2.A.

Fuel is transferred from the missile bay tanks to the wing tanks by a transfer line tee'd into the existing wing refueling system. The existing refueling shut-off valve in the aft inboard wing tank will prevent missile bay fuel from over filling the wing tanks.

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The missile bay transfer system is controlled by the pilot as follows: When the external tanks are depleted, the pilot turns on the forward missile bay transfer pumps. The aft missile bay transfer pumps remain off. The fuel flows into the wing tanks through the refueling lines and is controlled by the float pilot valve in the aft outboard wing tank. The fuel from the missile bay tank replaces fuel used from the aft inboard wing tank as it is consumed by the engine. When the forward missile bay tanks are empty, as indicated by the low level light, the pilot turns off the forward missile bay pumps. Normal operation follows until the wing fuel quantity is approximately 3300 lbs. The pilot then turns on the aft missile bay pumps and transfers the fuel into the wing tanks. The aft missile bay tanks are emptied in a short time, refilling the forward wing tanks. When the aft missile bay tank low level light indicates that the tank is empty, the pilot turns off the aft transfer pumps. The remaining wing tank fuel is then used.

A tank venting system provides proper pressure and vacuum relief for the missile bay tanks. The vent lines tee into the existing forward wing tank vent line.

Missile bay tank refueling is accomplished simultaneously with wing tank refueling through the missile bay transfer line. A regulator in the refueling lines reduces the pressure prior to entry into the missile bay tanks. When the tanks are full, the fuel flow is stopped by the refueling shut-off valve which is controlled by the float pilot valve. A refueling pressure operated vent valve opens the tanks to atmospheric pressure to prevent pressure build up during refueling.

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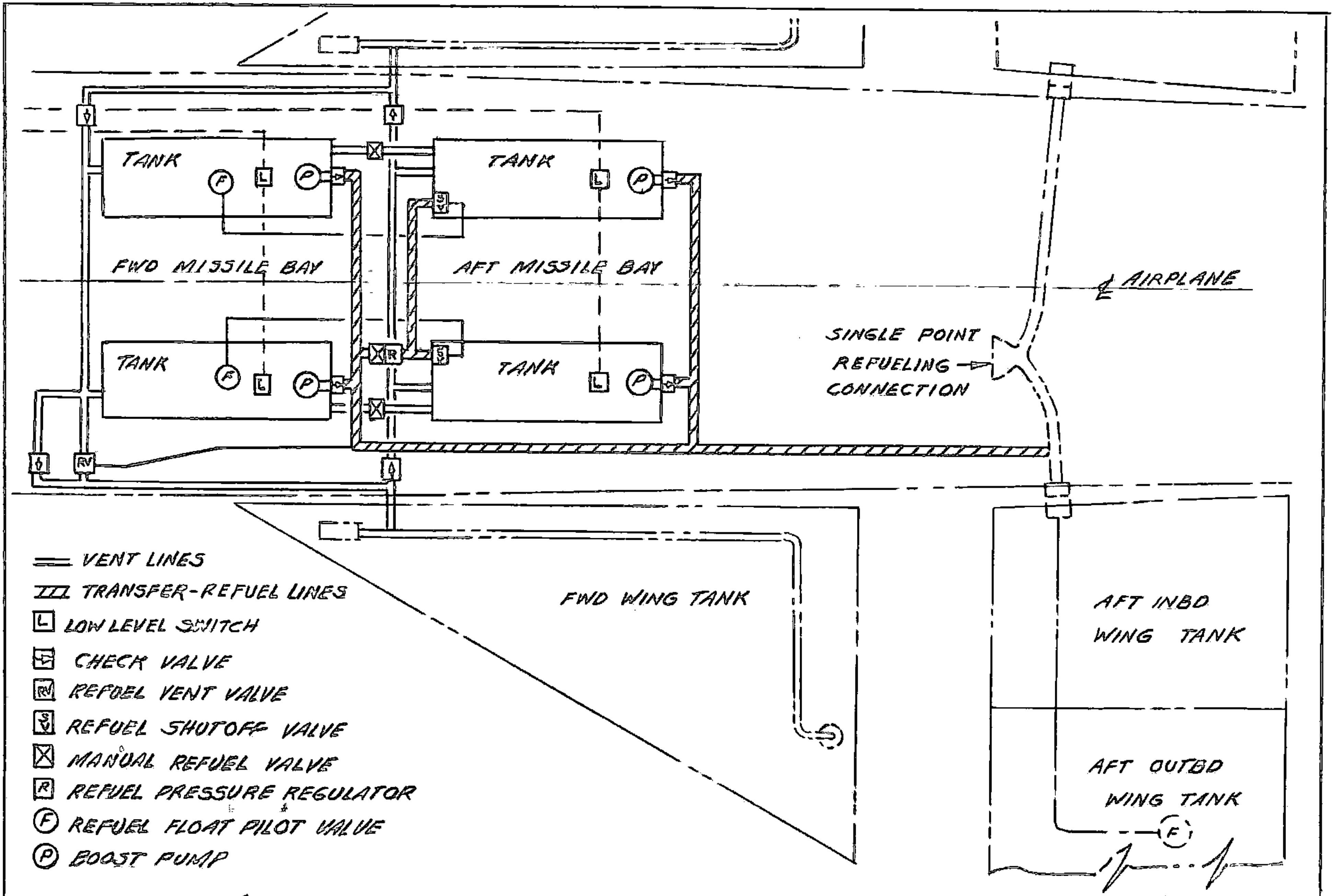


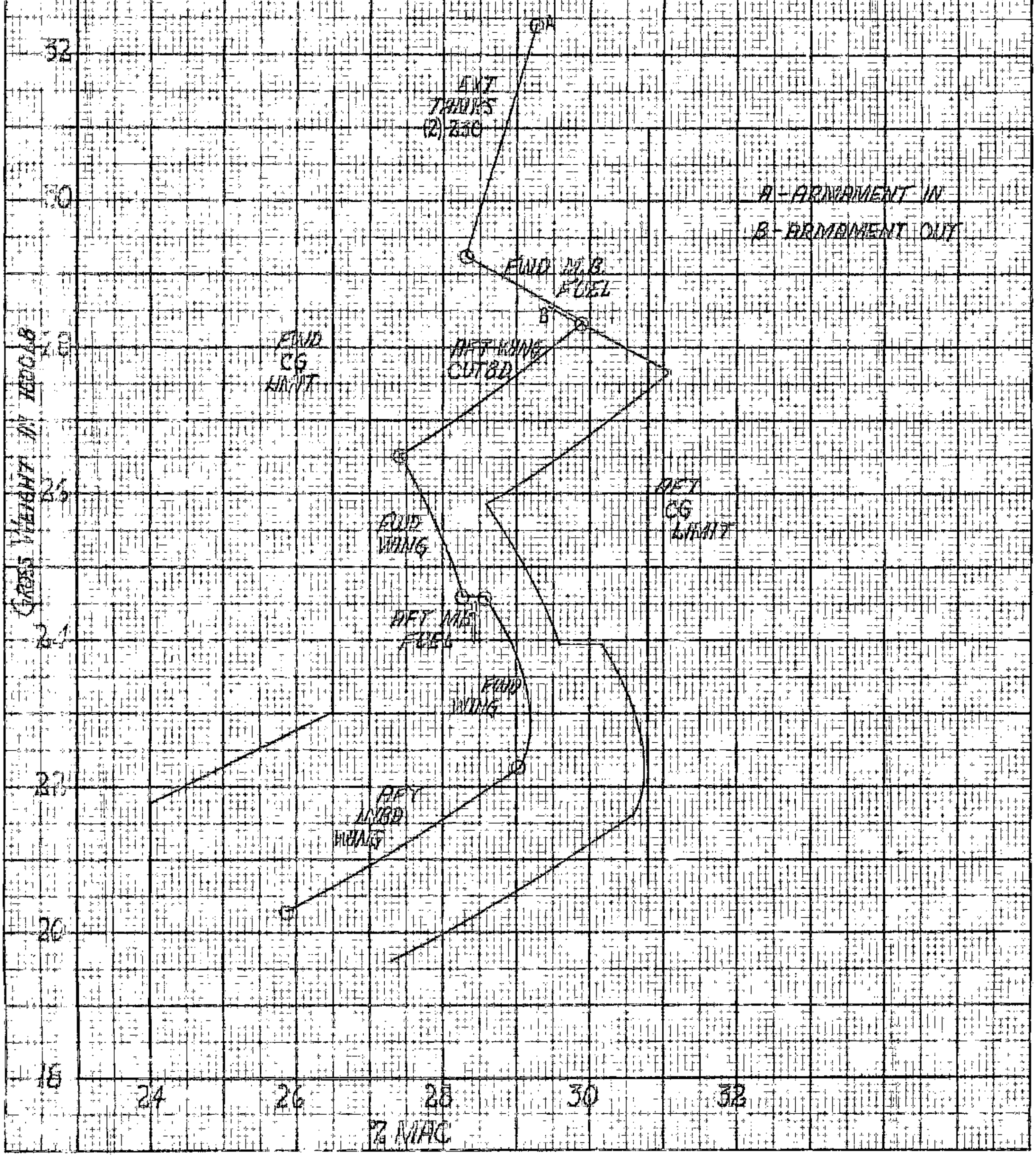
FIG. 1.A FUEL SYSTEM DIAGRAM-F102A - 4 MISSILE BAY TANKS (PUMPS)

FIG. 2.A

C.G. ENVELOPE

CONDITIONS

1. (2) GAR-1/2 AND (24) 2.75" FFAR
2. 293 GAL USABLE M.B. (MISSILE BAY) FUEL
3. 100 LB BALLAST
4. F-102-A, 459 CONFIGURATION



10 X 10 TO THE ½ INCH 359-11 KEUFFEL & ESSER CO.

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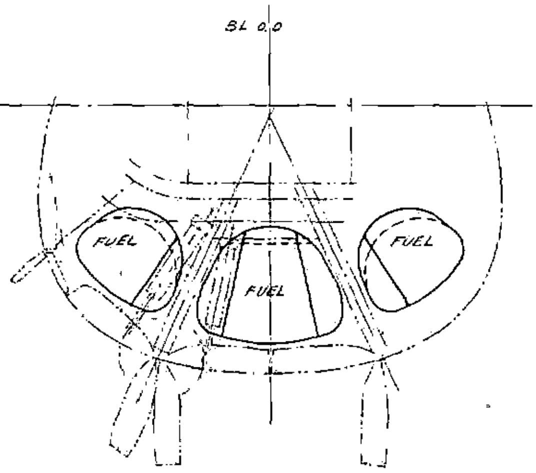
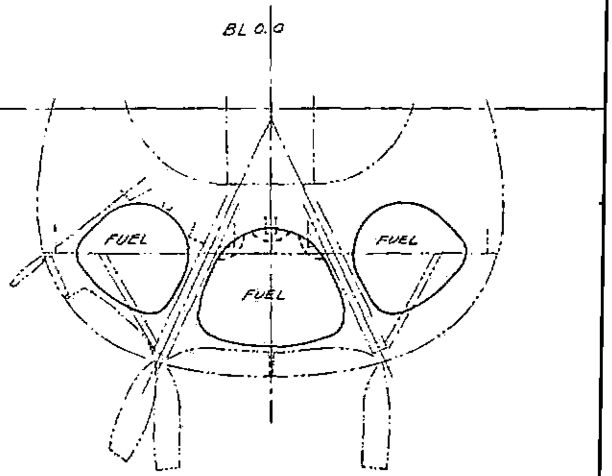
2D-28-08010

[redacted]

3



VIEW A-A

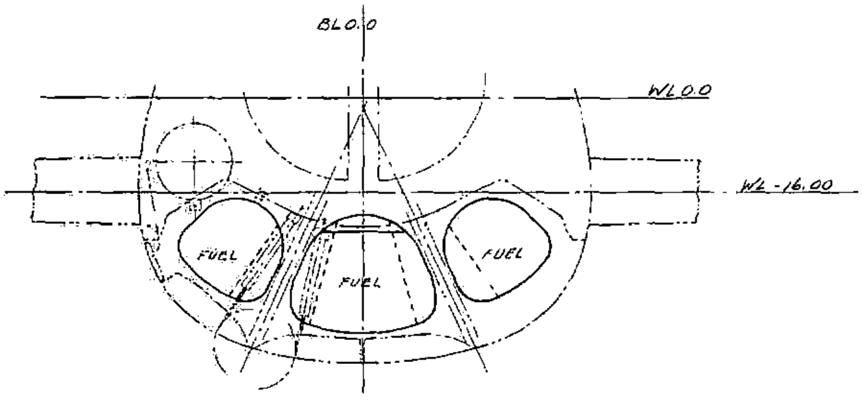


VIEW B-B

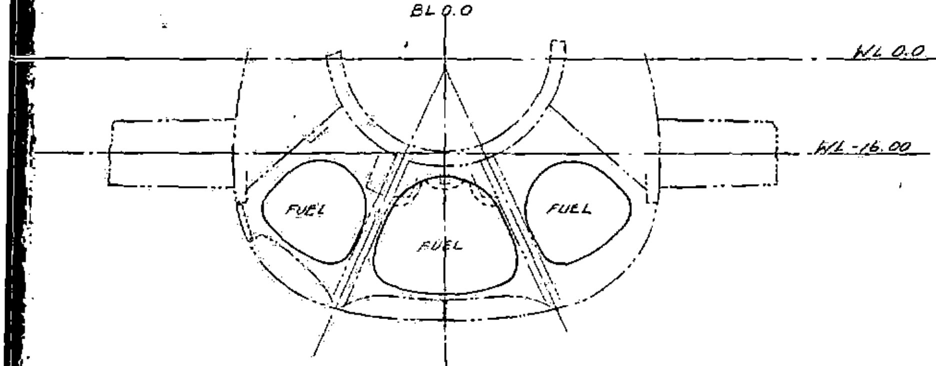
[redacted]

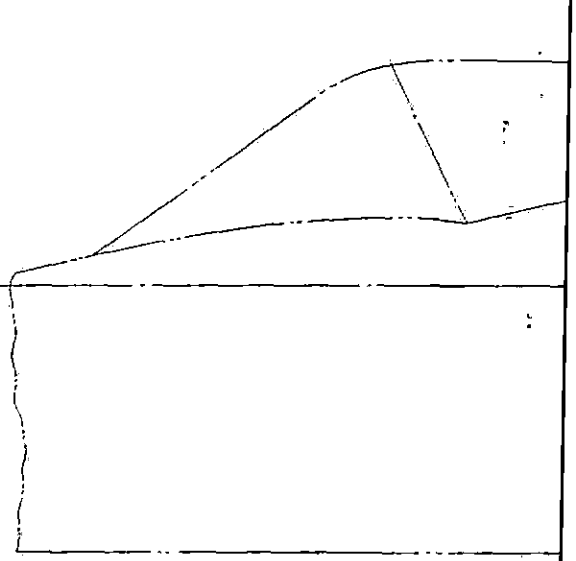
FUEL CAPACITIES

FORWARD MISSILE BAY	271 GALLONS
AFT MISSILE BAY	294 GALLONS
TOTAL	565 GALLONS

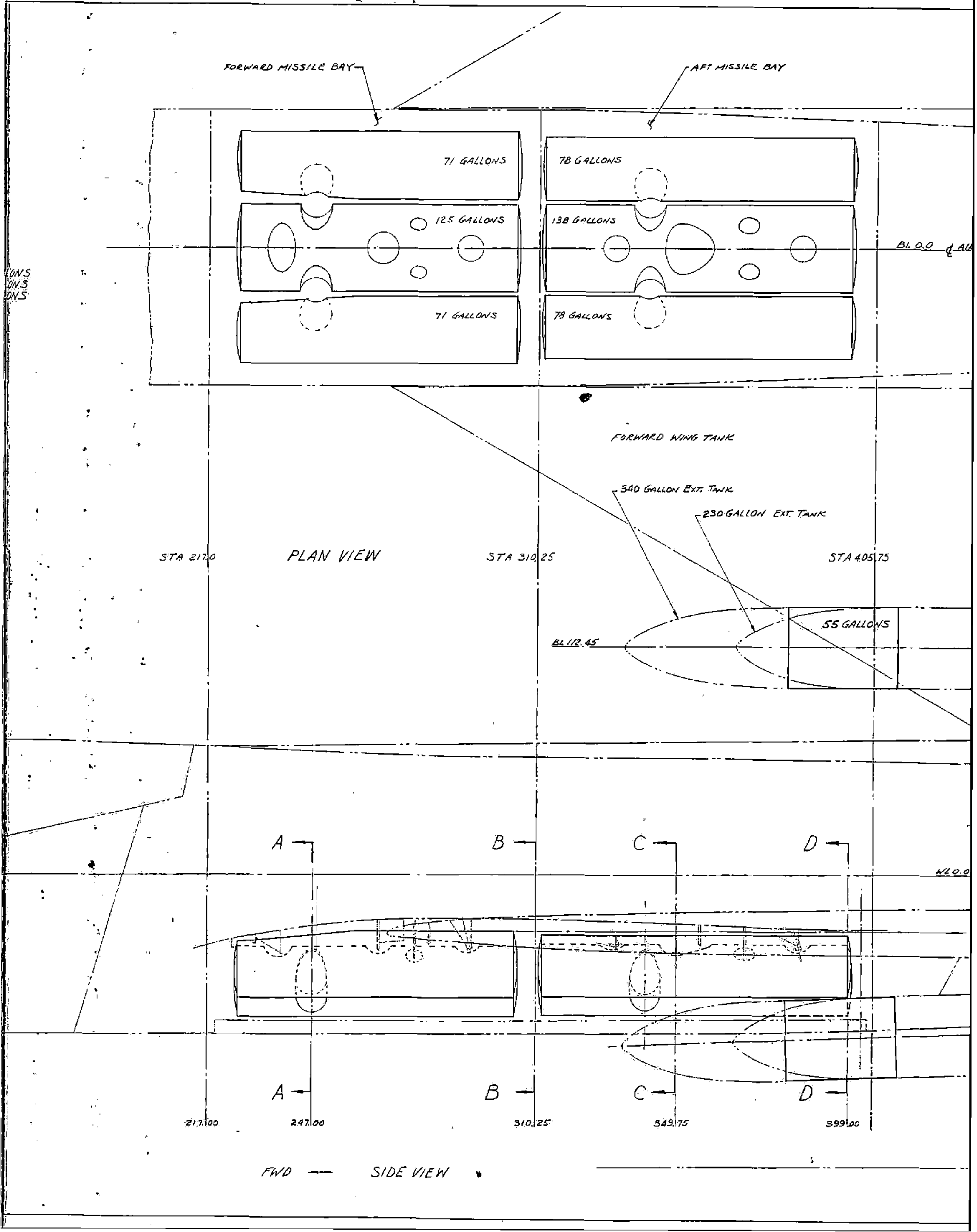


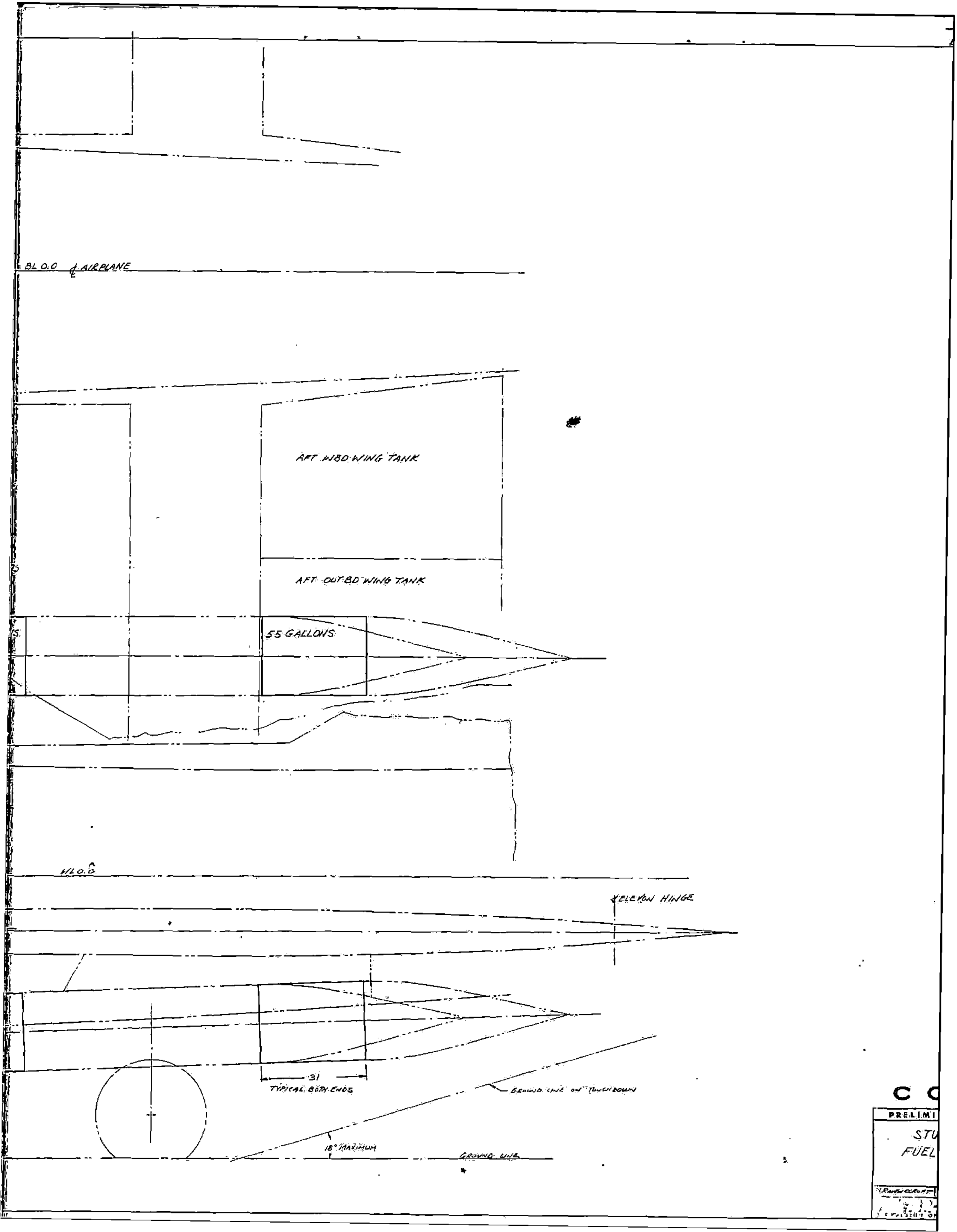
VIEW C-C





VIEW D-D


FORWARD MISSILE BAY
AFT MISSILE BAY
71 GALLONS
78 GALLONS
125 GALLONS
138 GALLONS
71 GALLONS
78 GALLONS
BL 0.0
LONS
ONS
ONS
FORWARD WING TANK
340 GALLON EXT. TANK
230 GALLON EXT. TANK
STA 217.0
PLAN VIEW
STA 310.25
STA 405.75
55 GALLONS
BL 112.45
A
B
C
D
WL 0.0
A
B
C
D
217.00
247.00
310.25
349.75
399.00
FWD
SIDE VIEW


BL 0.0 ℄ AIRPLANE
AFT INBD WING TANK
AFT OUTBD WING TANK
55 GALLONS
WL 0.0
℄ ELEVON HINGE
31
TYPICAL BOTH ENDS
GROUND LINE ON TOUCHDOWN
18° MAXIMUM
GROUND LINE
C C
PRELIMI
STU
FUEL

ZP-8-058

Page 14

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CONVAIR

PRELIMINARY DESIGN [illegible]

STUDY- F102A FERRY
FUEL TANKS IN MISSILE BAYS

1/10 3-5-58

—SAN DIEGO,
DIVISION OF GENERAL DYNAMICS CORPORATION

SD-58-08010

ZP-8-061

ZP-8-061

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C O N V A I R
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

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[redacted]

REPORT ZP-8-061
DATE 5 August 1958
MODEL F-102A

TITLE

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Modernization
Program
ZP-8-061
FEASIBILITY STUDY ONLY
FOR INTERNAL USE

5 August 1958

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F.L. Tripp

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY C R Tuttle

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NO.	DATE	BY	CHANGE	PAGES AFFECTED
1	Aug.27,"58	F.L.T.	Temp.Design Limit; Fwd & Rear Hem. Att.	14, 19, 21
2	Aug.27,'58	F.L.T.	App.C.; Mixed Loading Configuration	18.Add 32,33,34

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FORM 1612A-4

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II WEAPON SYSTEM IMPROVEMENTS	2
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APPENDIX A c.g. Envelopes	24
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SUMMARY

The cumulative effects of a series of previously proposed improvement items for the F-102A are described in terms of the aircraft performance and the weapon system effectiveness.

Aircraft performance is greatly improved by the JT-3C-22, the Vmax increasing from M = 1.18 to M = 1.6. Supersonic operational higher altitudes are also realized.

System effectiveness is increased by the addition of extended radar capability, CCM provisions and an infrared search and track system.

The number of interceptors that can be brought into Battle can be increased by means of the higher speed and longer range versions considered.

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I. INTRODUCTION

A series of individual improvements has been offered on the F-102A over the past several months. These improvements have consisted of aerodynamic, armament and fuel variations, and various systems improvements.

These improvements, exclusive of armament changes and varied fuel loadings, along with the installation of a higher-thrust, lower-weight engine are incorporated together into an improved F-102A. Armament changes and varied fuel loadings are then applied to this improved F-102. Performance effects are shown for these various configurations and compared to the tactical F-102A.

In addition, the effects of these improvements on the weapon system effectiveness are outlined.

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II. Weapon System Improvements

The basic improvements included in this modernization program provide greater aerodynamic and systems capabilities. Aerodynamic gains are realized by the installation of the P&W JT-3C-22 engine and the replacement of wing fences by the wing leading edge slots. The JT-3C-22 engine is a lighter-weight, higher-thrust version of the J-57.

Systems improvements are realized by the incorporation of extended radar and counter-countermeasure capabilities, the installation of an infrared search & track system, and the incorporation of improved navigational components.

The weapons system contribution of these improvements is discussed in the following section, along with a series of alternate missile bay and armament loadings that can effectively contribute to the aircraft capabilities under varying tactical conditions.

The inboard profile (Dwg. SD-58-08012) illustrates the general areas of the aircraft that are affected by the incorporation of these improvements and some of the possible missile bay loadings. The airplanes that have been described for comparison are:

A. Tactical F-102A (6-GAR-1/2 & 24-2.75" FFAR)

B. Tactical F-102A plus the aerodynamic and systems improvements (6-GAR-1/2 & 24-2.75" FFAR).

C. Tactical F-102A plus improvements, missile bay loading of 694 gal. fuel and an external armament loading of (2) MB-1 rockets.

D. Tactical F-102A plus improvements and a missile bay loading of (1) MB-1 rocket in the center missile bay (forward armament position) and 454 gal. fuel in the outer missile bays.

E. Same as A/C D except the MB-1 is replaced by (2) GAR-11 missiles.

F. Trailer & identification type F-102 (with improvements) full missile bay fuel (694 gal.) and with external tanks.

[redacted]

A. Performance

Various performance factors for the five configurations of the Improved F-102 and for the current tactical F-102A are shown in Figures 1 thru 7. A direct comparison of the aerodynamic improvements is illustrated by comparing configuration A (the tactical F-102A) to configuration B (J57-P-23 engine replaced by the JT-3C-22 engine and wing fences replaced by wing leading edge slots).

Performance data includes:

Fig. 1	Take-Off Distance (Mil + A/B Thrust)
Fig. 2	Time to Climb (40,000 & 50,000 ft.) (Mil + A/B Thrust)
Fig. 3	Time to Climb to 35,000 ft. (Mil thrust)
Fig. 4	Speed-Altitude Envelope (Mil. + A/B Thrust)
Fig. 5	Speed-Altitude Envelope (Mil Thrust)
Fig. 6	Subsonic Mission Radius
Fig. 7	Dash Mission Radius & Time to Intercept

The performance of the Improved F-102 is greatly increased over that of the tactical F-102 specifically in regards to take-off distance, minimum time to climb, and speed altitude capabilities.

The speed-altitude envelope is shown with respect to the flight envelope capabilities and extends beyond the original design criteria of the F-102A. However, it is considered feasible to operate beyond the $M = 1.5$ region with the retention of the existing "q" design limits. The maximum design temperature for the F-102A exists on the "q" limit at $M = 1.5$. However, this area is below the sustained flight capabilities of the airplane where the stagnation temperature is lower. Hence, on incorporation of these performance improvements, an investigation would be required to determine the time-temperature history of a maneuver carried out in this area. Should temperatures become critical, a constant stagnation temperature line could be incorporated into the flight limit line as in the F-106 A/C ($M = 1.9$ at 30,000 ft. to $M = 2.0$).

With regard to the aircraft subsystems, there is little or no additional electrical loads, no new problems of air conditioning or

[redacted]

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heat rejection created by the installation of the JT-3C-22 engine in the F-102A and the attendant increase in the Mach number capability envelope. In general, in the areas where the small increases do occur, they are localized special operating conditions which can be corrected by limiting operation in these areas.

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B. Systems Improvements

1. Extended Radar Capability - The extended radar capability has been outlined in CPO 58-173, "MG-10/MG-10T Fire Control System - Range Extension and Improved Target Discrimination". In essence, the changes and the significance of these changes are as follows:

a. A new glare shield is added to reduce the amount of ambient light striking the scope which will therefore aid in target detection and discrimination.

b. The range of the data link presentation is extended to make the data link display compatible with the radar display in the long range search modes. (See below).

c. Adds provisions for indicating antenna elevation to the pilot, which in differential altitude attacks under GCI will result in a greater probability of detection when the search scan pattern is centered about the target in elevation.

d. A short pulse in search is added to allow better range resolution for tracking the target (in search) through clutter after initial detection.

e. A five position range selector is added to allow the useage of longer range search for airborne targets, which increases detection capability of the system.

f. Extended lock-on range is provided to permit the radar to be locked on the target at ranges up to 20 nautical miles if target return is sufficiently strong.

g. An I.F. gain position marker and range placard is added to aid the pilot in setting the gain control for maximum detection and lock-on range.

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h. A range and azimuth scope overlay is added to aid the pilot in determining where on the scope to expect the target based on G.C.I. information. This would also be an aid in navigation from ground map and beacon information.

2. Counter-Countermeasure Capabilities - A series of counter-countermeasure additions to the MG-10 Fire Control System has been proposed to reduce the vulnerability of the system under certain ECM conditions. These additions and their effects are as follows:

a. Automatic RF tuning will tune the radar frequency away from the frequency of the jammer. This will aid in avoiding spot jammers of the slow tuning magnetron type.

b. A Home-on-Jam mode allows the radar to track a c-w barrage or spot jammer in angle. With this improvement, pursuit type attacks with the GAR-11 can be carried out with range to fire being obtained optically or by over powering the jammer.

c. Rapid range reset allows control of the range strobe unit with the hand control without changing the angular position of the radar. This speeds up re-acquisition of the target after loss of lock-on.

d. A 75 cycle scan frequency is inserted in the PRF generator resulting in a pseudo-random PRF. This provides additional capability against range-gate stealer repeaters.

e. A ferrite attnuator adds 30 db of dynamic range to the receiver, making it possible to mechanize a home-on-jam mode against high power jamming targets.

3. Installation of ARN-21 & 31. - The ARN-21 TACAN unit provides the capability of determining range and bearing. It replaces

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[redacted]

a unit which yielded bearing only. The ARN-31 provides an ILS function which when coupled with the AFCS provides automatic flight down to 50 ft. off the runway.

4. Infrared Search and Track System - The infrared system will significantly improve the overall combat effectiveness of the F-102 in such adverse combat situations as a strong ECM environment, a very low altitude attack or a malfunctioning radar system. Under such conditions at present, a successful attack is almost completely dependent upon use of the optical sight, which at night, is almost totally degraded.

The infrared system will provide precise steering information by a steering dot, which is positioned by the IR tracking head, to insure that the target will be within the field of view of the GAR-2 missile at launch. In addition, an aural tone is presented to the pilot for discrimination of a point source target from extended sources of background during the tracking phase.

The reference circle, in which the steering dot is centered, is designed to provide an indication that the IR signal strength is above the minimum threshold for GAR-2 lock-on. The Range Circle is controlled by the Intensity Rate Ranging system unit. The circle shrinks at a rate which is proportional to the rate with which the interceptor closes on the target. On coincidence with the reference circle, the interceptor is within firing range. These features will significantly enhance the effectiveness of the GAR-2 missile and can permit a completely passive attack.

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C. Weapons System Improvements

The addition of these improvements to the F-102A aircraft contribute significantly to the effectiveness of the weapon system.

1. Relative Battle Depth for Various Configurations. One measure of interceptor effectiveness is the number of interceptors which can be brought into battle against a given threat. This number is roughly proportional to the "battle-depth", i.e. the distance from the interceptor base that the first interceptor contacts the threat. In Figure 8, the battle depth versus radar coverage is shown for the F-102 configurations under study. It is to be noted that the increased fuel loadings (utilization of missile bay fuel) provide significant gains for longer radar ranges such as would be provided for inland bases or bases along the northern perimeter of the United States. At the shorter radar ranges, some gains are realized from the increased fuel and speed capabilities in utilization of a subsonic mission combined with a dash portion of the mission.

2. Zones of Effectiveness. In Figures 9, 10 and 11, the various F-102 study configurations are compared in terms of the speed and altitude capabilities of the threat against which each is to be effective. Target speed-altitude envelopes along which P_{dc} or P_{dc} X P_k are at least as great as the minimum values stated are for forward and rear-hemisphere attacks. Snap-up capability is included against targets with up to 15,000 ft. greater altitude than that of the interceptor. For rear hemisphere attacks, a larger zone for a high overall kill probability is noted for the clean configurations. Larger high altitude zones are noted for the Falcon-carrying aircraft as compared to those carrying MB-1's because of a limited P_k for the MB-1 in this area. No specific envelope is included for the GAR-11 version for head on attacks because of limited data available on P_k under these conditions. Maneuverability and lock-on ranges are expected to be comparable to those of the GAR-1D which has a limited forward hemisphere capability as indicated. However, because of its greater tolerance to large dispersions, the GAR-11 should be expected to have better capability in this area. Since the MB-1 is unguided, its kill probability is essentially independent of attack direction and therefore remains fairly high for forward hemisphere attacks.

3. Fire Control System Capabilities under Jamming Conditions.

The counter-countermeasure capabilities discussed under section B.2 above add to the attack effectiveness during the lock-on and tracking phases.

4. Missile Capabilities under Jamming Conditions. With hit-to-kill type missiles such as the GAR-1, relatively small changes in dispersion may result in substantial changes in the probability of kill, whereas large changes in dispersion may be tolerated by the GAR-11 due to its fuzing radius and lethal volume. This increased tolerance to dispersion results in enhanced missile capability in the presence of countermeasures.

The performance of the GAR-1D and GAR-11 against a variety of jammers which may be encountered tactically is indicated in Figure 12. The effect of certain countermeasures is to confuse the missile by imposing noise on the steering information, resulting in increased dispersion. Figure 12 indicates that successful operation can be attained by the GAR-11 in a countermeasures environment sufficiently severe to destroy the capability of the GAR-1D.

The MB-1 missile, being unguided and time fuzed, is not vulnerable to any countermeasure after launch. However, any countermeasure directed at the interceptor fire control system and designed to deny or degrade range information prior to missile launch will increase the dispersion of the MB-1 by an amount sufficient to seriously degrade the kill probability. Without a back-up armament for use under these countermeasure conditions, the MB-1 is considered inferior to the GAR-11 as a primary weapon. More specifically, the GAR-11 can be fired on home-on-jam mode, after obtaining rough range by optical, infrared or cross-over methods, under conditions where the ranging requirements of the MB-1 could not be met.

TABLE I

VERSION WEIGHTS

A/C	CONFIGURATION	Weight Empty	Take-Off Weight	Take-Off Weight Ext. Tanks	Fuel Wt.	Fuel Wt. with Ext. Tanks	Armament Weight
A.	Tactical F-102A	19,190	28,150	31,276	7,053	9,848	1,194
B.	Tactical F-102A plus improvements	18,835	27,794	30,920	7,053	9,848	1,194
C.	B with 694 gal. missile bay fuel & (2) Ext. MB-1	18,580	32,434	-	11,394	-	1,658
D.	B with 454 gal. missile bay fuel & (1) Int. MB-1	18,584	29,959	33,085	9,834	12,629	829
E.	B with 454 gal. missile bay fuel & (2) GAR-11	18,637	29,690	32,816	9,834	12,629	432
F.	B with 694 gal. missile bay fuel & Ext. tanks (No armament)	18,549	-	33,781	-	14,189	0

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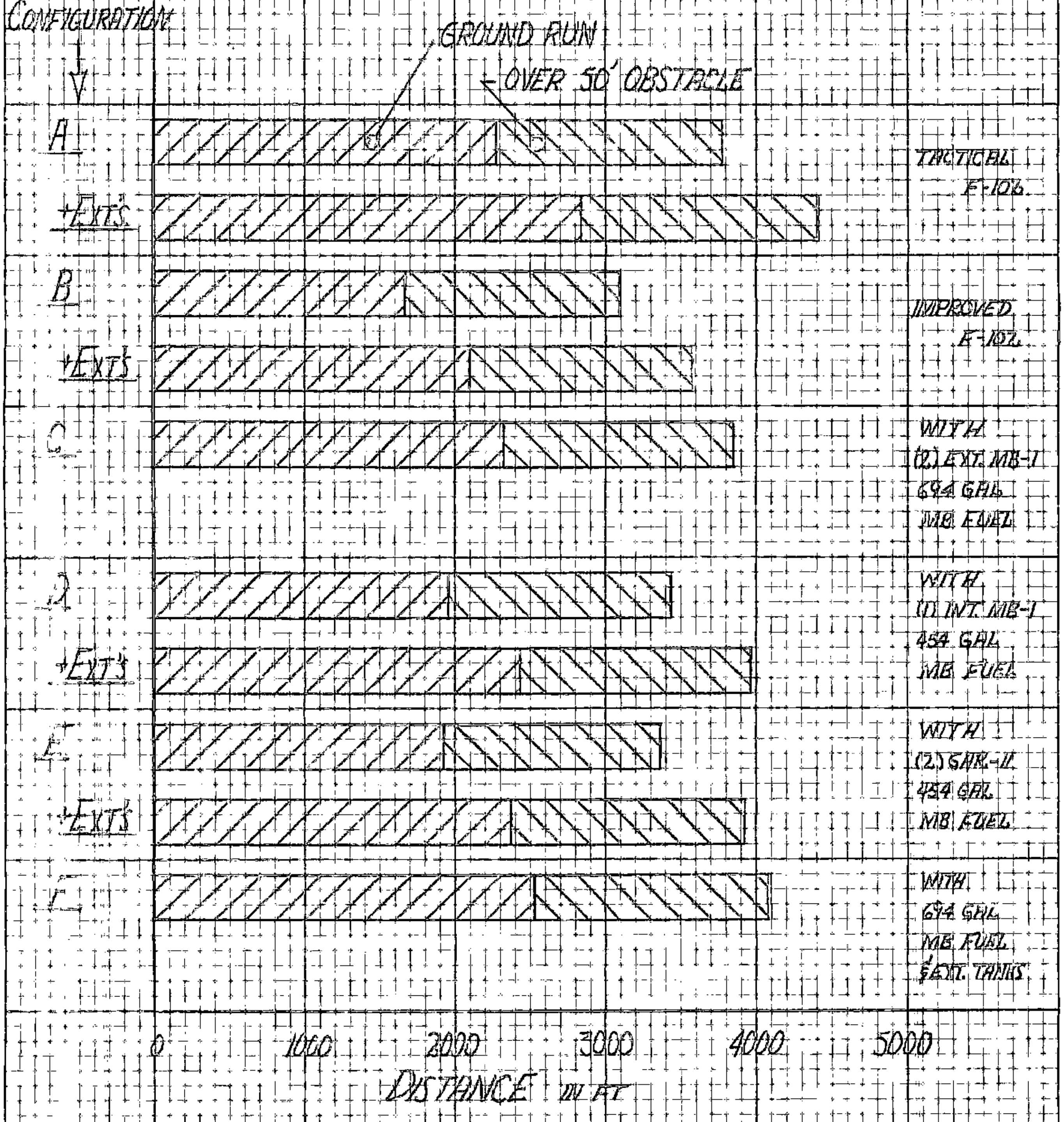
TAKE OFF DISTANCE COMPARISON-F102

MIL. + A/B THRUST

FIG. 1



[redacted]

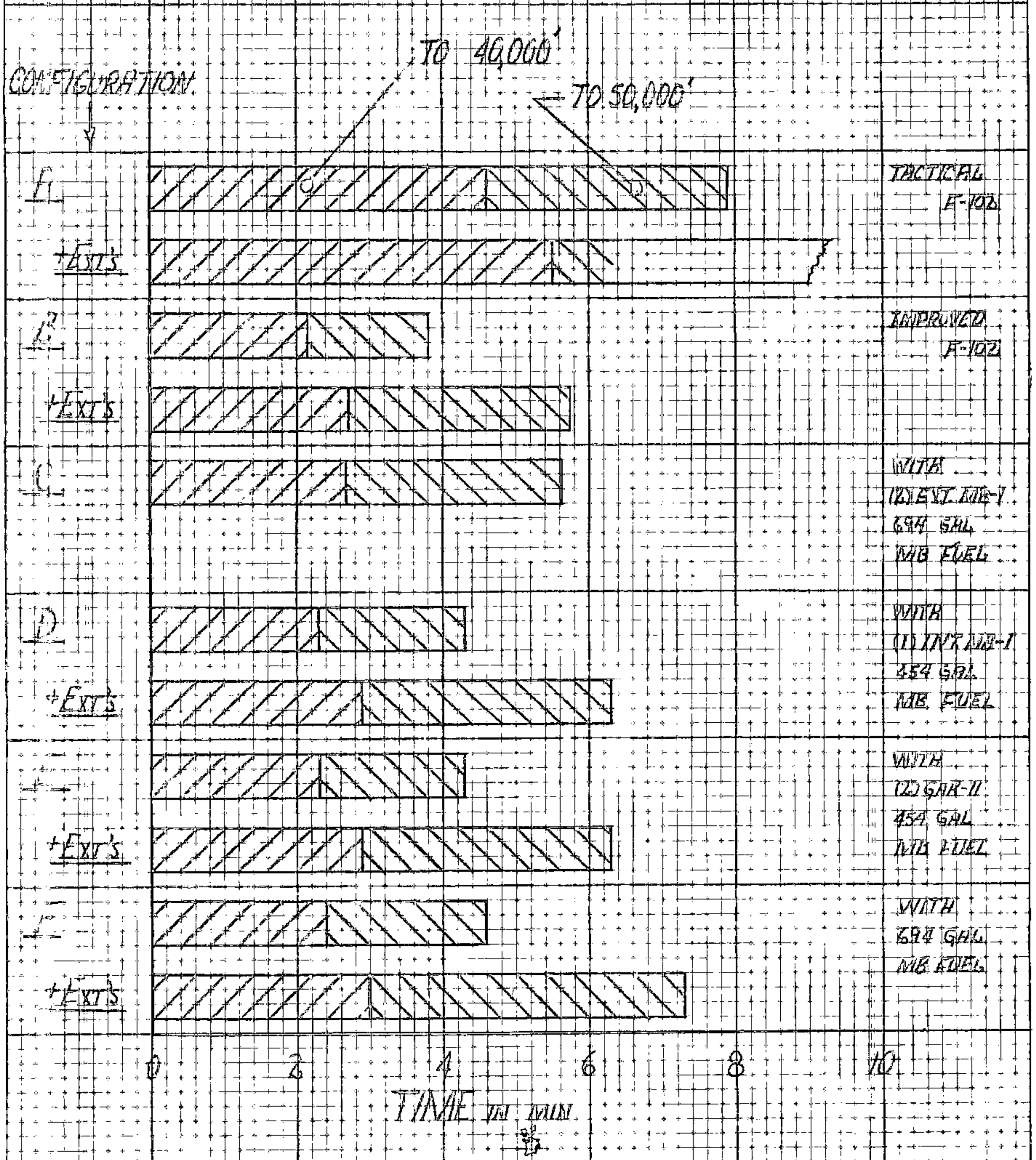
10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO.

[redacted]

TIME TO CLIMB TO ALTITUDE

MIL + A/B THRUST

FIG 2



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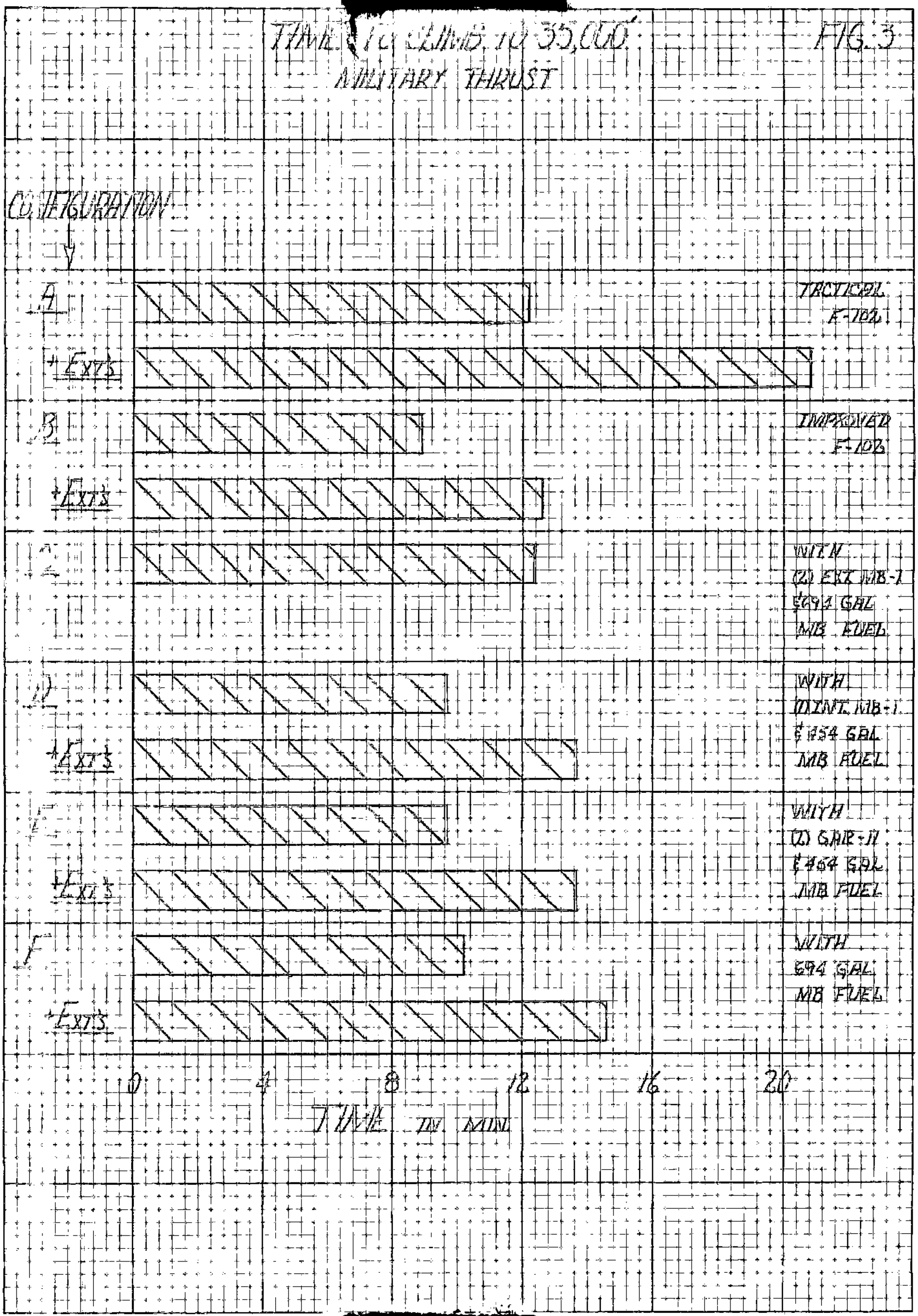
10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO

[redacted]



TIME TO CLIMB TO 35,000'

MILITARY THRUST

FIG. 3

[redacted]

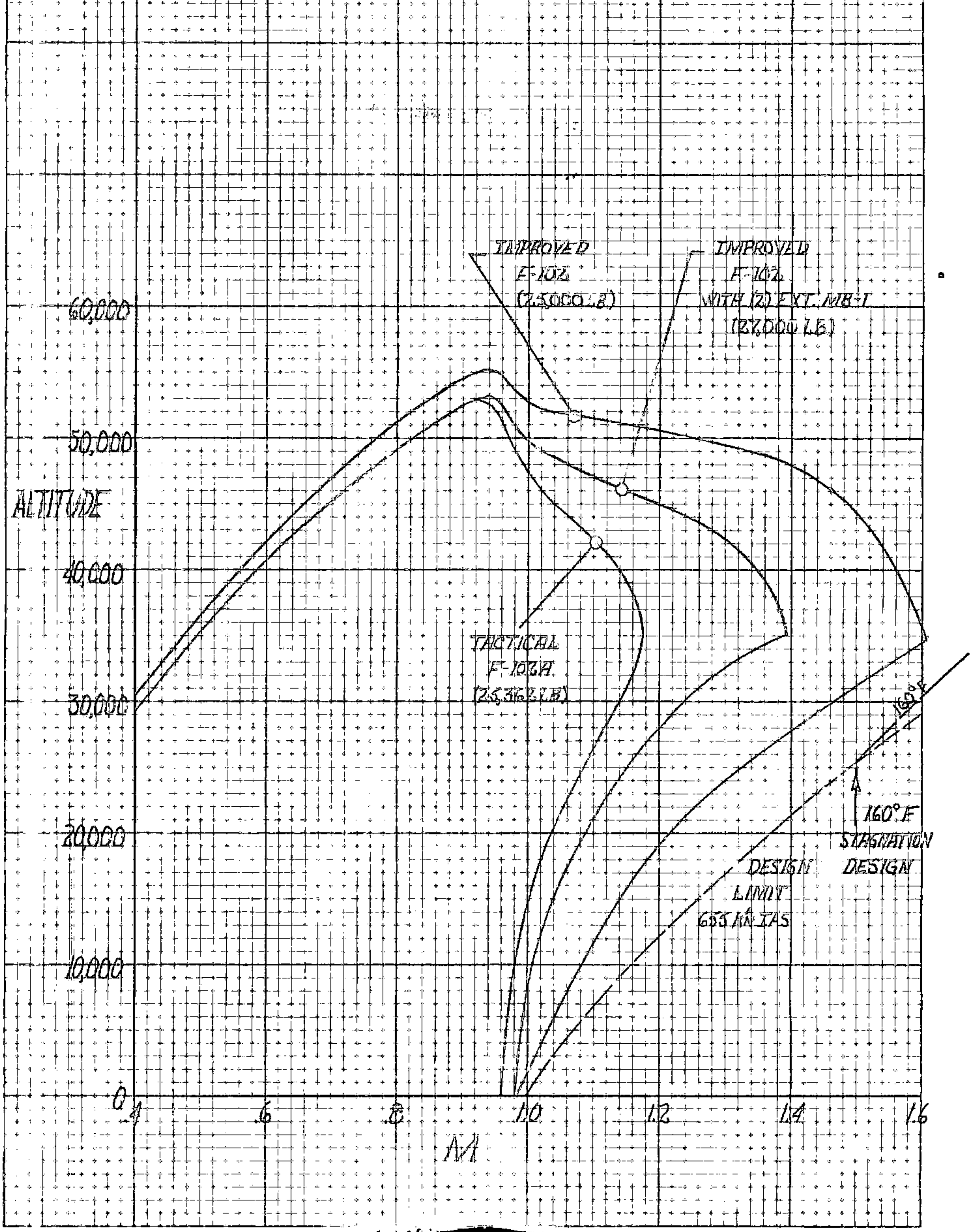
10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO.

[redacted]

SPEED-ALTITUDE ENVELOPE

MIL + A/B THRUST

FIG. 4



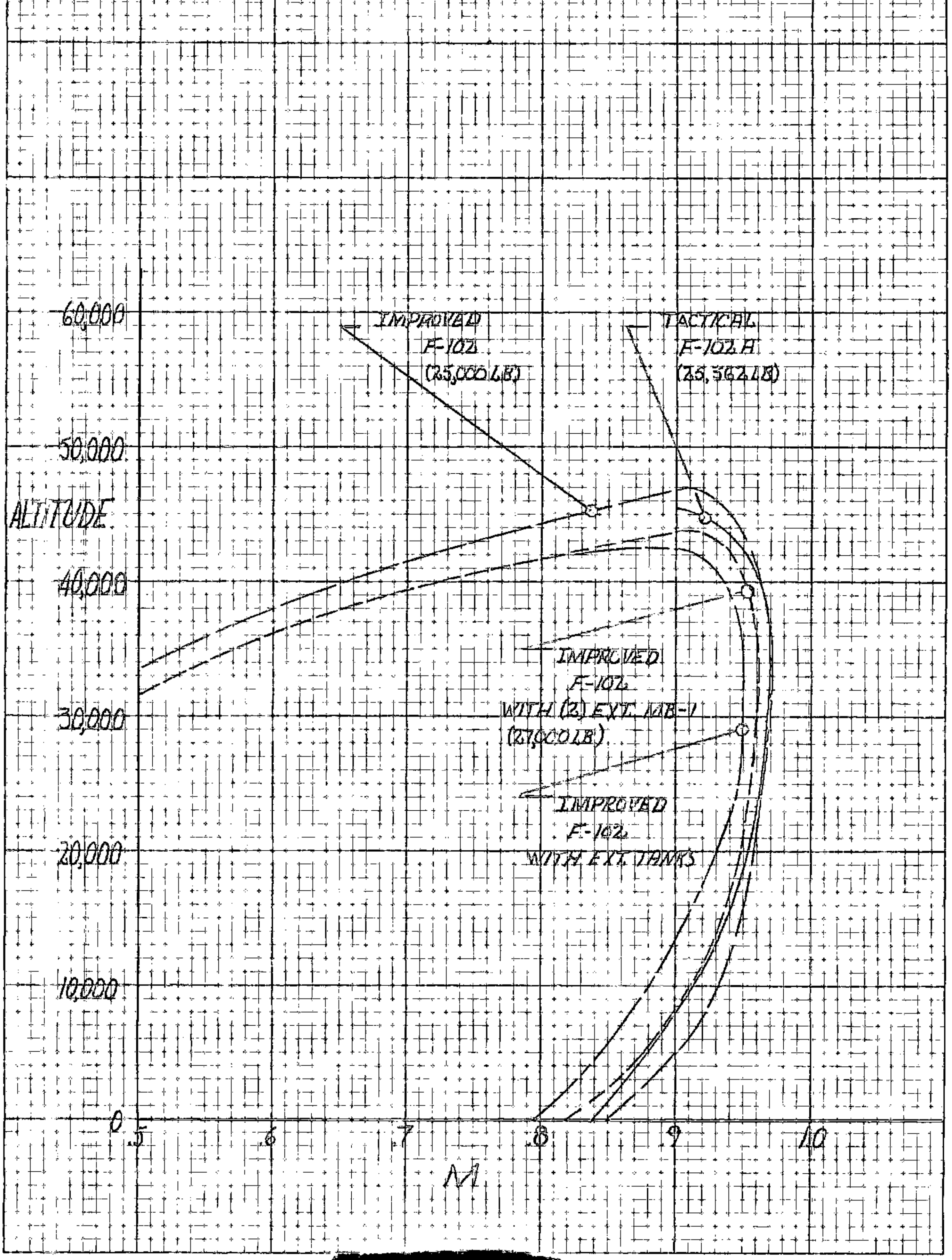
[redacted]

K+E 10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO. MADE IN U.S.A.

[redacted]

SPEED-ALTITUDE ENVELOPE FIG.5

MILITARY THRUST



[redacted]

10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO.

[redacted]

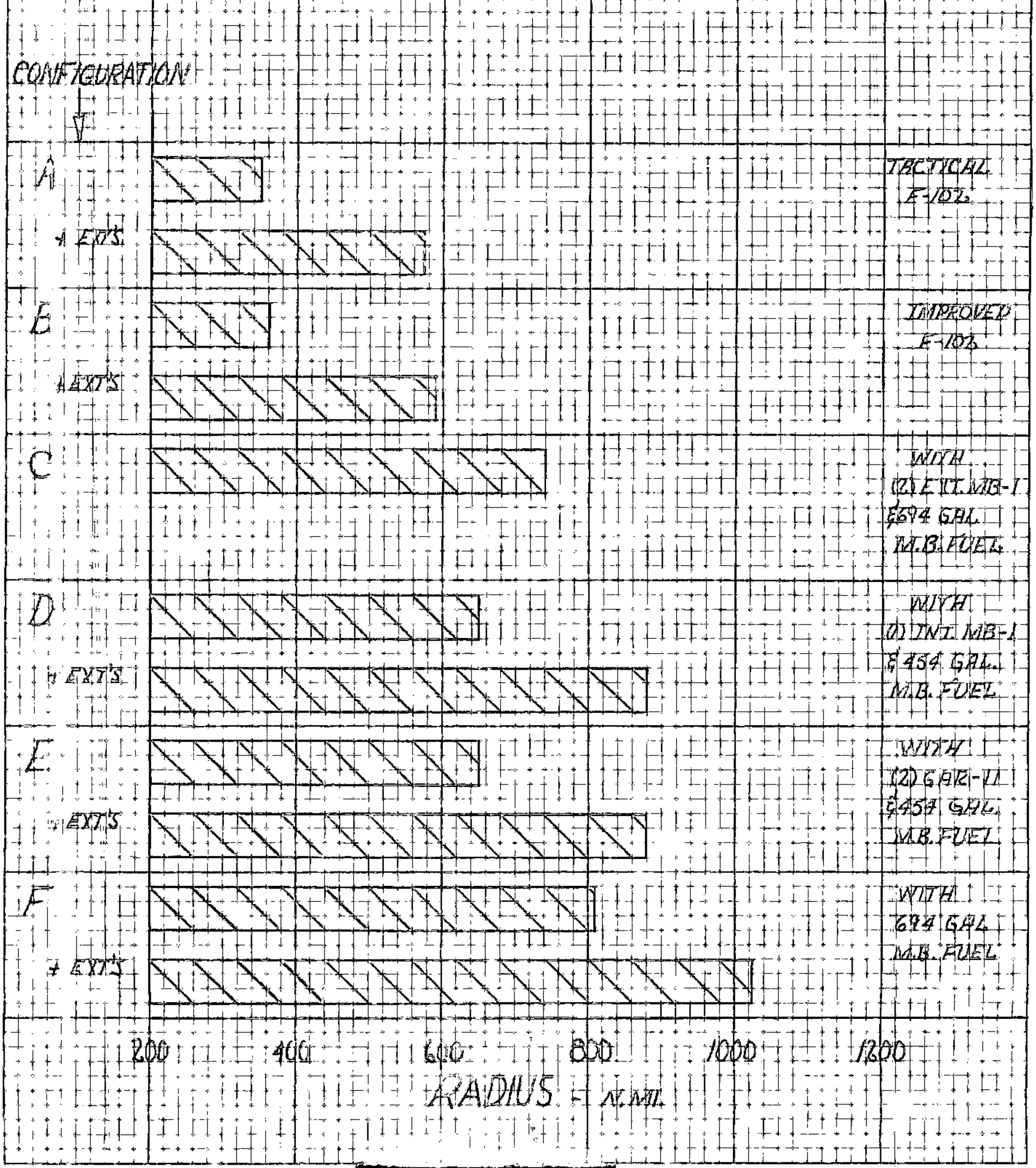
SUBSONIC MISSION RADIUS

FIG. 6

MISSION OUTLINE

1. T.O. ALLOWANCE 940 LB
2. MIL. POWER CLIMB TO 40,000 FT
3. M=.9 CRUISE OUT AT 40,000 FT
4. MAX. POWER CLIMB TO COMBAT CEILING.
5. 2 MIN MAX POWER COMBAT.
6. M=.9 RETURN AT 40,000 FT
7. RESERVE FUEL, 5% INITIAL + 20 MIN. MAX. S.L. ENDURANCE.

NOTE: DROP EXTERNAL TANKS WHEN EMPTY (IF CARRIED).



[redacted]

K&E 10 X 10 TO THE INCH 359-5 KEUFFEL & ESSER CO.

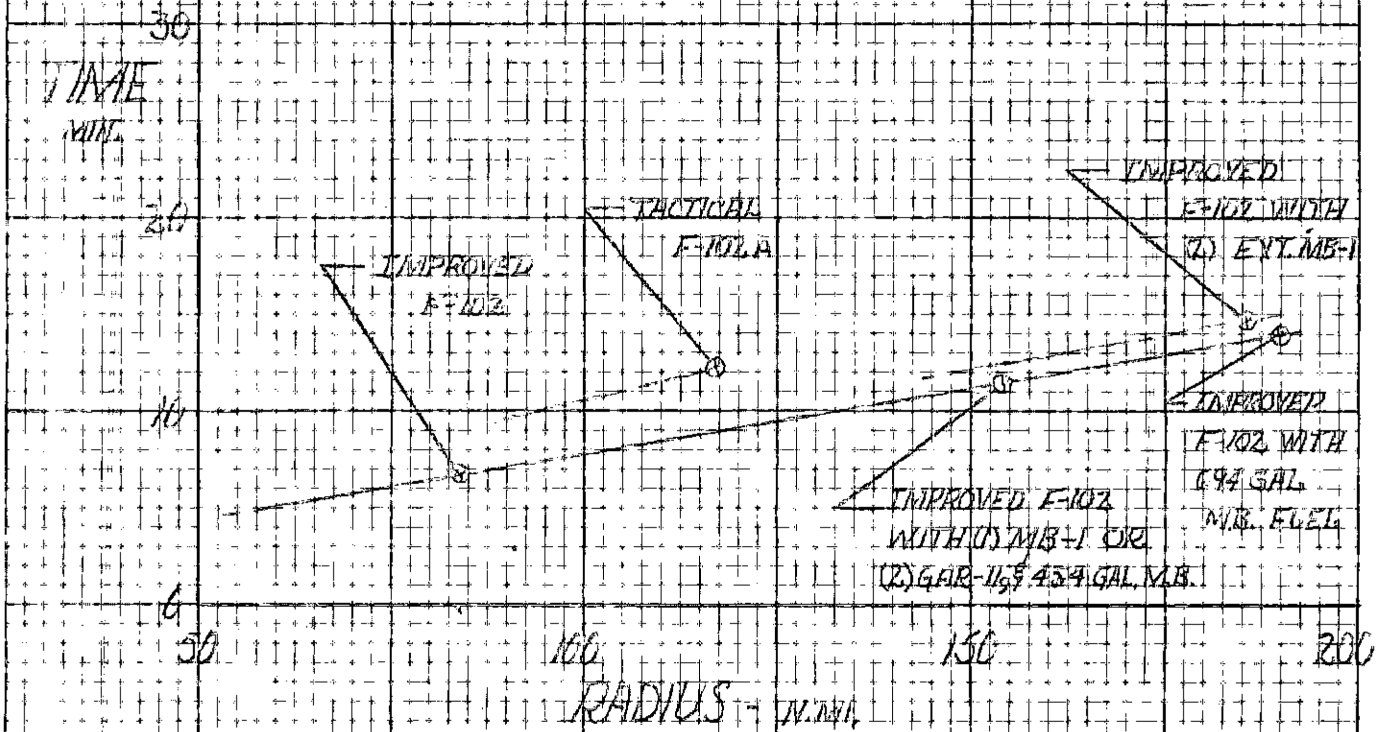
[redacted]

DASH MISSION RADIUS & TIME TO INTERCEPT

FIG. 7

MISSION OUTLINE

1. T.O. ALLOWANCE 940 LB.
2. MAX. POWER CLIMB TO 35,000 FT.
3. ACC. TO V_{max} AT MAX POWER.
4. MAX. SPEED CRUISE OUT, 35,000 FT.
5. 2 MIN. MAX. POWER COMBAT
6. MIL. POWER CLIMB TO 40,000 FT.
7. M=.9 RETURN AT 40,000 FT.
8. 5% INITIAL + 20 MIN. MAX END. RESERVE.



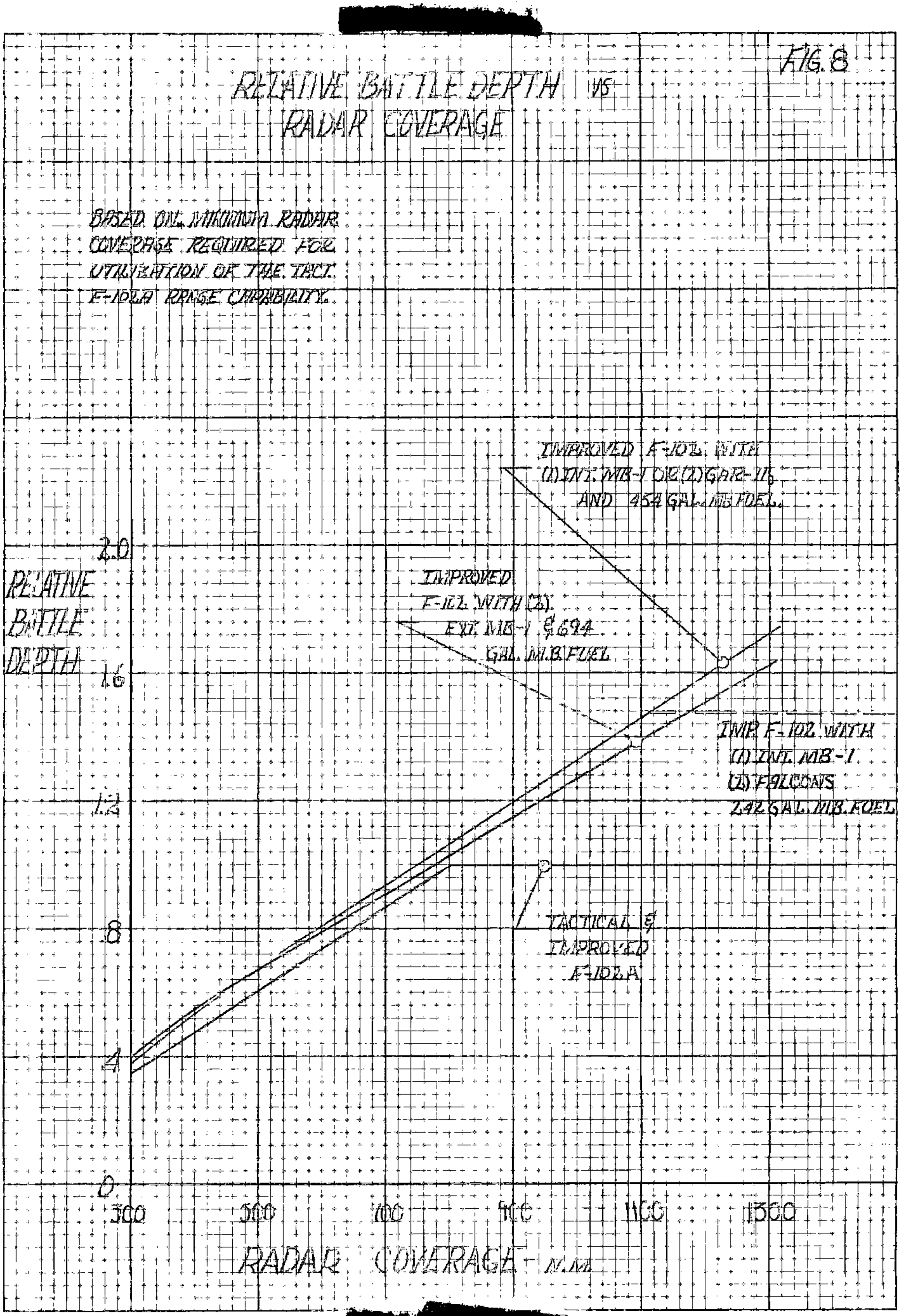
[redacted]

K+E 10 X 10 TO THE INCH 359-5 KEUFFEL & ESSER CO.

[redacted]

FIG. 8

RELATIVE BATTLE DEPTH VS RADAR COVERAGE

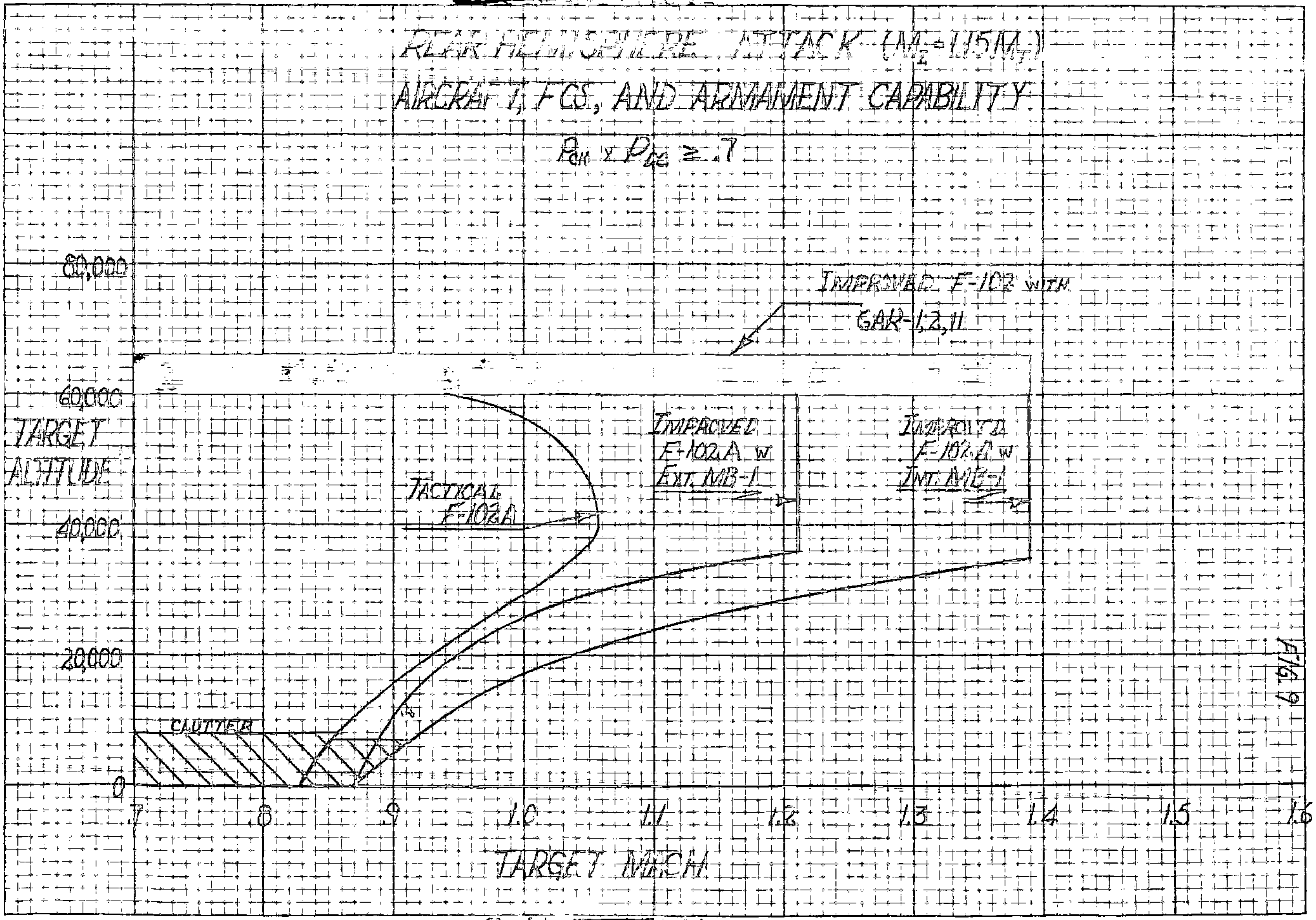


BASED ON MINIMUM RADAR COVERAGE REQUIRED FOR UTILIZATION OF THE TACT. F-102A RANGE CAPABILITY.

[redacted]

10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO.

10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO.



REAR HEMISPHERE ATTACK ($M_I = 1.15 M_T$)

AIRCRAFT I, FCS, AND ARMAMENT CAPABILITY

$P_{CH} \times P_{GG} \geq .7$

FIG. 9

[redacted]

K+E 10 X 10 TO THE INCH 359-5 KEUFFEL & ESSER CO.

AIRCRAFT & FCS CAPABILITY

$(P_{k} \geq .85)$

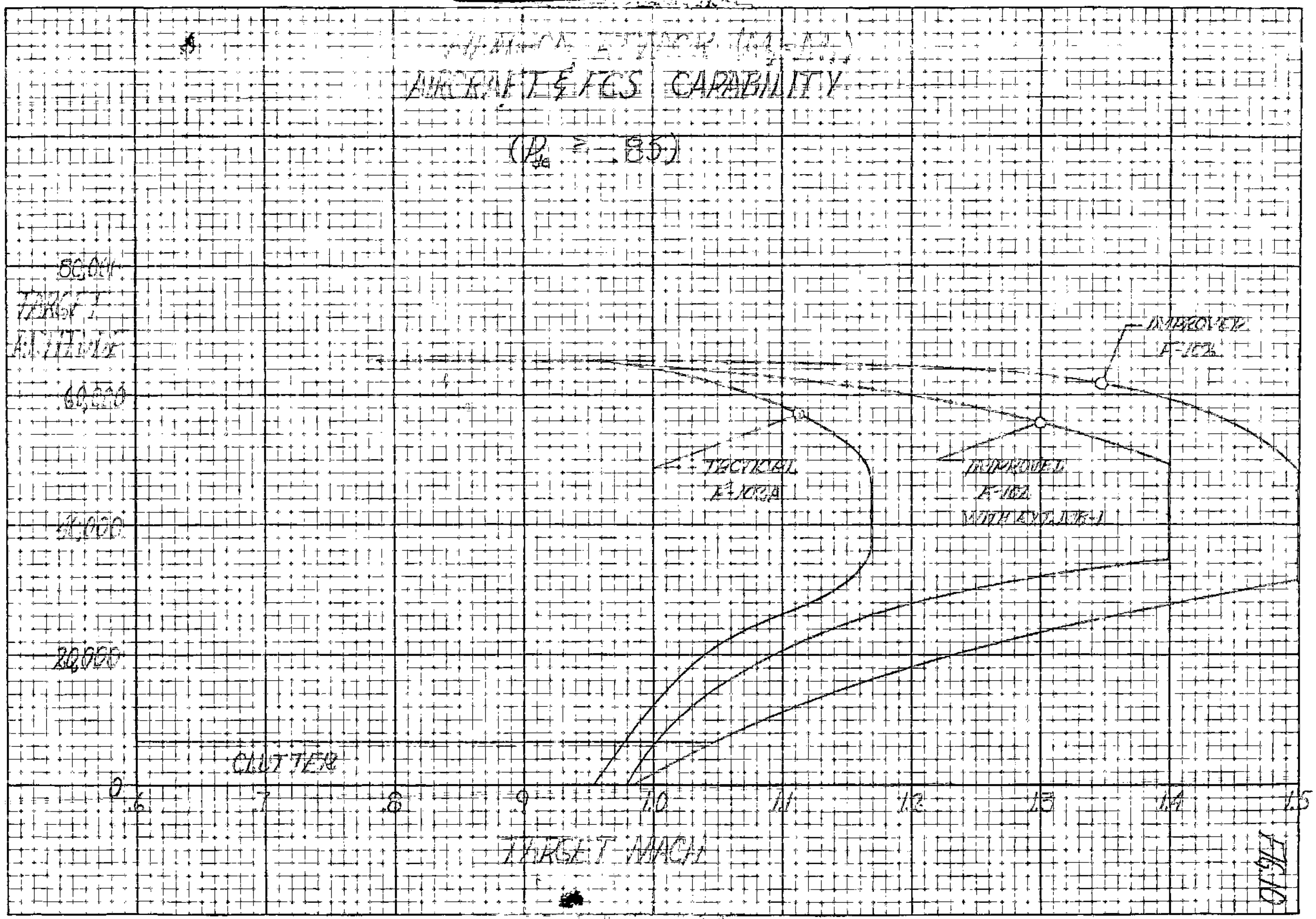


FIG. 10

[redacted]

K&E 10 X 10 TO THE INCH 359-5
KEUFFEL & ESSER CO.

[redacted]

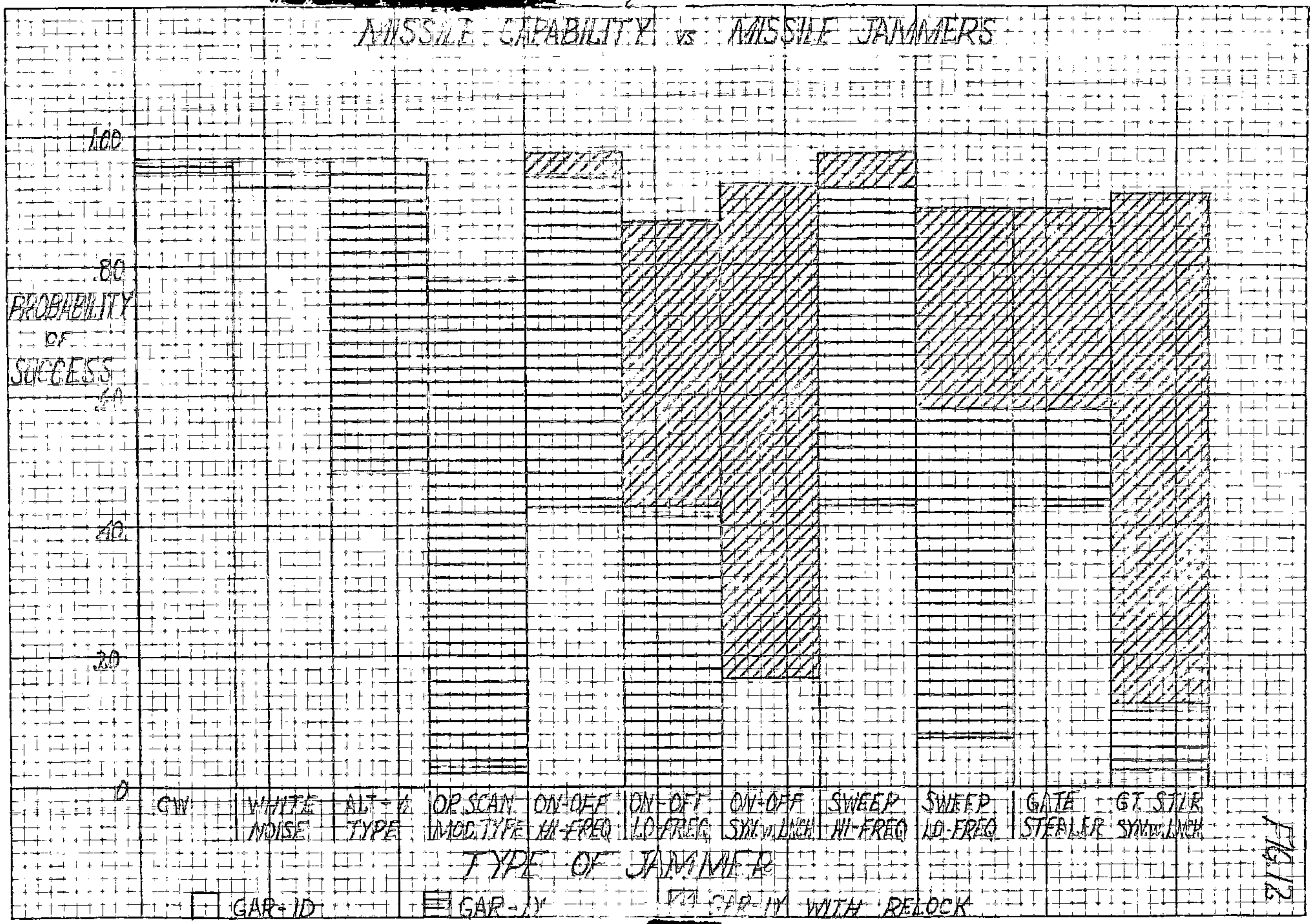


FIG.12

[redacted]

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APPENDIX A

The c.g. envelope for the configurations of the improved F-102 are given in Figures A-1 through A-5. All curves include the effect of the systems improvements and are based on using the steel version of the Pratt & Whitney JT-3C-22 engine.

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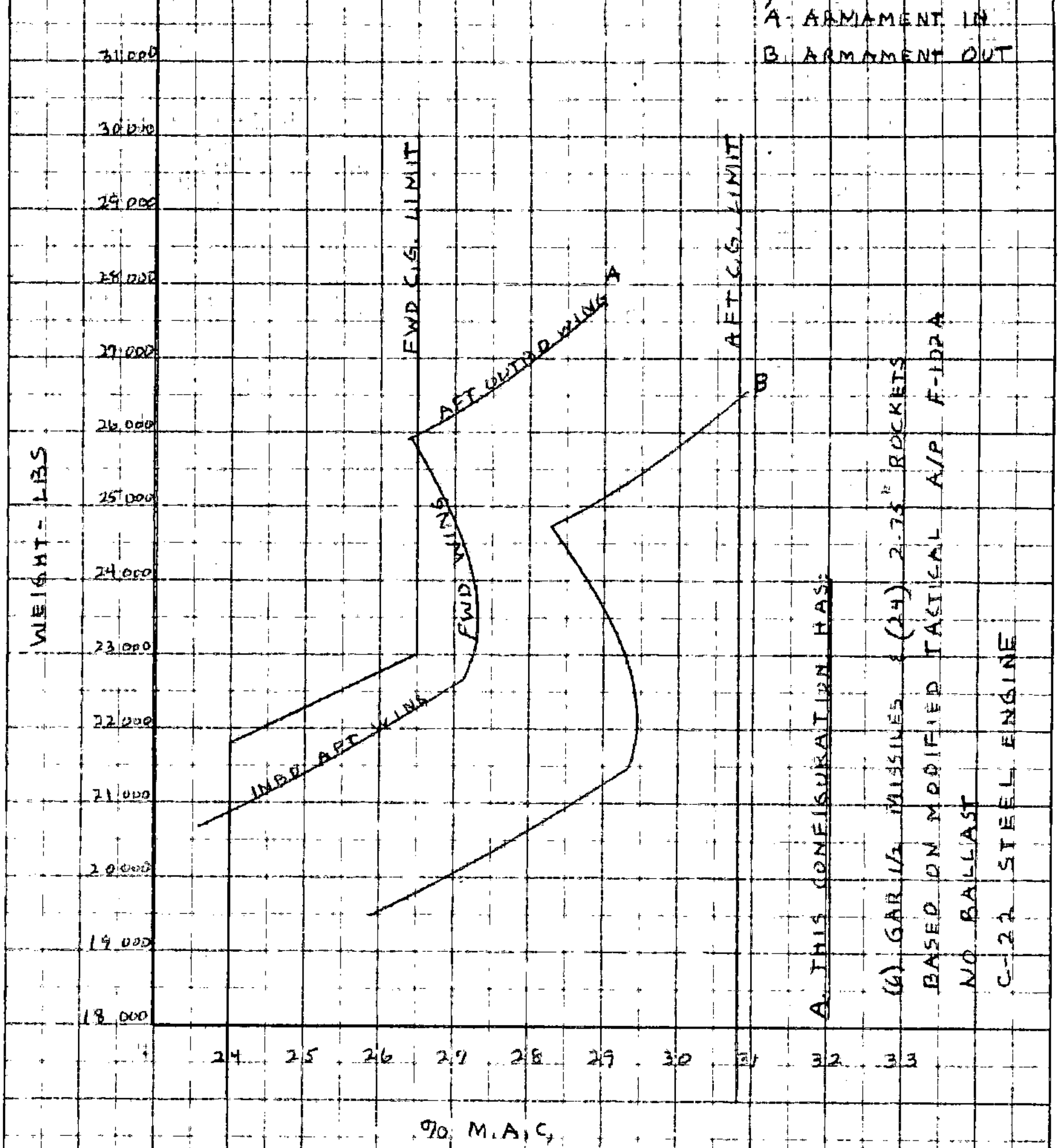
IMPROVED F-102B FIG. 8-1



A. THIS CONFIGURATION HAS:

(4) GAR 1A MISSILES & (24) 2.75" ROCKETS

BASED ON MODIFIED TACTICAL A/P F-102A

NO BALLAST

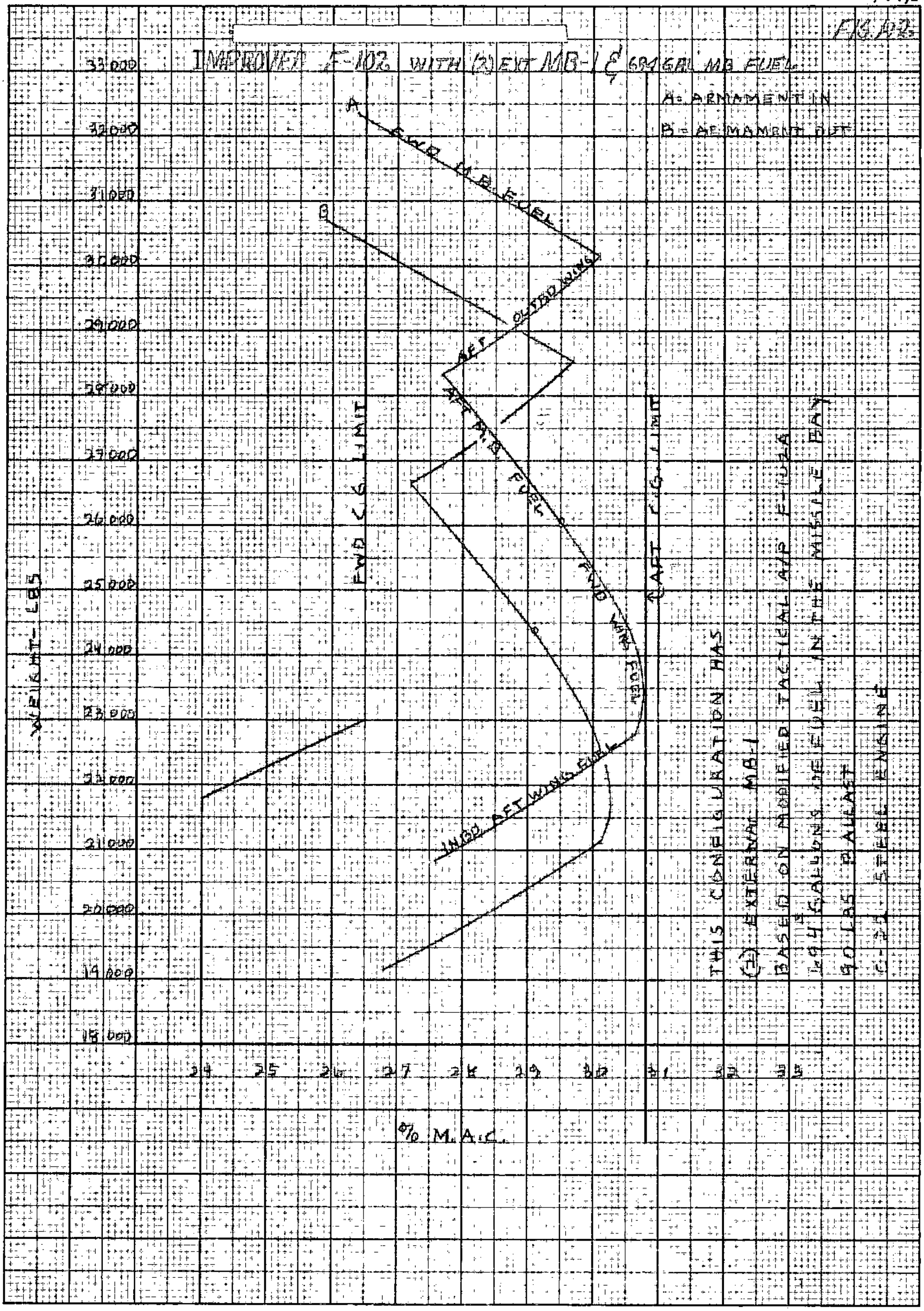
C-22 STEEL ENGINE

K&E 10 X 10 TO THE ½ INCH 359T-11G KEUFFEL & ESSER CO. ALBANENE

1/25/58

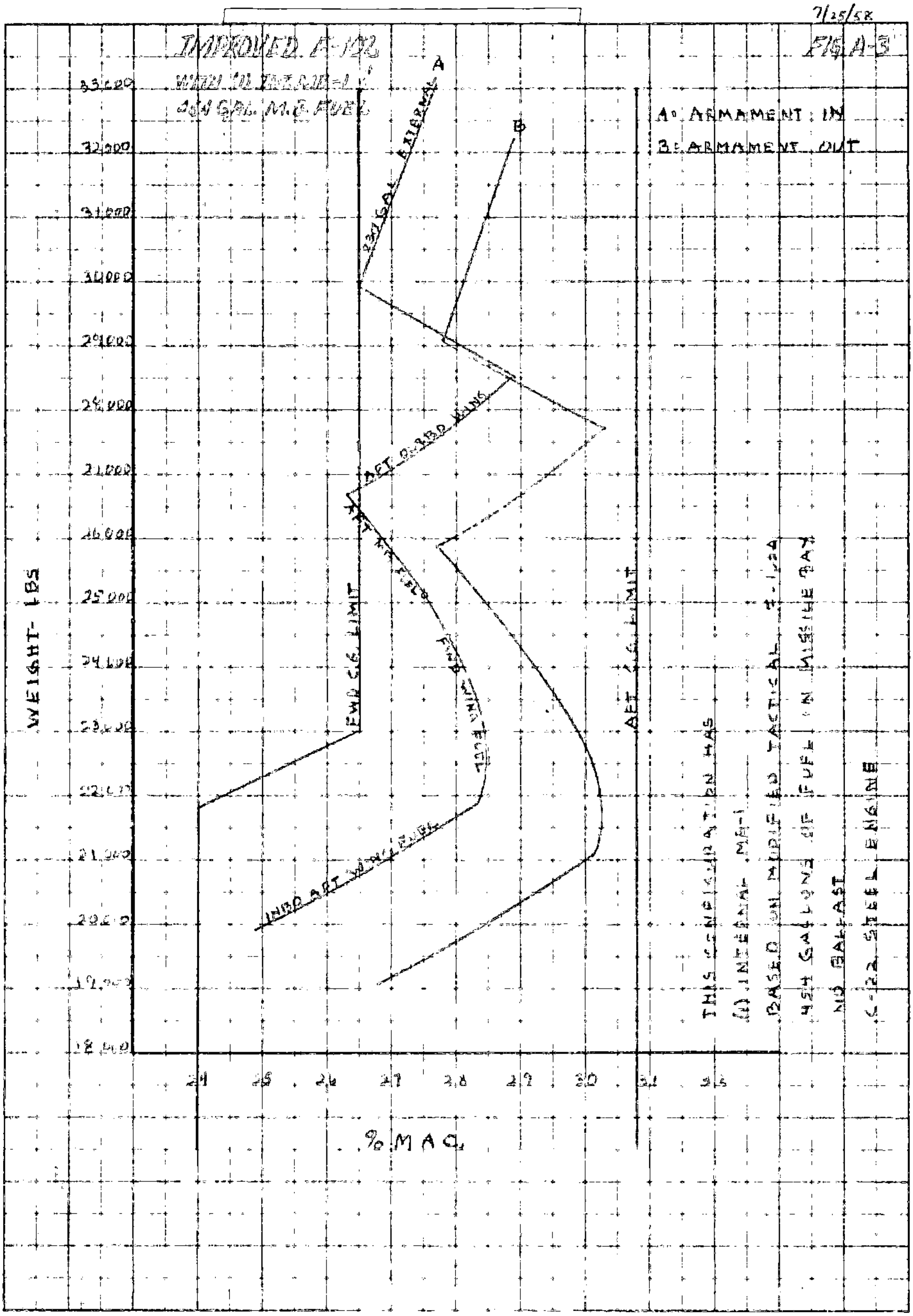
FIG. 102

IMPROVED F-102 WITH (2) EXT MB-1 & 694 GAL MB FUEL

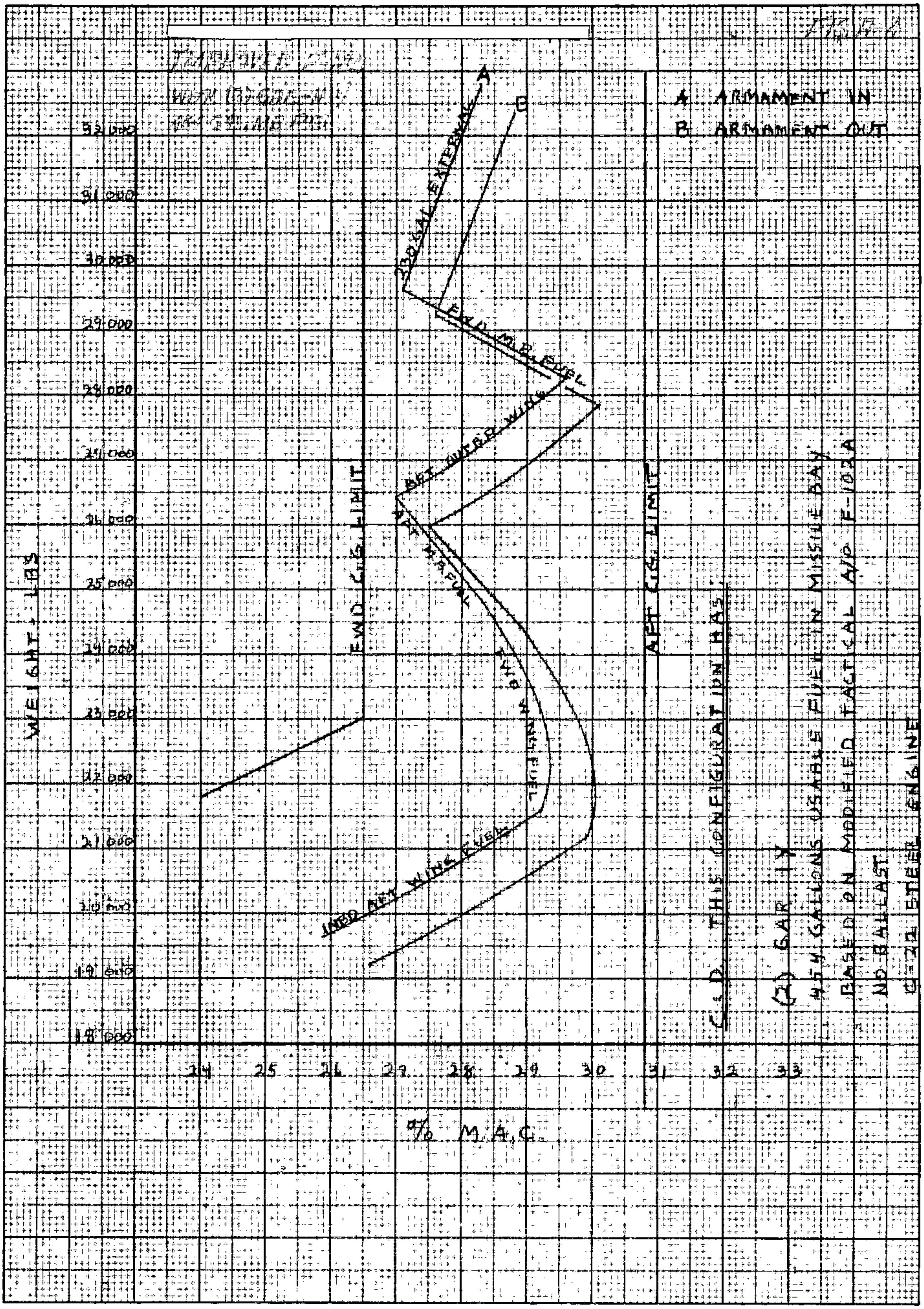


K+E 10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO. MADE IN U.S.A.
ALBANENE ®

7/25/58



K&E 10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO. MADE IN U.S.A. ALBANENE



C.G.D. THIS CONFIGURATION HAS:

(2) GAR 1X

454 GALLONS USABLE FUEL IN MISSILE BAY

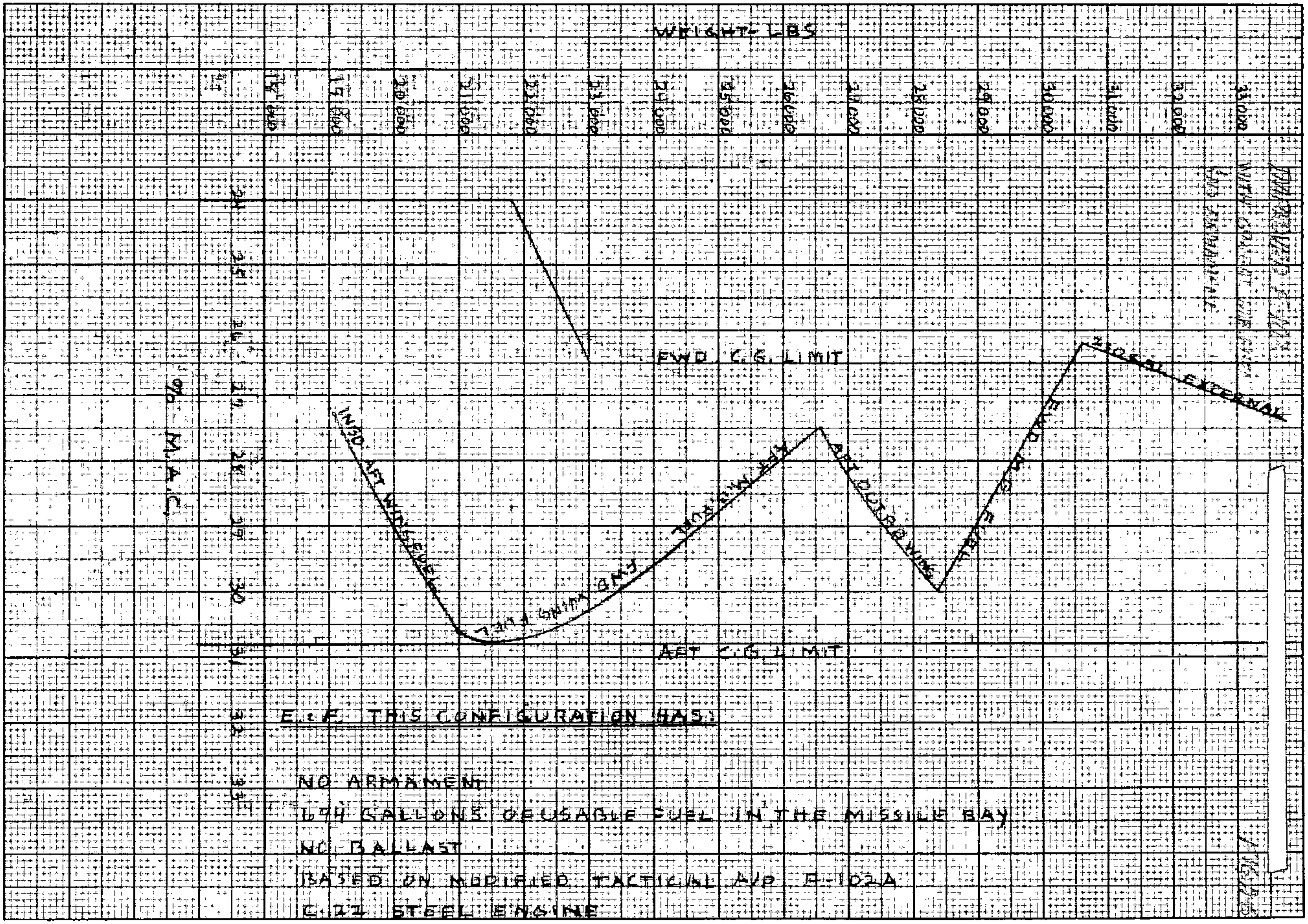
BASED ON MODIFIED TACT CAL A/C F-102A

NO BALLAST

J-57 STEEL ENGINE

K+E 10 X 10 TO THE ½ INCH 359T-11G KEUFFEL & ESSER CO. MADE IN U.S.A. ALBANENE ®

7/10/58 29



E.F. THIS CONFIGURATION HAS:

NO ARMAMENT
694 GALLONS OF USABLE FUEL IN THE MISSILE BAY
NO BALLAST
BASED ON MODIFIED TACTICAL A/P F-102A
C.22 STEEL ENGINE

[redacted]

APPENDIX B

Installation Task.

[redacted]

The task of installation of these improvements is outlined in the following documents:

1. CPO 58-173 "S/A-MG-10/MG-10T Fire Control System- Range Extension and Improved Target Discrimination".
2. CPO 58-95 "F-102 Replacement of Wing Fences with Leading Edge slots".
3. CPO 58-2-6 - "S/A-Addition of Passive Detection Capability to F/TF-102A Aircraft-Single Seeker Head".
4. ECP - 1220-6225 - "ARN-21 and ARN-31 Installation-Change in Production Effectivity".
5. CPO 58-41 - "GAR-1Y/CCM Modes Incorporation - Retrofit Program".
6. Missile Bay Fuel Tank Installation is covered in CPO 58-2-7 "Alaskan Command F-102A Extended Range Configuration".

The JT-3C-22 engine installation task is as follows:

a. Revise engine cooling system to new requirements.

b. Revise oil system using state-of-the-art components as needed for the JT-3C-22 requirements.

c. Revise fuel system, low pressure warning for engine capability.

d. Revise air intake system to provide for additional engine cooling air and revised oil cooler.

e. Revise fire detector system.

f. Revise ice detector system.

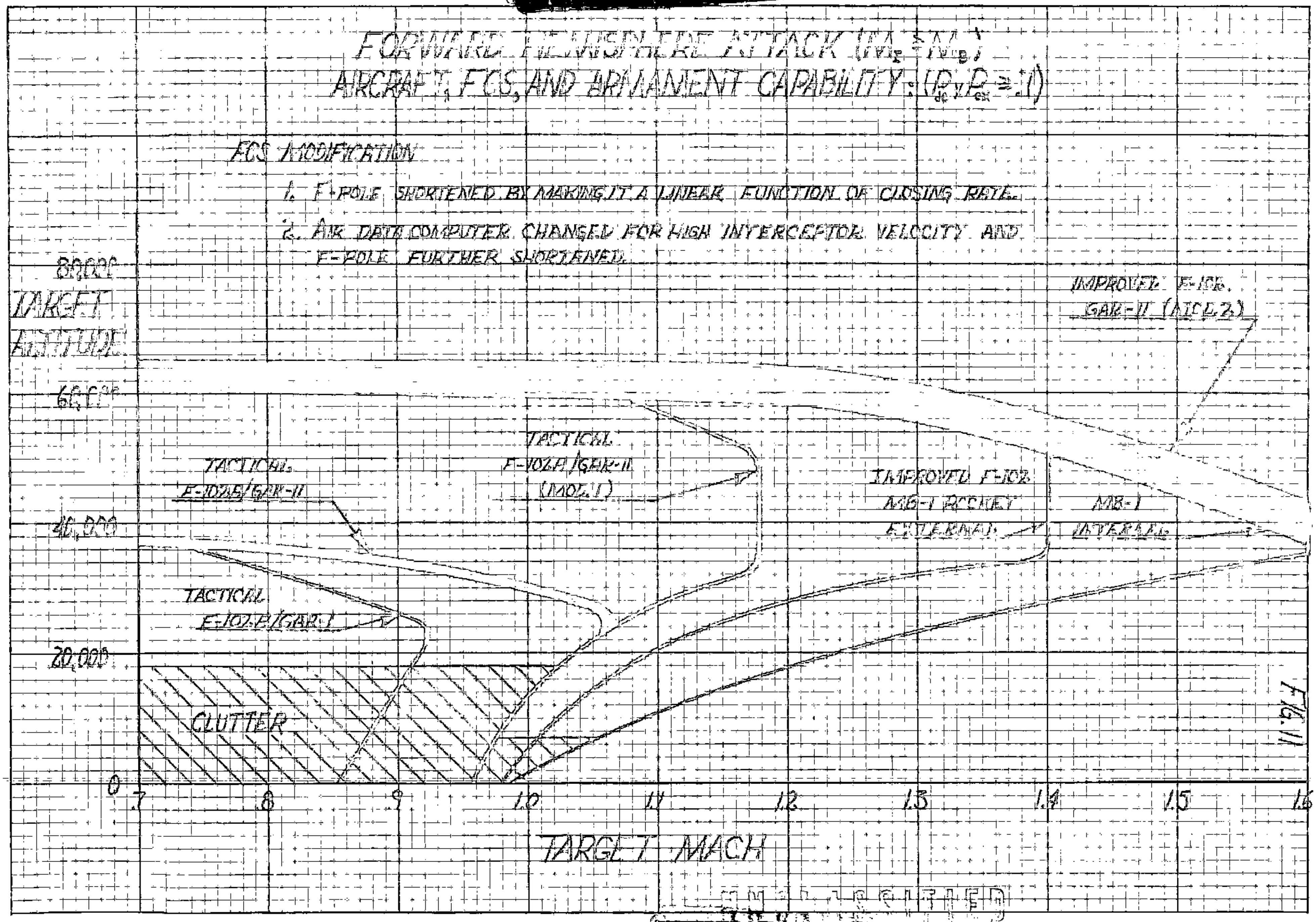
g. Revise tubing routings as required by new engine.

h. Redesign engine removal system.

i. Revise engine instrument operational range limits.

j. Redesign upper engine mount beam due to increased thrust.

[redacted]



FORWARD HEMISPHERE ATTACK ($M_I \geq M_B$)
AIRCRAFT, FCS, AND ARMAMENT CAPABILITY: ($P_{dc} \times P_{ex} \geq .1$)

FCS MODIFICATION

1. F-POLE SHORTENED BY MAKING IT A LINEAR FUNCTION OF CLOSING RATE.
2. AIR DATA COMPUTER CHANGED FOR HIGH INTERCEPTOR VELOCITY AND F-POLE FURTHER SHORTENED.

FIG. 11

[redacted]

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APPENDIX C

Mixed Loading Configuration

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Data has also been compiled on a configuration having a mixed armament loading along with some added fuel. This configuration consists of:

Improved F-102 with -

(1) MB-1 in the center missile bay (forward position),
(2) Talcon type missiles in the forward outer bays,
and 242 gallons of fuel in the aft outer bays.

The data added to the charts for this version is as follows:

	Normal Fuel	With External Tanks
Take off distance		
Ground Roll	1850 ft	2310 ft
Over 50' Obstacle	3310 ft	3830 ft
Time to Climb-mil + A/B		
To 40,000'	2.28 min.	2.85 min.
To 50,000'	4.17 min.	6.10 min.
Time to Climb - mil power		
35,000'	9.5 min.	13.3 min.
Subsonic Mission Radius	518 n.mi.	750 n.mi.
Dash Mission		
Radius	121 n.mi.	-
Time	9.2 min.	-

The weights for this configuration are:

Empty	18,847	Fuel Wt	8546
T.O. WT	29,191	Fuel W Ext	11,341
T.O. W Ext.	32,317	Arm. Wt	1085

The c.g. envelope is given in Figure C-1.

An alternate version consists of loading (2) GAR-11 missiles in place of the MB-1 rocket.

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K+E 10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO. MADE IN U.S.A.
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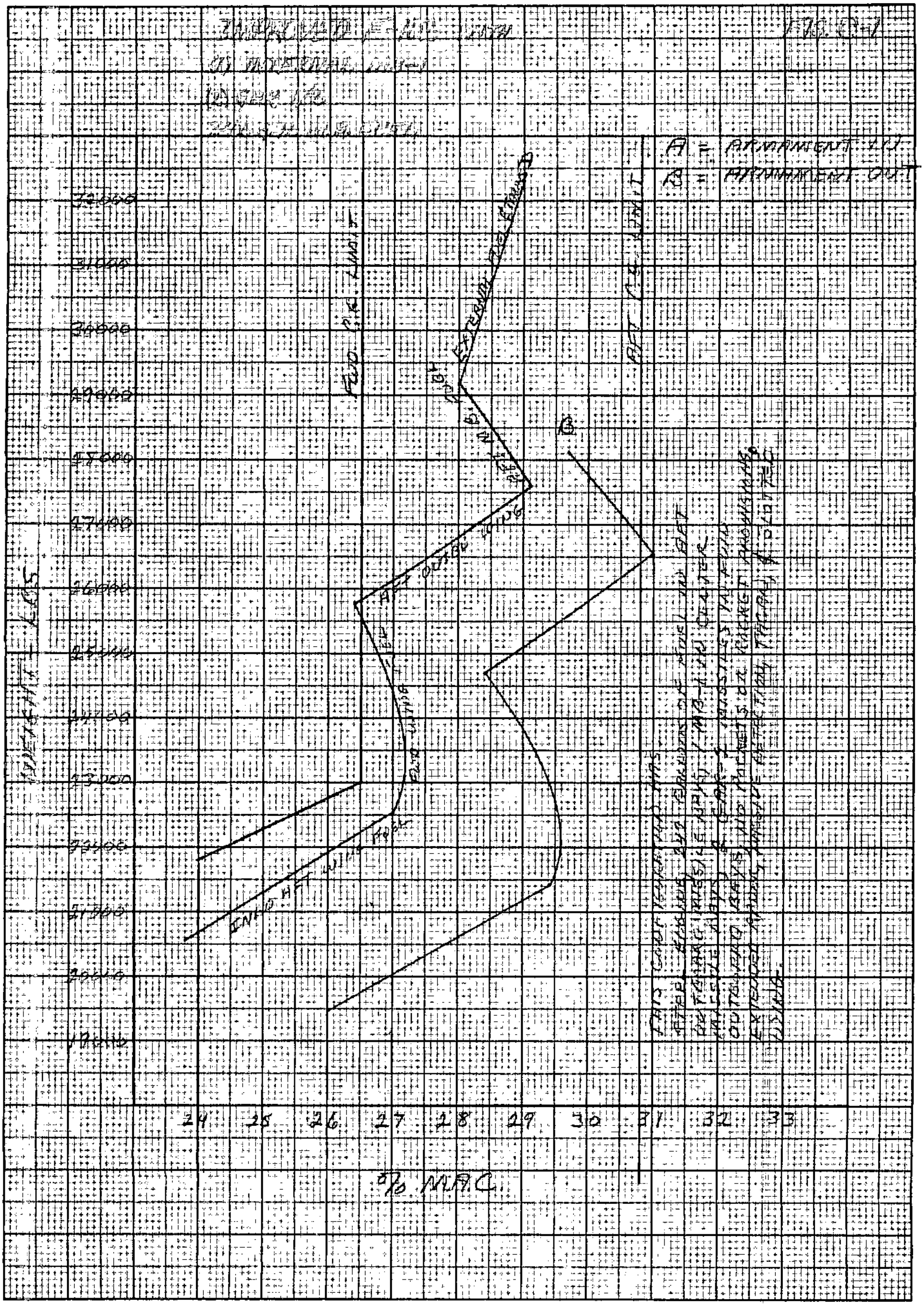

THIS CONFIGURATION HAS STEEL ENGINE, 242 GALLONS OF FUEL IN AFT RETRACTABLE MISSILE BAYS, 1 MB-1 IN CENTER MISSILE BAYS, 2 GAR-1 MISSILES IN FWD OUTBOARD BAYS, NO POCKETS OR ROCKET ARMAMENT, EXTENDED RADOME, MISSILE DETECTION, THERM, & SLOTTED WING.

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ZP-8-030

[redacted] UNCLASSIFIED

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-030
DATE 16 Nov. 1955
MODEL 8

[redacted]

TITLE

PRELIMINARY DESIGN STUDY
OF
ADVANCED F-102 INTERCEPTOR

ZP - 8 - 030

GR. 4
DOWNGRADED AT 3 YEAR INTERVALS; DECLASSIFIED AFTER 12 YEARS.
DOD DIR 5200.10

[redacted]

PREPARED BY Harold O. Wendt
H. E. Dickard
L. B. Christensen

GROUP Preliminary Design
REFERENCE
APPROVED BY A. R. Lambert
A. R. Lambert

NO. OF PAGES 22
NO. OF DIAGRAMS

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REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED

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RM 1812 A-4

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INTRODUCTION

The purpose of this study is to investigate the next step in the evolution of the F-102 type of airplane. This step is one which has as a target a prototype airplane in about three years.

This prototype would be a component of a weapon system that would be effective against a Mach 2 mass raid at 60,000 feet as a minimum and take maximum advantage of the projected contiguous coverage.

This report describes this prototype airplane with three possible engine installations.

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SUMMARY

A preliminary design study of an advanced F-102 airplane has developed configurations with greatly improved performance characteristics compared with those of the F-102B.

Four similar configurations with three different engines are described in this report.

A target date of 1959 appears feasible for a prototype of one of the configurations.

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CONFIGURATION STUDIES

Variables Studied:

Studies were made in the Preliminary Design Group of various changes to the F-102B configuration that would result in improved airplane or system performance. Among these were the following:

1. Canard control system
2. 40-inch radar antenna
3. Increased radar power
4. Larger thrust engines
5. Biconvex airfoil section
6. Variable camber leading edge
7. Precompressor cooling with water
8. Flush canopy
9. Guided nuclear warhead rocket
10. Pilot escape capsule
11. Increased directional stability

Aerodynamic studies were also made to determine the effects of these changes on airplane performance. After analyzing the individual effects, four combinations of these changes were chosen for investigation.

Configuration with JT4B-21 Engine:

The first combination to be studied is shown in Figures 1 and 2. This configuration included the following changes from the F-102B:

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1. Canard control system
2. 40-inch radar antenna
3. Pratt & Whitney JT4B-21 engine
4. Precompressor cooling with water
5. Biconvex airfoil section (3.85% thick)
6. Variable camber leading edge
7. Pilot escape capsule
8. Increased power in radar
9. Guided nuclear warhead rockets
10. Larger external fuel tanks (two 515-gals.)

The canard control system consists of a freely-floating, 45° delta surface located with the hinge line at Station 90. The canard is located low on the fuselage to keep the wake of this surface below the wing and away from the engine inlet. Actuation of the canard surface may be by several methods, but at present the best method appears to be by an irreversible trailing edge tab on the canard surface controlled by the pilot's stick. Damping of the main surface may be provided if analysis or tests show that it is necessary to insure against flutter. Longitudinal control feel will be provided similar to the way it is now obtained on the F-102B.

All trimming and control will be done by the canard and the elevons will become flaperons; that is, they will be used as ailerons and landing flaps with deflections of five degrees when used as flaps. The use of this surface makes unnecessary the use of the fuel transfer system as used on the F-102B. Further discussion of the canard surface can be found in Reference 1.

The 40-inch radar antenna is possible by moving the radar antenna back in the airplane's nose and by a slight modification of the radome lines to prevent reflection of the radar energy. Gimballing of the antenna has been assumed to be about axes in the plane of the face of the antenna. Radar power

[redacted]

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[redacted]

has been increased from 250 KW to 500 KW. Reference 2 discusses the improved radar performance. The higher speed of this airplane requires that a higher-temperature-resisting radome be used. The Plaskon 911-11 resin with Fiberglass in the F-102A radome is satisfactory up to about 300°F. VIBRIN X-1047 resin is satisfactory to about 500°F and PDL-7-669 is satisfactory for between 500° - 600°F. Both of these materials have specific gravities and dialectric characteristics similar to those of the F-102A, but a test program will have to be conducted to determine the most satisfactory material.

The Pratt & Whitney J75 of the F-102B has been replaced by an advanced version known as the JT4B-21. This engine has a static sea level rating of 25,000 pounds with afterburner and is expected to be available by August of 1958. This engine fits into the F-102B airframe with very minor modifications. However, the increased thrust of the dry engine plus the increased thrust from precompressor cooling, requires that the engine inlets be increased from 1060 sq. in. to 1840 sq. in. This requires a refairing of the fuselage lines for some distance back from the inlets. 1540

Precompressor cooling consists of injecting water into the engine air ducts downstream of the last shock wave (see Reference 3). The water tank will have heat added from an airplane source to keep it from freezing when the airplane is flying and from an external power source with thermostatically controlled electric heaters when on the ground during freezing temperatures. Insulation may be incorporated around the water tank to conserve heat. A water tank with a capacity of 180 gallons is provided.

A biconvex airfoil section with a thickness ratio of 3.85% has replaced the Case XIV camber on the 0004-65 airfoil of the F-102B. The biconvex airfoil forward of the 50% chord point is parabolic and aft of this point it is identical with that of the F-102B.

The leading edge of the wing has a nose flap to provide variable camber. This flap hinges along a line 10% of the wing semi-span. The same wing area as on the F-102B is used as on all airplanes in this study. The wing is constructed of stainless steel sandwich skin with titanium spars and ribs. The wing construction is covered in more detail in Reference 4. The fuselage is to be constructed of titanium.

[redacted]

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[redacted]

The vertical tail area has been increased to 120 square feet to offset the decrease in lift curve slope that occurs at the higher Mach number that this airplane can attain. The vertical tail will have construction similar to that of the wing.

The pilot escape capsule is of the type being developed by Stanley Aviation Corporation for the Air Force based on the F-102A. (Reference 5). A pressure-tight door closes under the pilot when he decides to escape and the whole windshield, canopy, seat and part of the cockpit is ejected by the thrust of a RATO unit. A stabilizing boom keeps the ejected unit from tumbling and a static line attached to the fuselage releases the drogue chute when the escape capsule has cleared the airplane. When the speed is reduced and the capsule is below, say, 10,000 feet altitude, an automatic release permits the drogue parachute to deploy the main parachute. The pilot lands in the escape capsule, protected from the high dynamic pressure, low ambient pressure and low ambient temperatures.

There is space in the missile bay for two unguided Ding Dongs or for two guided nuclear warhead rockets (hereafter called "Guided Ding Dongs"). The guided Ding Dong shown in Figure 2 is based on a short study to see if a guided weapon could be designed to fit into the same space as the unguided Ding Dong. This weapon is based on a smaller sized warhead than the one in the unguided Ding Dong, but with at least as large a yield. In all airplane configuration studies, weight allowances for the larger, unguided Ding Dongs have been made (815 pounds each) although it is anticipated that a guided version with new warhead development might be made for less weight. Another weapon, a larger unguided Ding Dong of 100 KT yield, appears useful for forward area interceptors. Such a missile, weighing approximately 1800 pounds, could also be carried in the missile bay as an alternate to two Ding Dongs.

One Ding Dong and four Falcon missiles could be carried as in the F-102B, if the missile bay fuel tank were to be removed and this fuel carried in the water tank and the water carried in smaller tanks around the ducts.

[redacted]

Internal fuel tanks consist of the following:

[redacted]

1.	Main fuselage tanks	189 Gals.	1228 Lbs.
2.	Missile bay tank	212 Gals.	1375 Lbs.
3.	Main wing tanks	890 Gals.	5790 Lbs.
4.	Aft wing tank (formerly a transfer tank)	233 Gals.	1514 Lbs.
	Totals	1524 Gals.	9907 Lbs.

Table 1 shows the group weight statement for this airplane and Figure 3 indicates how the center of gravity varies with airplane weight.

[redacted]

TABLE I.

GROUP WEIGHT STATEMENT

JT 4 B - 21

Wing		4,021
Tail		1,035
Fuselage		5,563
Landing Gear		1,432
Surface Controls		467
Engine Section		33
Propulsion Group		8,487
Engine	5,910	
Air Induction	1,190	
Exhaust System	236	
Cooling System	84	
Lubrication System	53	
Fuel and Water Injection System	912	
Engine Controls	29	
Starting System	73	
Instruments and Navigation		137
Hydraulics and Pneumatics		357
Electrical Group		602
Communications and Electronics		2,253
Armament Provisions		235
Furnishings and Equipment		360
Air Conditioning and Anti-Icing		355
Auxiliary Gear		60
Weight Empty		25,397
Useful Load		13,603
Pilot	215	
Oil - Trapped and Engine	60	
Trapped Fuel	274	
Trapped Water	17	
Internal Fuel	9,907	
Water	1,500	
Armament	1,630	
Gross Take-Off Less Externals		39,000
External Tanks and Pylons		581
External Fuel		6,700
Gross Take-Off with Externals		46,281

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Configuration with X84B Engine

The second combination studied differed mainly from the first one in that a General Electric X84B engine of 29,100 pounds static sea level thrust with afterburner has replaced the Pratt & Whitney JT4B-21 with precompressor cooling. Like the first combination, the following items differ from those of the F-102B:

1. Canard control system
2. 40-inch radar antenna
3. Increased power in radar
4. Biconvex airfoil section (3.85% thick)
5. Variable camber leading edge
6. Pilot escape capsule
7. Guided nuclear warhead rockets
8. Larger external tanks (two 460-gals).

An inboard profile of this airplane is shown in Figure 4. Even with the same wing area as the F-102B, this airplane is able to take-off and land in about the same distances as the F-102B because of the canard tail (see Reference 1).

Internal fuel tanks with the following capacities are provided in this airplane:

1.	Main fuselage tank	189 Gals.	(1228 Lbs.)
2.	Missile bay tank	212 Gals	(1375 Lbs.)
3.	Between duct tank	350 Gals.	(2275 Lbs.)
4.	Main wing tank	890 Gals.	(5790 Lbs.)
5.	Aft wing tank (formerly a transfer tank on F-102B)	233 Gals. 1874 Gals.	(1514 Lbs.) (12,182 Lbs.)

[redacted]

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[redacted]

Table 2 presents the group weight statement for this airplane and Figure 5 shows the variation of center of gravity with weight.

Configuration with J89 Engine:

The third combination studied differs mainly from the first in that an Allison J89 engine of 37,500 pounds static sea level thrust replaces the Pratt & Whitney JT4B-21 with precompressor cooling. Configurations with and without precompressor cooling were studied. Like the first configuration, this airplane differs from the F-102B because of the addition of the following items:

1. Canard control system
2. 40-inch radar antenna
3. Increased power in the radar
4. Biconvex airfoil section (3.85% thick)
5. Variable camber leading edge
6. Pilot escape capsule
7. Guided nuclear warhead rockets
8. Larger external tanks (two 460-Gals.)

This configuration is shown in Figure 4 using the lines indicated for the J89 engine.

With the same wing area as the F-102B, the canard tail makes it possible for this airplane to take off and land in about the same distance as the F-102B, even with the fairly large weight increase (see Reference 1).

Table 3 shows the weight and balance of this airplane without precompressor cooling and Figure 6 shows the c.g. travel as a function of airplane weight during the mission. Proper fuel sequencing keeps the center of gravity inside of the allowable limits. The canard tail eliminates the need for the fuel transfer system of the F-102B.

[redacted]

TABLE II
GROUP WEIGHT STATEMENT
X - 84 B

Wing		4,065
Tail		1,035
Fuselage		6,007
Landing Gear		1,412
Surface Controls		467
Engine Section		33
Propulsion Group		9,382
Engine	6,090	
Air Induction	1,871	
Exhaust System	236	
Cooling System	84	
Lubrication System	53	
Fuel System	946	
Engine Controls	29	
Starting System	73	
Instruments and Navigation		137
Hydraulics and Pneumatics		357
Electrical Group		602
Communications and Electronics		2,253
Armament Provisions		235
Furnishings and Equipment		233
Air Conditioning and Anti-Icing		355
Auxiliary Gear		60
Weight Empty		26,633
Useful Load		14,376
Pilot	215	
Oil - Trapped and Engine	60	
Trapped Fuel	289	
Trapped Water	---	
Internal Fuel	12,182	
Water	---	
Armament	1,630	
Gross Take-Off Less Externals		41,009
External Tanks and Pylons		516
External Fuel		5,948
Gross Take-Off With Externals		47,473

TABLE III

GROUP WEIGHT STATEMENT

XJ 89-A1

Wing		4,104
Tail		1,035
Fuselage		6,137
Landing Gear		1,412
Surface Controls		467
Engine Section		33
Propulsion Group		10,526
Engine	7,350	
Air Induction	1,755	
Exhaust System	236	
Cooling System	84	
Lubrication System	53	
Fuel System	946	
Engine Controls	29	
Starting System	73	
Instruments and Navigation		137
Hydraulics and Pneumatics		357
Electrical Group		602
Communications and Electronics		2,253
Armament Provisions		235
Furnishings and Equipment		233
Air Conditioning and Anti-Icing		355
Auxiliary Gear		60
Weight Empty		27,946
Useful Load		14,376
Pilot	215	
Oil - Trapped and Engine	60	
Trapped Fuel	289	
Trapped Water	---	
Internal Fuel	12,182	
Water	---	
Armament	1,630	
Gross Take-Off Less Externals		42,322
External Tanks and Pylons		516
External Fuel		5,948
Gross Take-Off With Externals		48,786

[redacted]

Internal fuel tanks with the following capacities are provided in this airplane:

1.	Main fuselage tank	189 Gals.	(1228 Lbs.)
2.	Missile bay tank	212 Gals.	(1375 Lbs.)
3.	Between-duct tank	350 Gals.	(2275 Lbs.)
4.	Main wing tanks	890 Gals.	(5790 Lbs.)
5.	Aft wing tank	233 Gals.	(1514 Lbs.)
	(formerly a transfer tank)	1874 Gals.	(12,182 Lbs.)

Table 4 shows the weight summary of the same airplane with precompressor cooling.

[redacted]

TABLE IV.

GROUP WEIGHT STATEMENT

XJ 89-A1

PRECOMPRESSOR COOLING

Wing		4,104
Tail		1,035
Fuselage		6,137
Landing Gear		1,412
Surface Controls		467
Engine Section		33
Propulsion Group		10,968
Engine	7,350	
Air Induction	1,999	
Exhaust System	236	
Cooling System	84	
Lubrication System	53	
Fuel System	1,144	
Engine Controls	29	
Starting System	73	
Instruments and Navigation		137
Hydraulics and Pneumatics		357
Electrical Group		602
Communications and Electronics		2,253
Armament Provisions		235
Furnishings and Equipment		233
Air Conditioning and Anti-Icing		355
Auxiliary Gear		60
Weight Empty		28,388
Useful Load		15,893
Pilot	215	
Oil - Trapped and Engine	60	
Trapped Fuel	289	
Trapped Water	17	
Internal Fuel	12,182	
Water	1,500	
Armament	1,630	
Gross Take-Off Less Externals		44,281
External Tanks and Pylons		516
External Fuel		5,948
Gross Take-Off with Externals		50,745

[redacted]

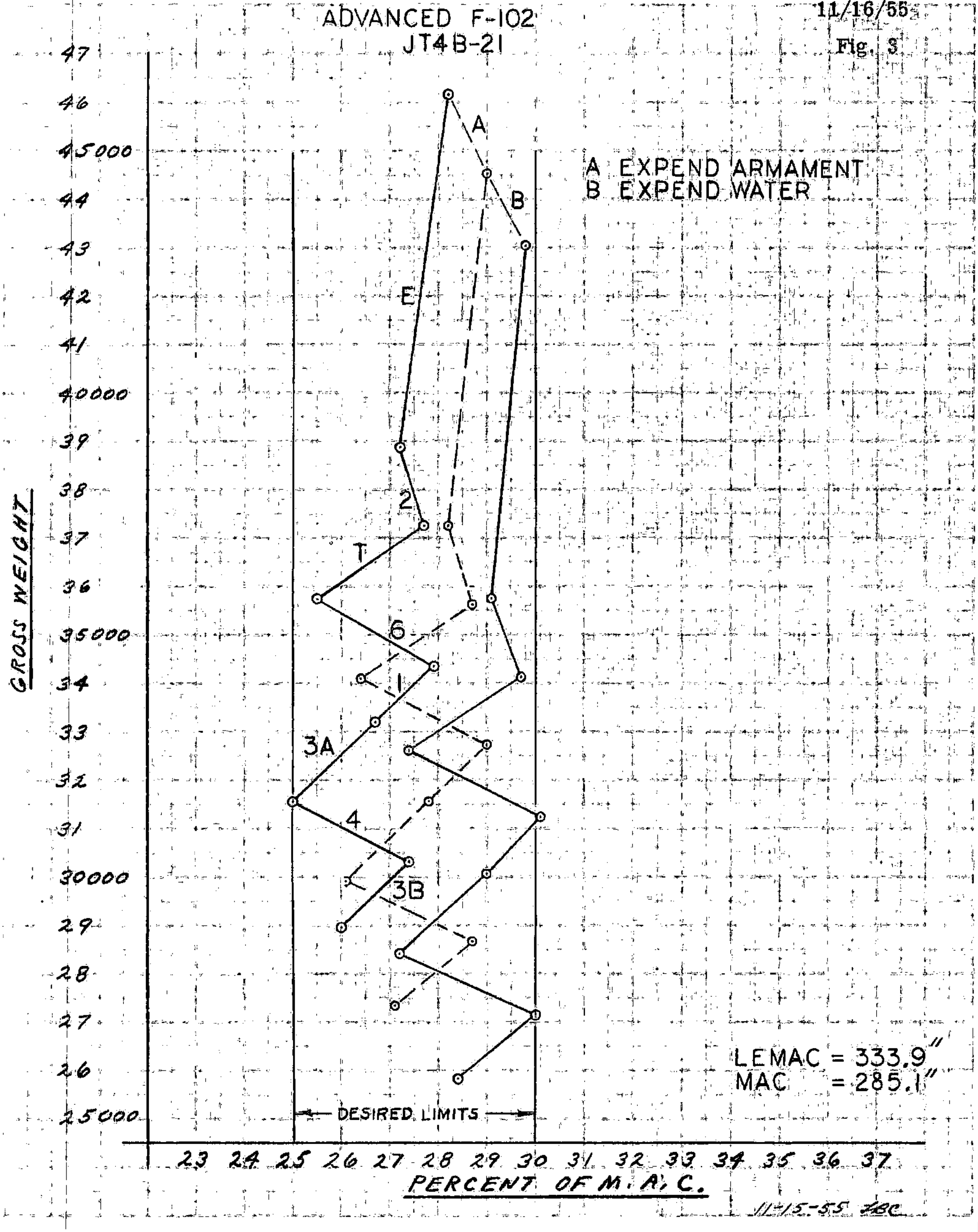
CONCLUSIONS

A target date of 1959 appears feasible to develop a prototype of the advanced F-102 equipped with the Pratt & Whitney JT4B-21 engine and precompressor cooling. This airplane has a take-off gross weight of 39,000 pounds without external tanks and 46,300 pounds with two 515-gallon tanks and pylons.

An intensive development program as outlined in the reference reports should make it possible to produce a prototype as described in this report in about three years.

[redacted]

Fig. 3



11-15-55 FBC

Fig. 5

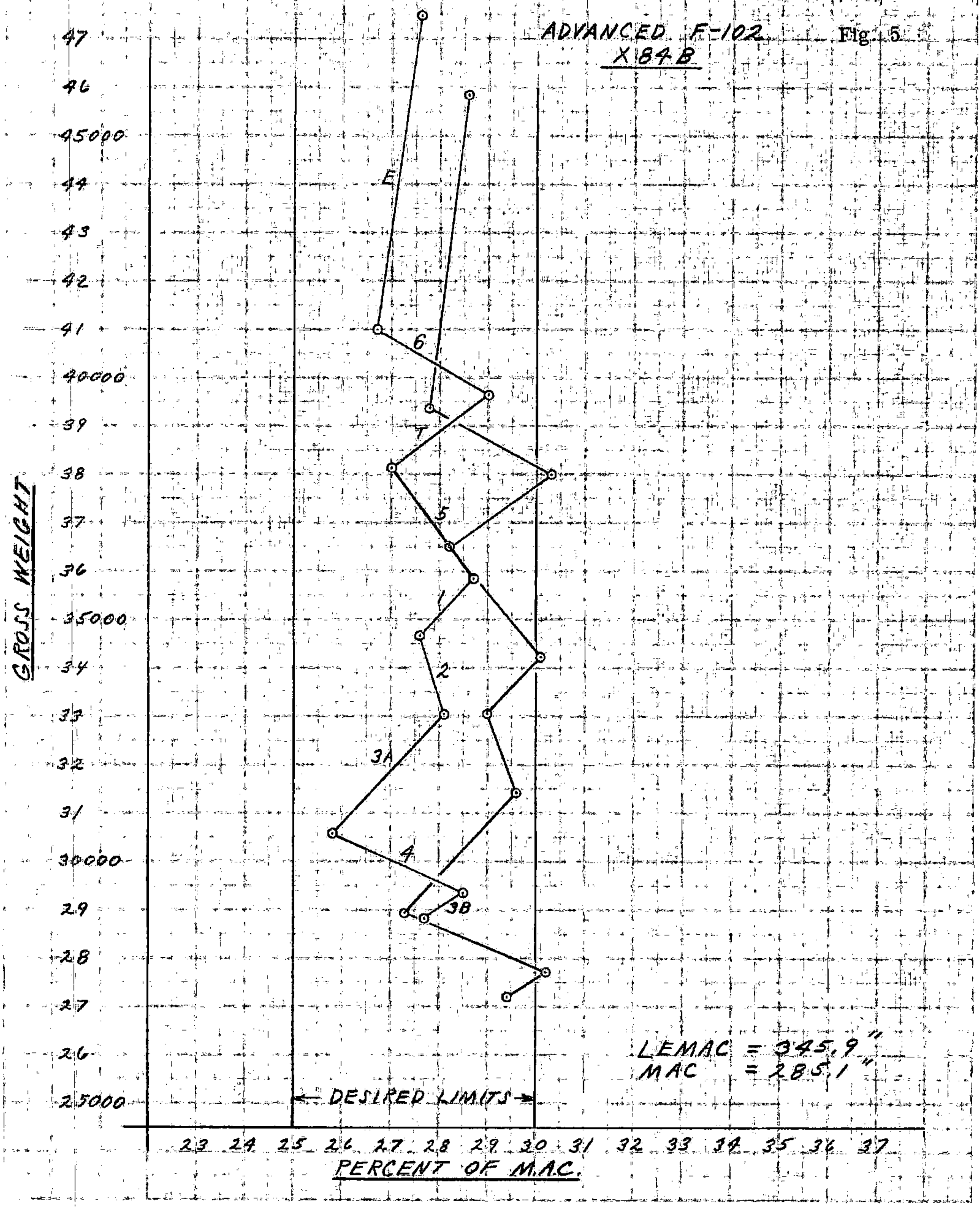
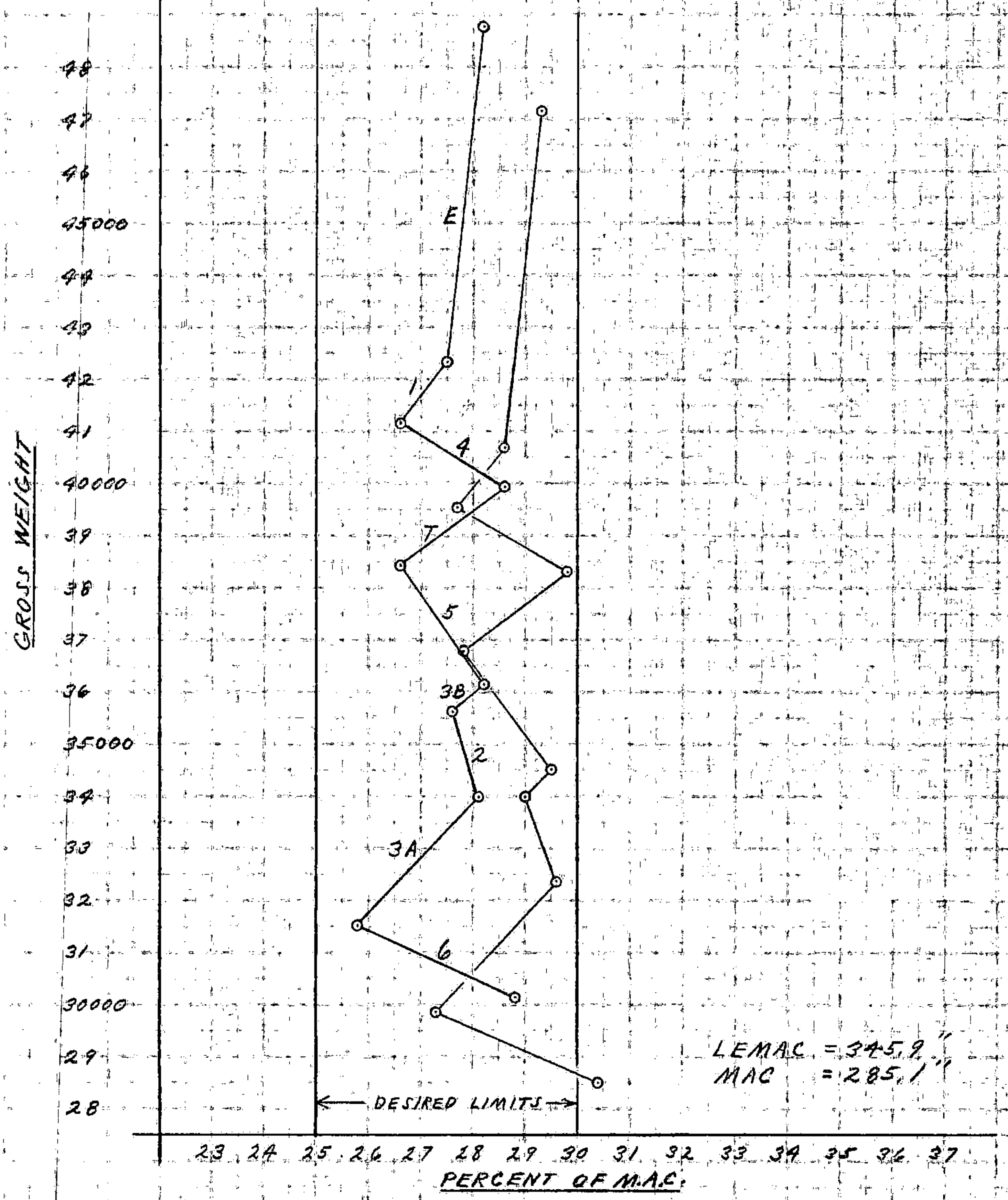

ADVANCED F-102
X 84 B
GROSS WEIGHT
47
46
45000
44
43
42
41
40000
39
38
37
36
35000
34
33
32
31
30000
29
28
27
26
25000
E
6
7
5
1
2
3A
4
3B
← DESIRED LIMITS →
LEMAC = 345.9"
MAC = 285.1"
23
24
25
26
27
28
29
30
31
32
33
34
35
36
37
PERCENT OF M.A.C.

Fig. 6

ADVANCED F-102
XJ89-A1



LEMAC = 345.9"
MAC = 285.1"

[redacted]

REFERENCES

1. Mawhinney, R. F., Jr. "Estimated Performance, Stability and Control Characteristics - Advanced F-102", Convair-San Diego Report ZA-8-018, dated 18 November 1955 - Secret.

2. Lehman, D. B., Osborn, Palmer, Stamper, J. C., "Systems Considerations for an Advanced Interceptor in Continental Air Defense", Convair-San Diego Report ZO-8-012, dated 30 November 1955 - Secret.

3. Thermodynamics Group, "Preliminary Study Results and Proposed Development Program for the Propulsion System of the Advanced F-102 Program", Convair-San Diego Report T-I-341, dated 23 November, 1955 - Confidential.

4. Westbrook, R. C. and Hughes, J. E., "Proposed Structural Material and Structural Arrangement for the Advanced F-102 Airplane", Convair-San Diego Report ZS-224, dated 8 November 1955 - Secret.

5. D'Anna, P. W., "Preliminary Design Study Report, Seat Capsule; Pilot's Emergency Escape", Stanley Aviation Corporation, Doc. No. 184, dated 16 September 1953 - Confidential.

6. Thermodynamics Group, "Preliminary Study of Aerodynamic Heating, Cabin Conditioning and Equipment Cooling Problems and Proposed Development Program for the Advanced F-102 Airplane", Convair-San Diego Report T-I-339, dated 21 November 1955 - Confidential.

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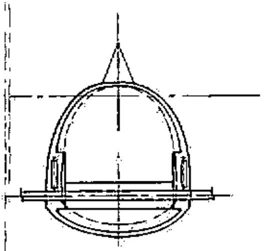
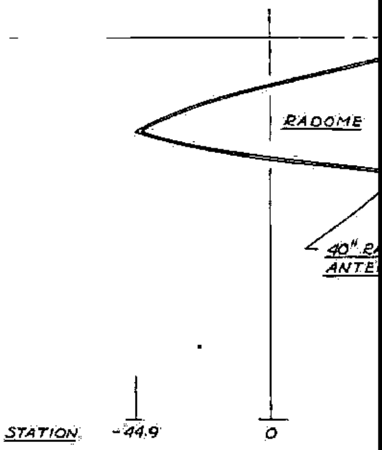
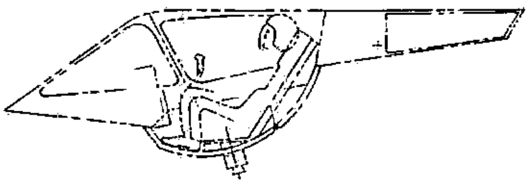
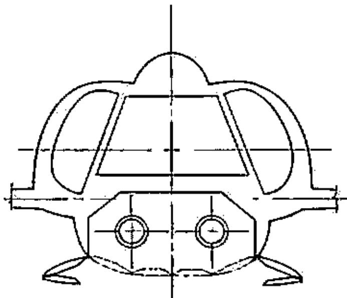


SEC. STA. 102





ESCAPE CAPSULE



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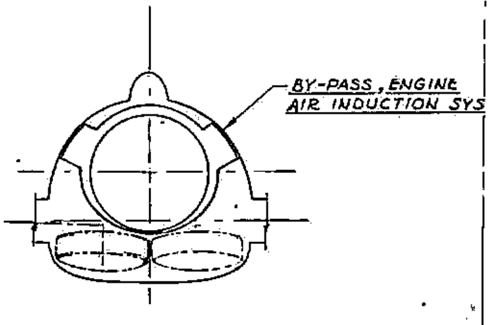
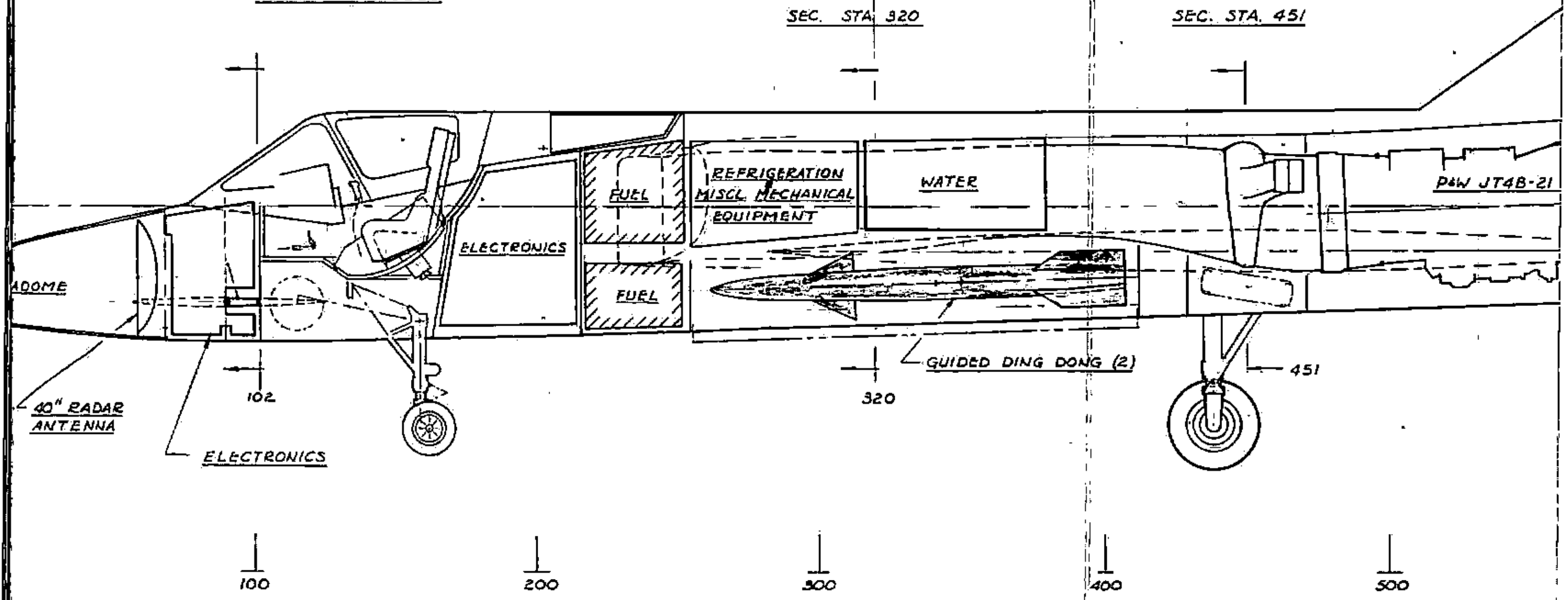
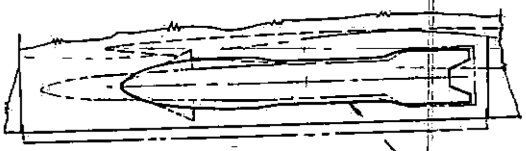


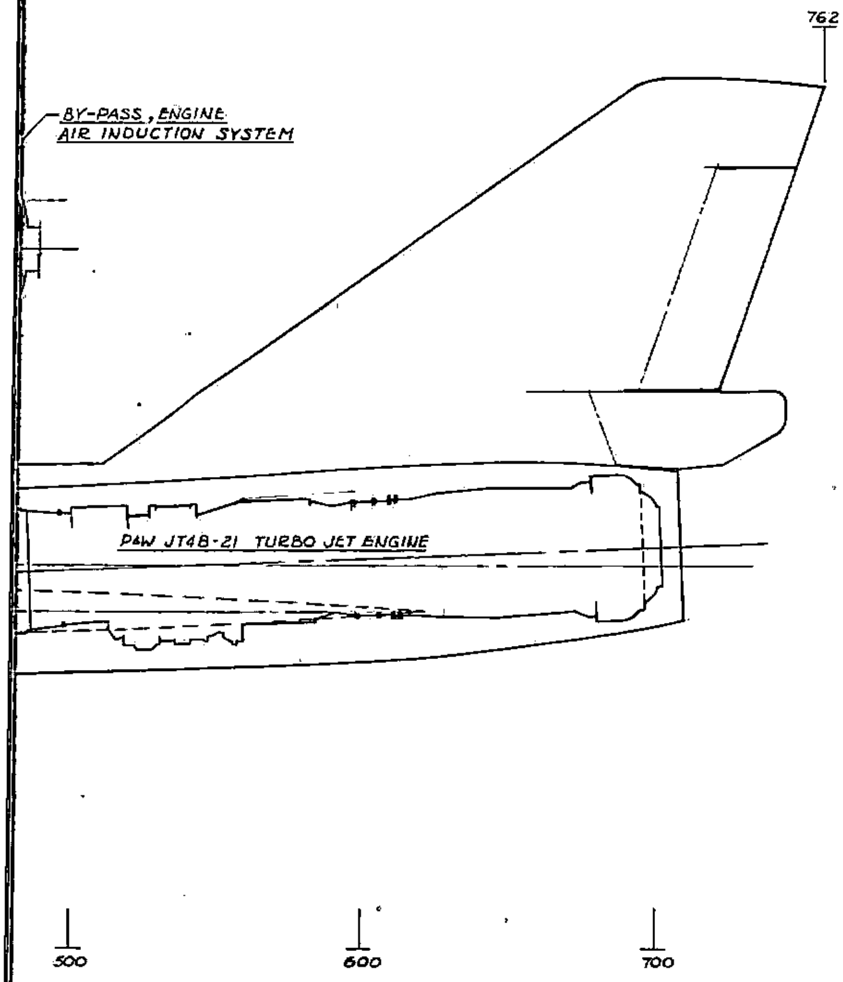
SEC. STA. 451





ALTERNATE ARMAMENT
UNGUIDED DING DONG (2)

FIG. 2


762
BY-PASS, ENGINE AIR INDUCTION SYSTEM
P&W JT4B-21 TURBO JET ENGINE
500
600
700

CONFIDENTIAL

CONVAIR

PRELIMINARY DESIGN DRAWING

INBOARD PROFILE - ADVANCED
F102 WITH P&W JT4B-21 ENGINE
& PRECOMPRESSOR COOLING

BY | CHECKED | APPROVED | SCALE 1/20 | DATE 31 OCT '55

DATA

	WING	VERTICAL TAIL	CANARD	SPEED BRAKE
AREA, SQ. FT.	697	120	61	14.65 (PROJECTED)
SWEEP, L.E., DEG.	60	55	45	
ASPECT RATIO	2.2	1.0	4.0	
AIR FOIL	(1)	(1)	(2)	
DIHEDRAL DEG.	0	0	0	
INCIDENCE DEG.	0	0		

(1) BI-CONVEX, L.E. TO .5c
0003.85-65, .5c TO T.E.

(2) BI-CONVEX, .03 MAX t/c

PROPULSION

ENGINE P&W JT4B-21 TURBO-JET
WITH PRECOMRESSOR COOLING

THRUST, S.L.S., LB.- MIL, 16500 MAX 25000

FUEL (J P4)

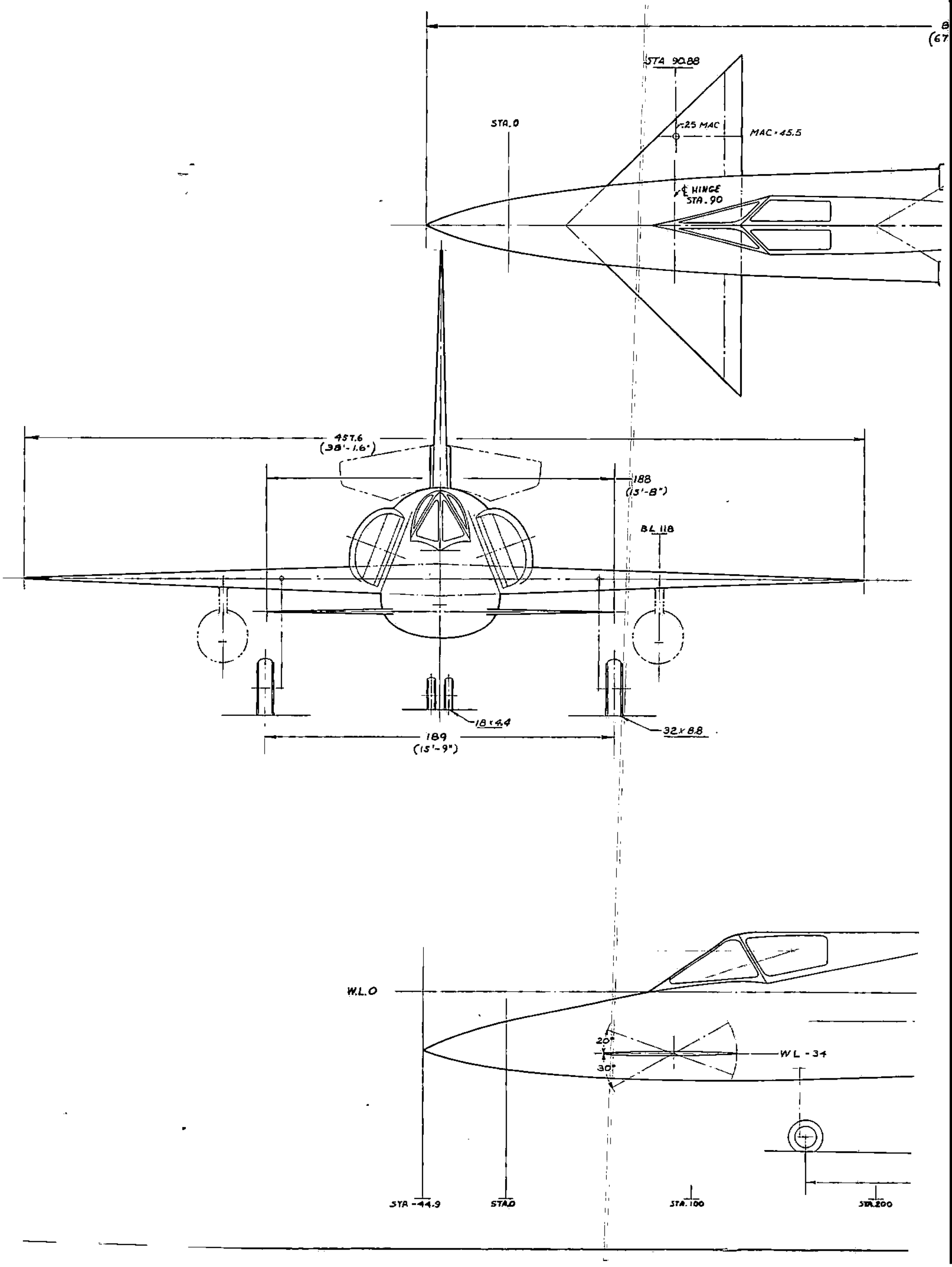
CAPACITY TOTAL, GAL/LB	2555 / 16607
INTERNAL, GAL/LB	1524 / 9907
EXTERNAL, GAL/LB	1031 / 6700

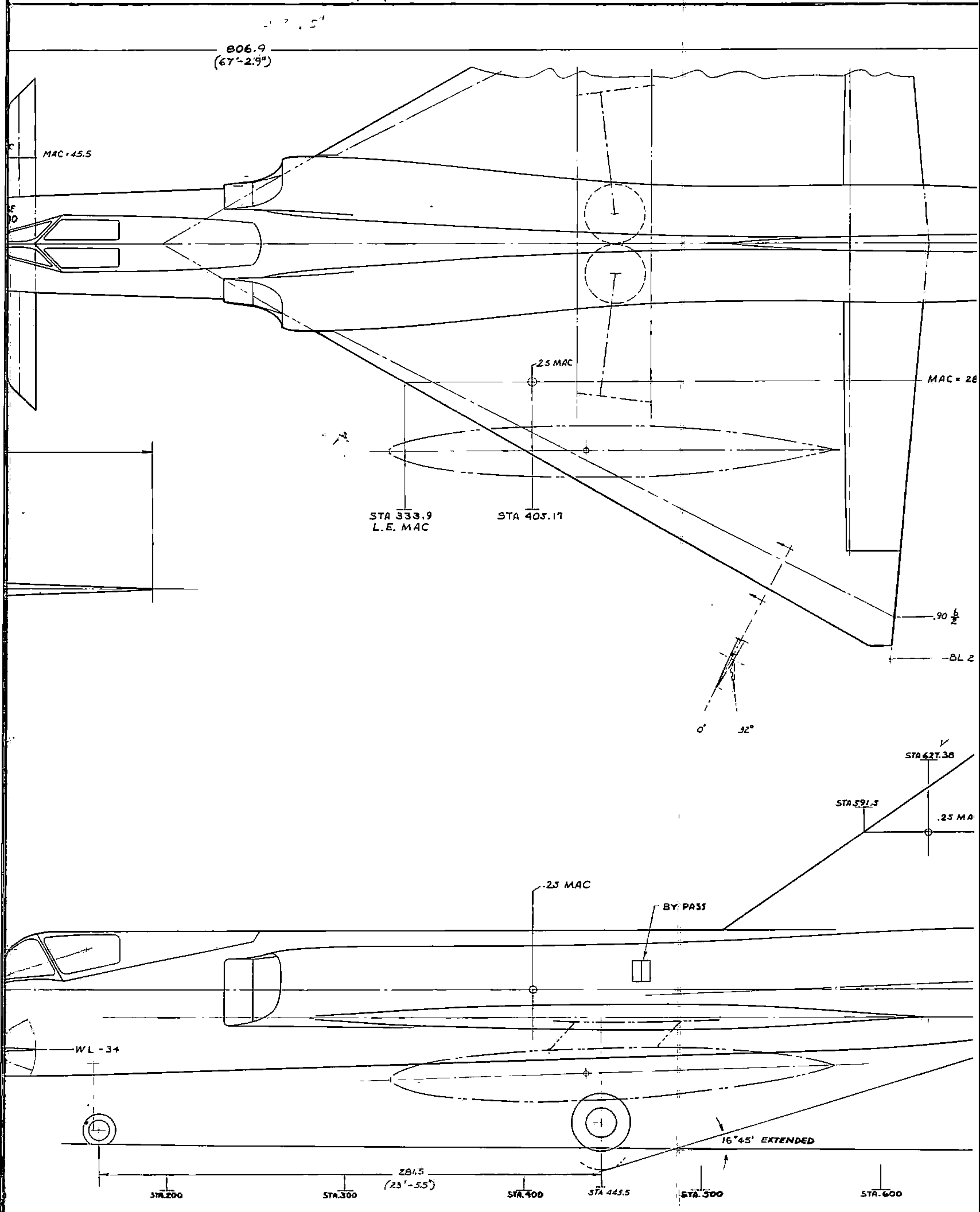
COOLANT (WATER)

CAPACITY TOTAL, GAL / LB	181 / 1500

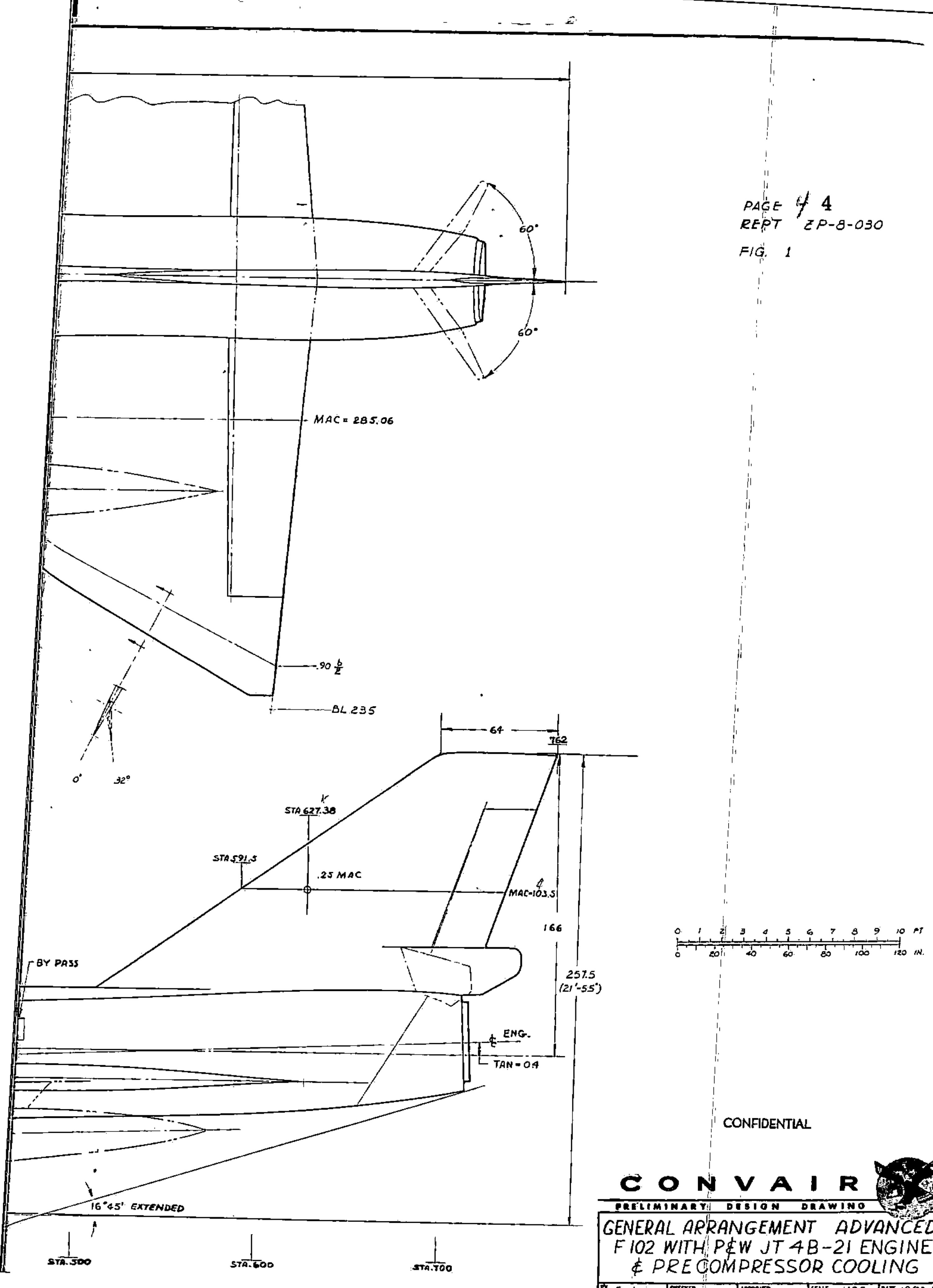
AIR INDUCTION SYSTEM

INLET CAPTURE AREA (TOTAL), SQ. IN.	1840 ~~1460~~
BY-PASS AREA (TOTAL), SQ. IN.	200


8
(67
STA 90.88
STA. 0
.25 MAC
MAC = 45.5
℄ HINGE
STA. 90
457.6
(38'-1.6")
188
(15'-8")
BL 118
18 x 4.4
32 x 8.8
189
(15'-9")
W.L.0
20°
30°
W L -34
STA -44.9
STA 0
STA. 100
STA 200


806.9
(67'-2.9")
MAC = 45.5
.25 MAC
MAC = 28
STA 333.9
L.E. MAC
STA 405.17
.90 b/2
BL 2
0°
32°
STA 627.38
STA 591.5
.25 MA
.25 MAC
BY PASS
WL -34
16°45' EXTENDED
281.5
(23'-5.5")
STA 200
STA 300
STA 400
STA 443.5
STA 500
STA 600

PAGE 4 4
REPT ZP-8-030
FIG. 1



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CONVAIR
PRELIMINARY DESIGN DRAWING

GENERAL ARRANGEMENT ADVANCED-
F 102 WITH P&W JT 4B-21 ENGINE
& PRECOMPRESSOR COOLING

BY Szekmi | CHECKED | APPROVED | SCALE 1:20 | DATE 10/28 '55

CONVAIR—SAN DIEGO, CALIFORNIA | SD-55-08076

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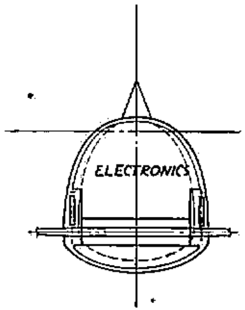
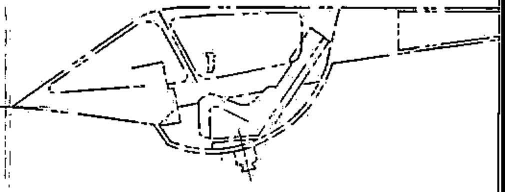
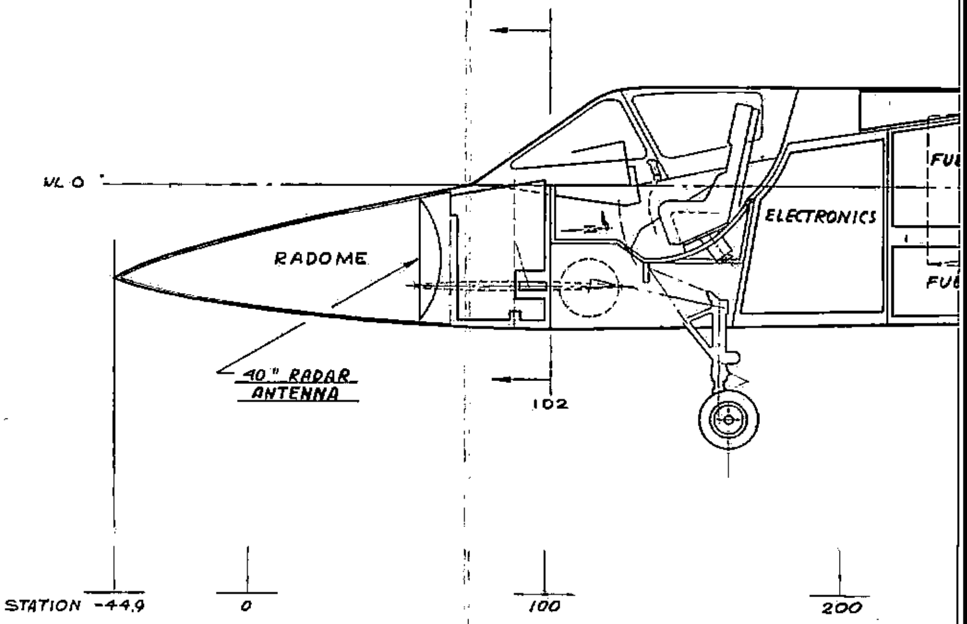


SEC. STA. 102



ESCAPE CAPSULE



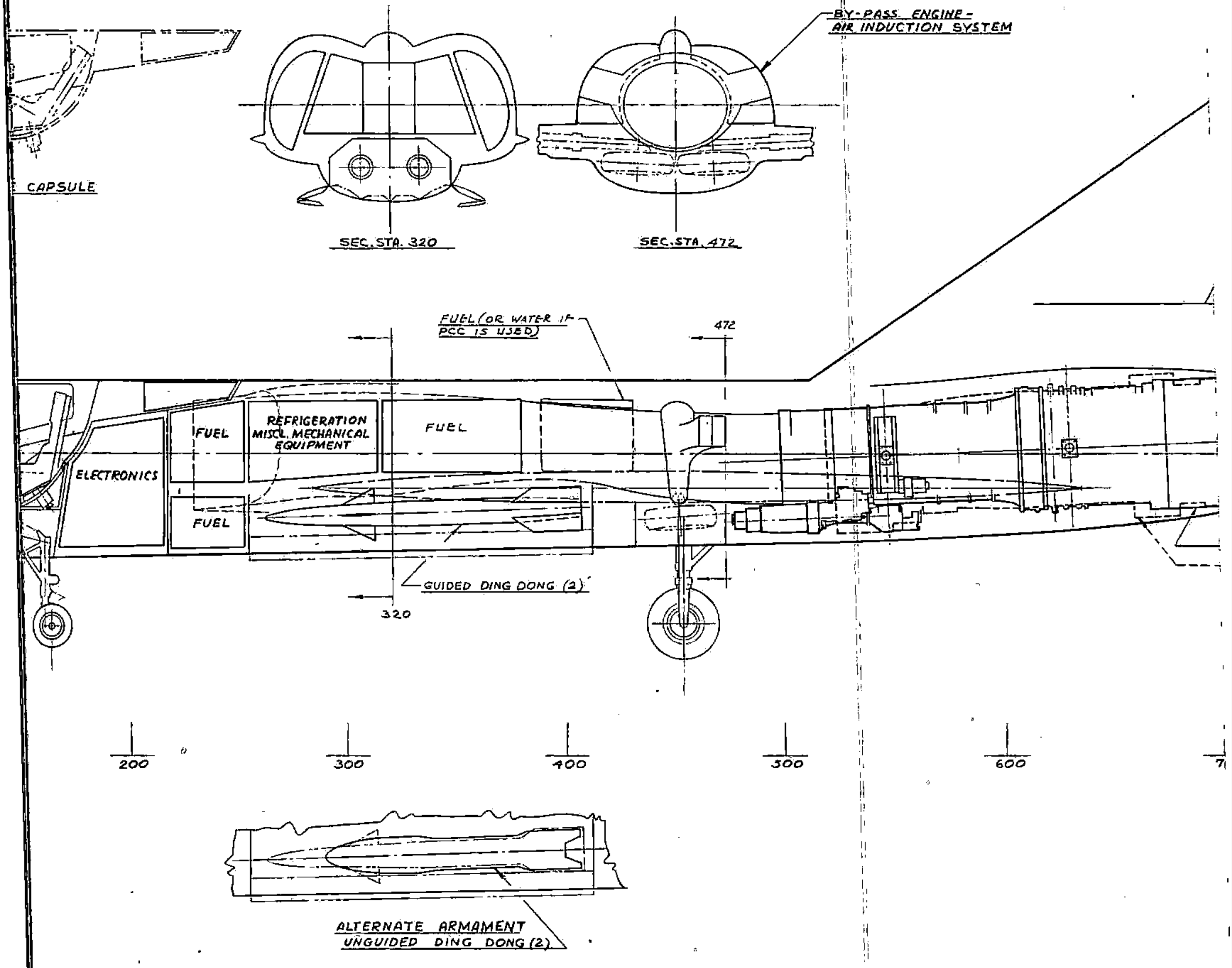
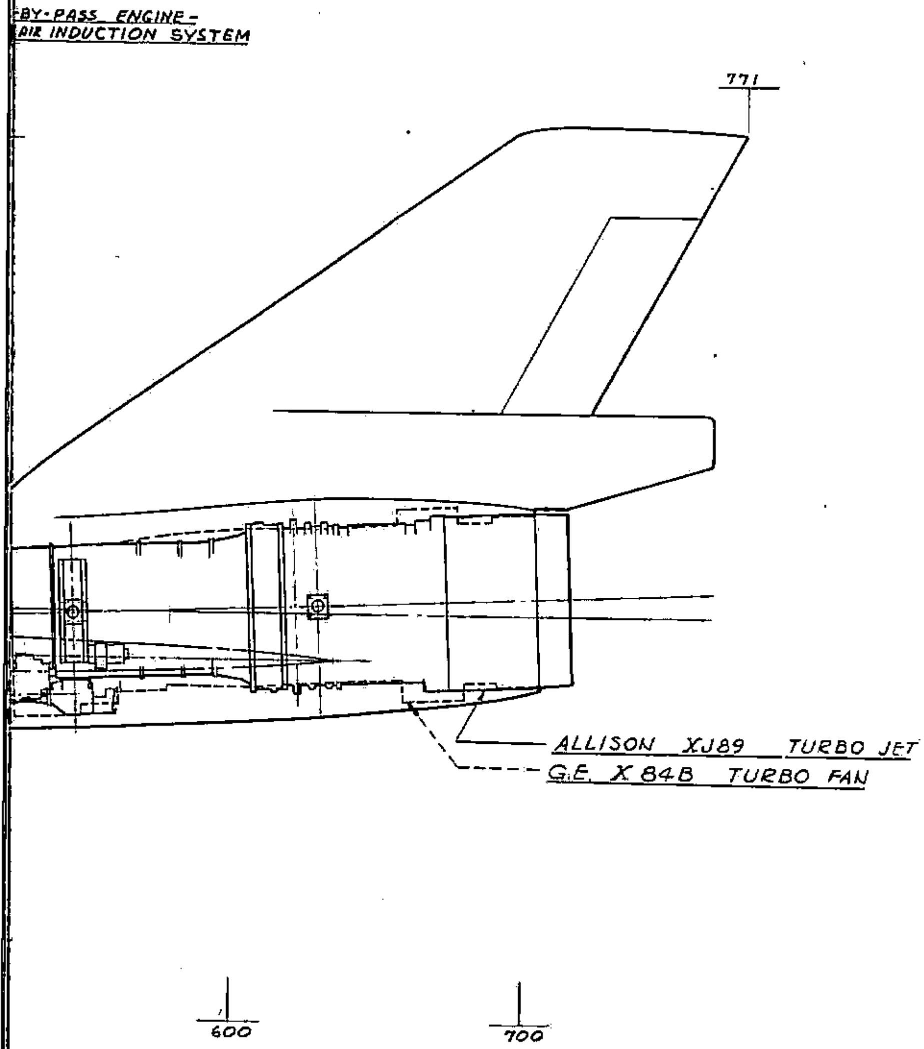

BY-PASS ENGINE - AIR INDUCTION SYSTEM
CAPSULE
SEC. STA. 320
SEC. STA. 472
FUEL (OR WATER IF PCC IS USED)
472
ELECTRONICS
FUEL
REFRIGERATION MISCL. MECHANICAL EQUIPMENT
FUEL
FUEL
GUIDED DING DONG (2)
320
200
300
400
500
600
ALTERNATE ARMAMENT UNGUIDED DING DONG (2)

FIG. 4



CONFIDENTIAL

C O N V A I R

PRELIMINARY DESIGN DRAWING

INBOARD PROFILE - ADVANCED F102
WITH G.E. X84B OR ALLISON XJ89 ENGINE
& PRE COMPRESSOR COOLING

BY Seskocs	CHECKED	APPROVED	SCALE 1:20	DATE NOV. 16 '55

CONVAIR—SAN DIEGO, CALIFORNIA SD-55-08080

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Filed in Mes
Tube

[redacted]

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP8-063
DATE June 1959
MODEL

UNCLASSIFIED

TITLE

F102

STRIKE ATTACK AIRPLANE

MISCELLANEOUS STUDIES

PREPARED BY T.M. Hemphill
T. M. Hemphill

GROUP

REFERENCE

CHECKED BY R. C. Lingley
R. C. Lingley

APPROVED BY A. R. Lambert
A. R. Lambert

Classification Canceled
CD Form 254   10 June 60
Authorized by: Date
M.C. Tomlin 6130 6-11-68
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[redacted]

FORM 1812 A-4

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FORWARD

This report supplements Convair Report ZP-59-08002 dated May 1959, and is to answer specific questions about the F102 Strike Attack Aircraft.

The report is divided into three sections. The first section shows a new internal arrangement of the mid lower fuselage space to provide more internal fuel. This arrangement also provides for specific electronics different from that originally planned. The resulting armament loadings and performance changes are also shown in the first section.

In the second section the basis for the performance estimates is given.

The third section is devoted to a brief determination of the fatigue life of the aircraft in the strike attack role.

UNCLASSIFIED

[redacted]

TABLE OF CONTENTS

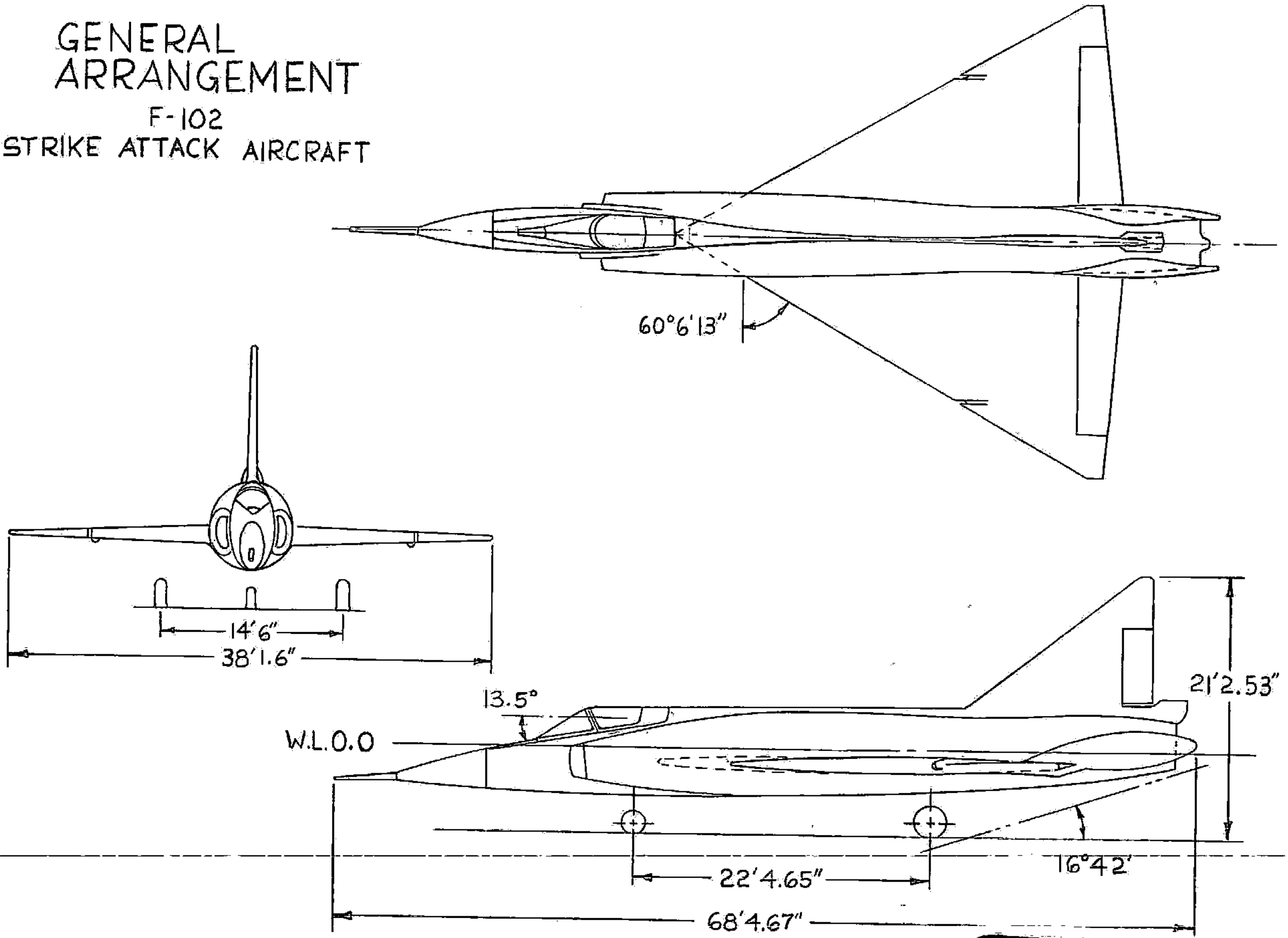
ITEM	PAGE
FORWARD	i
GENERAL ARRANGEMENT	1
F102 DESCRIPTION	2
INBOARD PROFILE	3
INBOARD PROFILE - DESCRIPTION	4
TYPICAL LOADINGS	7
ELECTRONICS	9
WEIGHTS	11
PERFORMANCE	12
MISSIONS	14
DRAG AND THRUST EXTIMATE BASIS - SECTION II	20
FATIGUE LIFE - SECTION III	34

[redacted]

GENERAL ARRANGEMENT

F-102

STRIKE ATTACK AIRCRAFT



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F-102 STRIKE ATTACK AIRCRAFT DESCRIPTION

The F-102 Strike Attack Aircraft is an all-weather weapon system designed to destroy enemy ground targets and make air interceptions under good conditions of visibility. *

It is a single seat supersonic fighter, an outgrowth of the F-102A, the USAF's only operational supersonic all-weather interceptor. The fuselage features a pointed nose radome, a bubble canopy, side-located engine air inlet ducts, and a "waisted" midsection designed to the area rule.

The F-102 has no horizontal tail. It is equipped with trailing edge elevons which act as elevators and ailerons. The cockpit controls are the conventional stick and rudder pedals.

Another important feature is the construction of the main fuel tanks which are integral with the wing. These tanks carry a large portion of the required fuel leaving adequate fuselage space for electronic equipment and armament.

Convair has developed techniques for rapid automatic check-out (RACO) of airplane systems by electronic means. This feature can be included if desired by the customer.

The F-102 Strike Attack Aircraft is powered with a Pratt and Whitney J-57 dual spool axial flow jet engine equipped with an afterburner.

* Sidewinder limitation.

[redacted]

F-102 STRIKE ATTACK AIRCRAFT

INBOARD PROFILE DESCRIPTION

The F-102 Strike Attack airplane differs from the Interceptor as follows:

(a) The nose section ahead of the pilot's compartment is revised to accommodate radar and fire control systems with air-to-ground and air-to-air capabilities instead of air interception only.

(b) An air refueling system has been added.

(c) The pilot's canopy is changed to the "bubble" type.

(d) The missile bay in the bottom center of the fuselage is revised to accommodate bombs and additional fuel.

(e) Cockpit instruments and control panels are those appropriate to strike attack missions.

(f) Four wing stations, two on each side, have been provided to carry pylon mounted weapons or fuel. One of the wing stations on each side is outboard of the landing gear strut-to-wing attachment; the other is inboard.

(g) The wheels, brakes, and other landing gear components are redesigned to take the increased gross weights utilized in strike attack missions.

(h) An emergency arresting hook is added.

(i) Wherever necessary to accommodate the new loads; bombs, extra fuel, new equipment, etc., local structural revisions to maintain existing strength are made.

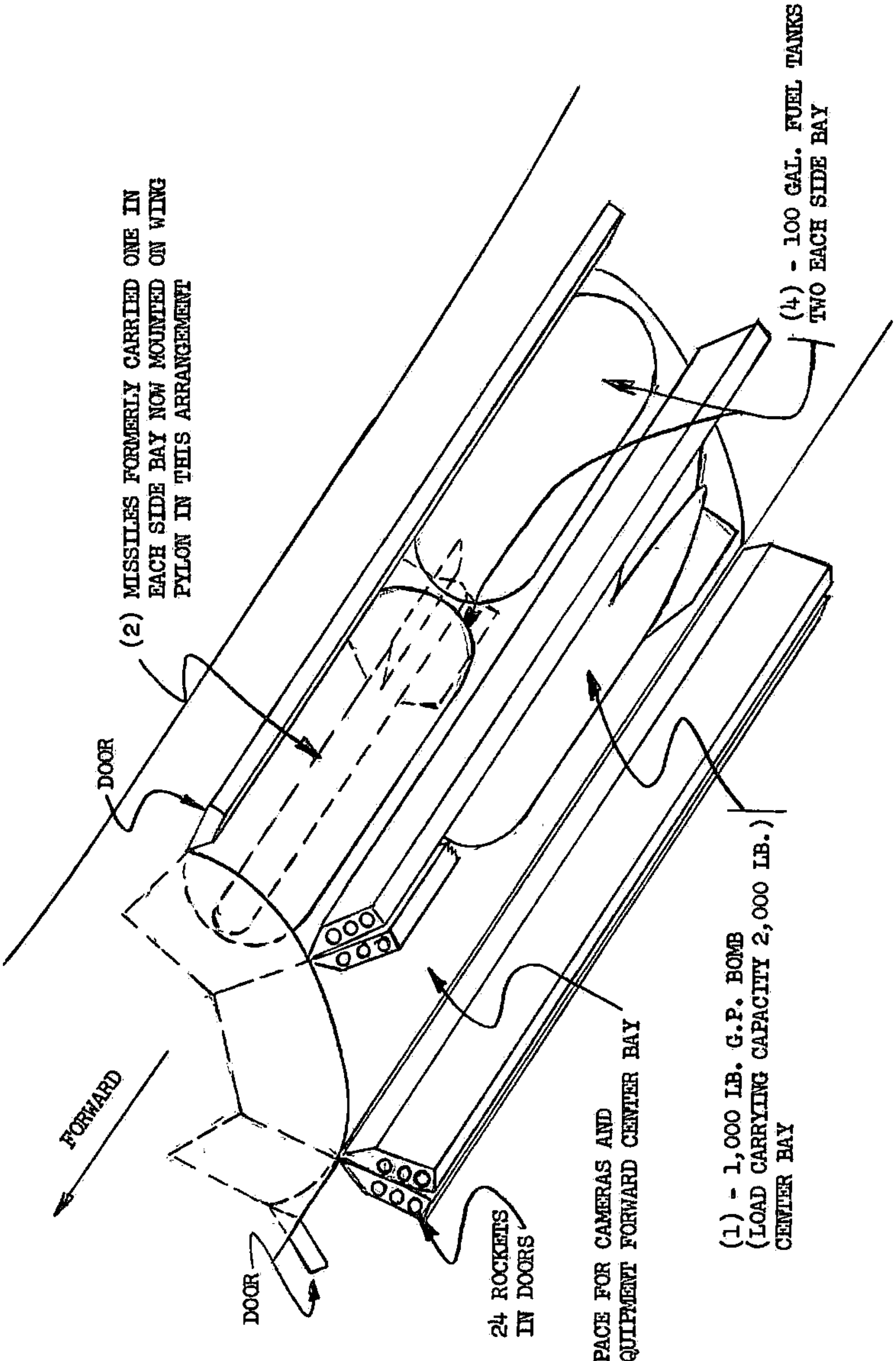
(j) Space provisions are made for a gun camera and strike assessment cameras.

(k) Wing leading edges are revised to include small fore and aft slots to take the place of fences which are removed.

(l) The engine is changed to the P & W JT3C-24 from the J57-P-23.

[redacted]

FUSELAGE INTERNAL ARRANGEMENT


(2) MISSILES FORMERLY CARRIED ONE IN EACH SIDE BAY NOW MOUNTED ON WING PYLON IN THIS ARRANGEMENT
(4) - 100 GAL. FUEL TANKS TWO EACH SIDE BAY
DOOR
FORWARD
DOOR
24 ROCKETS IN DOORS
SPACE FOR CAMERAS AND EQUIPMENT FORWARD CENTER BAY
(1) - 1,000 LB. G.P. BOMB (LOAD CARRYING CAPACITY 2,000 LB.) CENTER BAY

[redacted]

INTERNAL ARRANGEMENTS

There are several possible arrangements of armament and fuel in the mid-fuselage bay. The arrangement most favorable for armament loading and performance in combat, is when two sidewinders, one bomb and two fuel tanks are carried internally. This arrangement plus four wing stations, on which pylon mounted weapons or fuel tanks may be carried, provides a total of seven stations. This allows a maximum of flexibility for armament loadings with a maximum of armament carried internally. The characteristics of the airplane with this arrangement were shown in Convair Report ZP59-08002.

The arrangement in this report favors the air patrol mission. Fuel tanks in the fuselage replace the sidewinders which must, when carried, utilize wing stations. Although the sidewinders occupy wing stations that could be utilized for other armament, they allow the carriage of maximum internal fuel, and their drag is small. They cause a reduction in high speed of only M = .005 in the supersonic region and a negligible reduction subsonically.

[redacted]

STATION LOADING CAPABILITIES

The station loading capabilities chart (page 6) indicates in a general way what kind of stores can be accommodated in F-102 Strike Attack aircraft from a weight and space standpoint. The specific provisions for stores will be negotiated to satisfy the requirements of individual customers.

Stores vary widely in their dimensions, weight and drag. For this reason the chart is not intended to be limiting or all inclusive.

For example, the inboard wing stations are ahead of some positions of the landing gear during gear retraction or extension and, therefore, some of the long streamlined weapons cannot be carried without interference. On the other hand, short 1,000 lb. general purpose types can be carried there.

[redacted]

ELECTRONICS

ARMAMENT CONTROL

1. Radar System
2. Bomb Computer
3. Director Sight
4. Range Computer (sidewinder)
5. Inertial Reference (stable table)
6. Air Data Computer
7. Armament Transmitter and Control
8. Voice Recorder Provisions (for reconnaissance only)

COMMUNICATIONS, NAVIGATION AND LANDING SUBSYSTEMS

1. Doppler Navigator
2. AFCS - Pilot Assist
3. Navigation Computer (PHI)
4. UHF Communications Transceiver
5. Automatic Direction Finder
6. Tacan Navigation System
7. I.F.F./SIF Transponder
8. Intercom Set

WEAPON SYSTEM CAPABILITY

RADAR: For this study of the supersonic strike attack aircraft a lightweight, compact, all-purpose radar system was chosen. At least two such radars are available with the following functions: Ground mapping, air-to-ground ranging, air-to-air search and autotracking, terrain avoidance, and contour mapping. High resolution in the ground mapping mode is attained by the use of monopulse reception techniques. Radar signals can readily be used as computer inputs for the purpose of armament control, bombing control and navigation. Performance obtained will vary with equipment selected. Typical ranges are: Air-to-ground mapping 80-100 miles, air-to-air search 18-25 miles and air-to-air auto-track 10-25 miles.

[redacted]

BOMBING SYSTEM: Using radar or visual attack a variety of bombing modes including toss, loft and over-the shoulder may be utilized. For long slant ranges (between 25,000 and 45,000 feet) a standard deviation of 16 mils referred to the slant range at pull-up (CEP of 400 feet when launched at a slant range of 25,000 feet) is expected.

NAVIGATION: The expected positional accuracy of the doppler navigator is one percent of the distance traveled. The known position of the aircraft can be up-dated during the flight by inserting new positional data into the computer from radar, TACAN or visual sighting of known points. When the aircraft is within line of sight of a TACAN ground station the TACAN equipment has a position accuracy of a fraction of a mile.

COMMUNICATION: The UHF communications transmitter receiver has a line of sight range to another aircraft or to a ground station of 280 miles at 40,000 feet altitude.

AIR-TO-AIR MISSILE: Sidewinder infrared missiles may be utilized against enemy aircraft using either the optical or radar tracking modes of the aircraft fire control system.

[redacted]

F-102 STRIKE ATTACK AIRCRAFT

WEIGHT SUMMARY ESTIMATE

WEIGHT EMPTY		20,148
Crew (Including Chute)		230
Survival Kit		45
Fuel - Usable		9,653
Fuel - Unusable		203
Oil - Trapped and Engine		62
Bomb (1) (Internal)		1,000
Rockets - 2.75" (24) (In doors)		432
GROSS WEIGHT - CLEAN		31,773
Add Sidewinders (2) External)	+	310
Add External Tanks - 450 Gal.	+	6,226
GROSS WEIGHT - LONG RANGE STRIKE		38,309
Remove Bomb	-	1,000
Remove Sidewinders	-	310
*Remove Fuel	-	599
Add Large Bomb	+	2,000
GROSS WEIGHT - MAXIMUM INTERNAL BOMB LOAD		38,400

GROSS WEIGHT		38,400
Remove Externals	-	6,226
Remove Large Bomb	-	2,000
Add Bombs (3)	+	3,000
**Add External Tanks - 230 Gal.	+	3,126
Add Fuel	+	599
GROSS WEIGHT - BOMBING MISSION		36,899
Remove Bombs	-	3,000
Remove Externals - 230 Gal.	-	3,126
Add Externals - 450 Gal.	+	6,226
Add Missile Bay Fuel Tank and Fuel		
WEIGHT EMPTY	+	110
USEFUL LOAD	+	975
GROSS WEIGHT - FERRY MISSION		38,084
Add Sidewinders (2)	+	310
GROSS WEIGHT - AIR PATROL MISSION		38,394

[redacted]

* Fuel limited by gross weight.

** 450 Gal. tanks filled, would cause gross weight to exceed limit.

[redacted]

PERFORMANCE

In the following pages the performance of two configurations of the F102 Strike Attack Aircraft are compared. In one configuration, the fuselage missile spaces are utilized to carry extra fuel. In the other they are used to carry Sidewinder missiles. For this comparison the latest engine data available was used. As was stated previously the drag of two sidewinders is a very small percentage of the total airplane drag. As a consequence, it makes little difference whether they are carried internally or externally on this size of aircraft. On a smaller airplane the effect of external weapon carriage could be appreciable. Also if air-to-air weapons other than sidewinders were selected for installation, the effect of hanging them externally might be noticeable.

The net effect on mission radii or endurance of carrying the sidewinders externally in this study appears to be a gain. This is due to the fact that the loading conditions are better. That is, with standard droppable aircraft tanks it is possible to load the airplane to a gross weight more nearly approaching the permissible maximum, 38400 lbs.

Referring back to the weight summary page 11, it will be noted that for most of the loading conditions the maximum gross weight (38400 lb) is closely approached. In fact in the case of the "Maximum Internal Bomb Load" and the "Bombing Mission" the fuel load had to be limited. In one case this was done by not filling the tanks. In the other it was done by utilizing 230 gal. rather than 450 gallon external tanks.

With the configuration described in the original report ZP59-08002 where missiles were carried internally rather than extra fuel, it was not necessary to limit the fuel because there was insufficient fuel capacity to overload the airplane under most conditions. On the other hand the maximum T.O. gross weight was not reached for many loading conditions because of lack of fuel capacity. Thus the mission radii or endurance were limited in some cases by fuel on board at take-off.

[redacted]

F-102 STRIKE ATTACK AIRCRAFT

PERFORMANCE SUMMARY

NO EXTERNAL STORES

			P and W JT 3C-24 Engine
1.	Maximum rated engine thrust	Lb.	18,000
2.	Take-off weight		38,309*
3.	V_{max} A/B power at 35,000 feet	Mach	1.5
4.	Time to accelerate 0.9M to 1.5M	Min.	5.0
5.	Take-off distance over 50 foot obstacle - maximum thrust	Ft.	5,800*
6.	Landing distance over 50 foot obstacle - with drag chute deployed.	Ft.	4,900
7.	Ferry range	N. Mi.	Over 2,000*
8.	Load factor at 25,500 lb. gross	-	7.0

* With 450 gal. tanks external.

MISSION COMPARISON

	F-102 Sidewinders Carried Externally - Provides More Fuel	F-102 Sidewinders Carried Internally - Provides More Armament
Air Patrol Mission	(See curve page)	
High-Low-High Strike Mission - 100 N. Mi. Run in	630	495
High-Low-High Strike Mission - 50 N. Mi. Run in	755	615
Low-Low-Low Strike Mission - 100 N. Mi. Run in	400	345
Low-Low-Low Strike Mission - 50 N. Mi. Run in	418	360

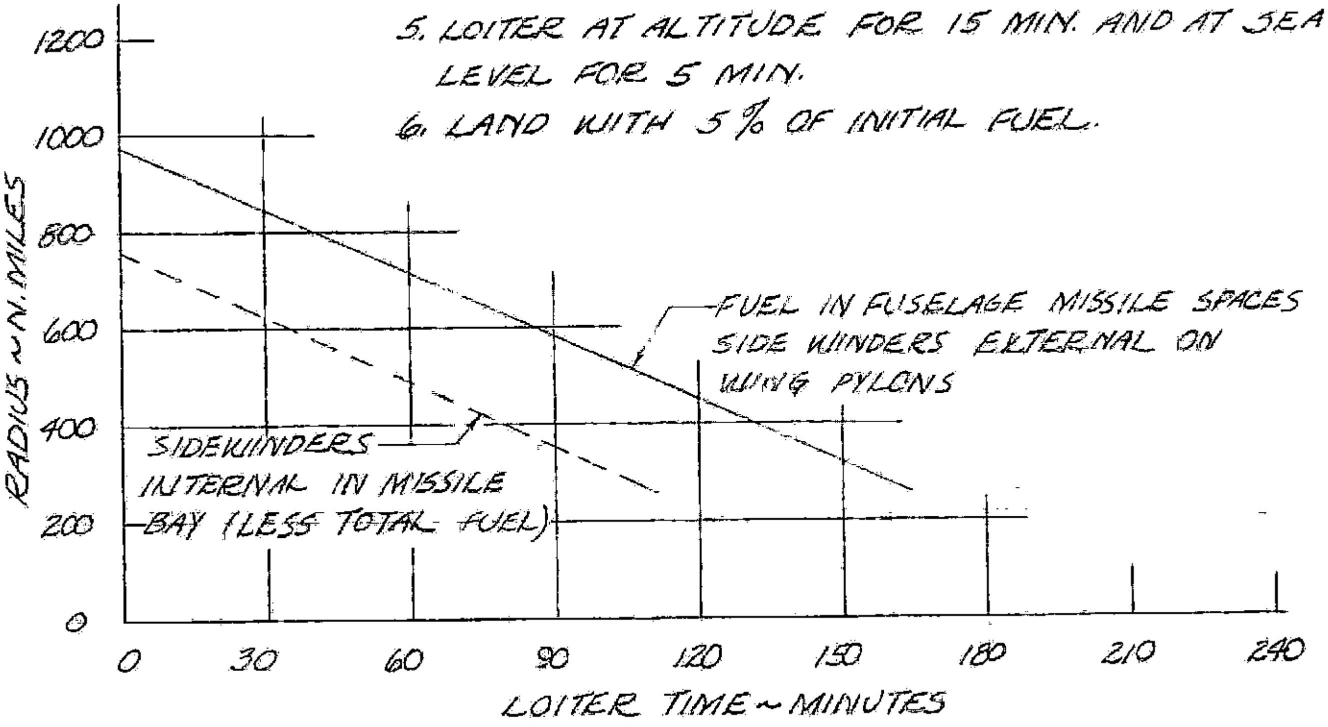
[redacted]

F-102 STRIKE ATTACK AIRCRAFT

AIR PATROL MISSION

[redacted]

1. TAKE-OFF, CLIMB TO 35,000 FEET AND CRUISE OUT.
2. LOITER, THEN ACCELERATE AND CLIMB TO 50,000 FEET.
3. COMBAT ALLOWANCE OF 5 MIN. AT V_{MAX}.
4. RETURN HOME AT OPTIMUM SPEED AND ALTITUDE.
5. LOITER AT ALTITUDE FOR 15 MIN. AND AT SEA LEVEL FOR 5 MIN.
6. LAND WITH 5% OF INITIAL FUEL.



[redacted]

F-102A STRIKE ATTACK AIRCRAFT

High-Low-High Strike Mission

100 Nautical Mile Run-in and Run-out

1. Warm-up, taxi and T. O.
2. Climb to optimum cruise alt. (elapsed time since T. O. = 0.6 hr.).
3. Cruise out at optimum alt., M = 0.90.
4. Descend to S. L. (No distance credit, no fuel used).
5. 100 N. Mi. run-in, (M = 0.85) and drop tanks as indicated, combat (V_{max}, Mil. power) and run-out (M = 0.85) (elapsed time since T. O. before combat = 1.3 hr.)
6. Climb to optimum cruise alt.
7. Cruise back, M = 0.90.
8. Loiter for 15 min. at alt., M = 0.87.
9. Descend to S. L. (No distance credit, no fuel used).
10. Loiter for 5 min. at S. L., M = 0.3.
11. Land with reserve of 5% initial fuel load (total mission time = 2.8 hr.).

50 Nautical Mile Run-in and Run-out

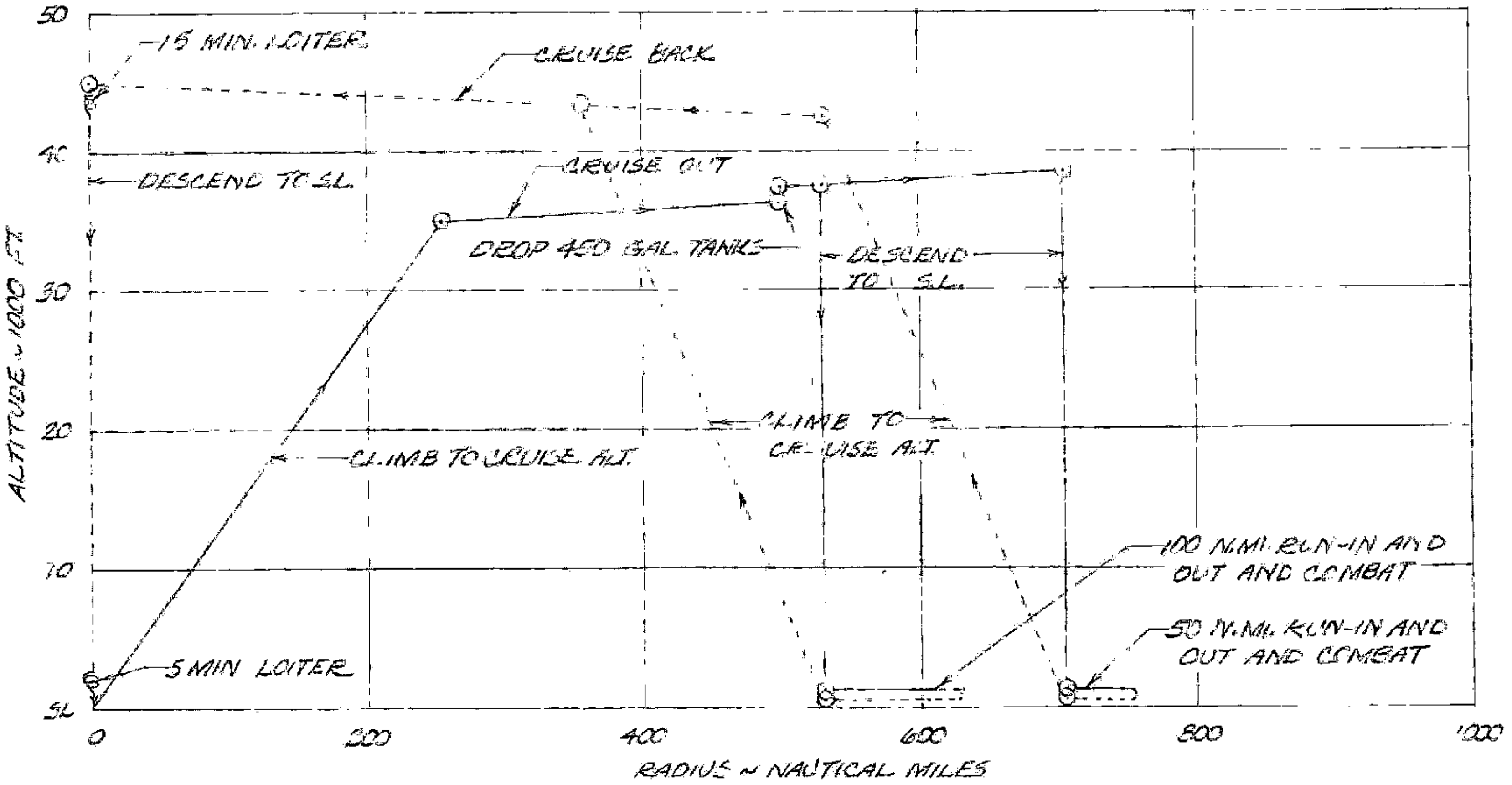
The radius with a 50 N. Mi. run-in and out is 755 N. Mi., elapsed time from T. O. to combat is 1.5 hr. and total mission time is 3.5 hr.

[redacted]

F-102 STRIKE ATTACK AIRCRAFT

HIGH-LOW-HIGH STRIKE MISSION

2-450 GALLON EXTERNAL FUEL TANKS AND 2-SIDEWINDERS EXTERNAL
1-1000 LB BOMB AND 34-2.75 IN. ROCKETS INTERNAL

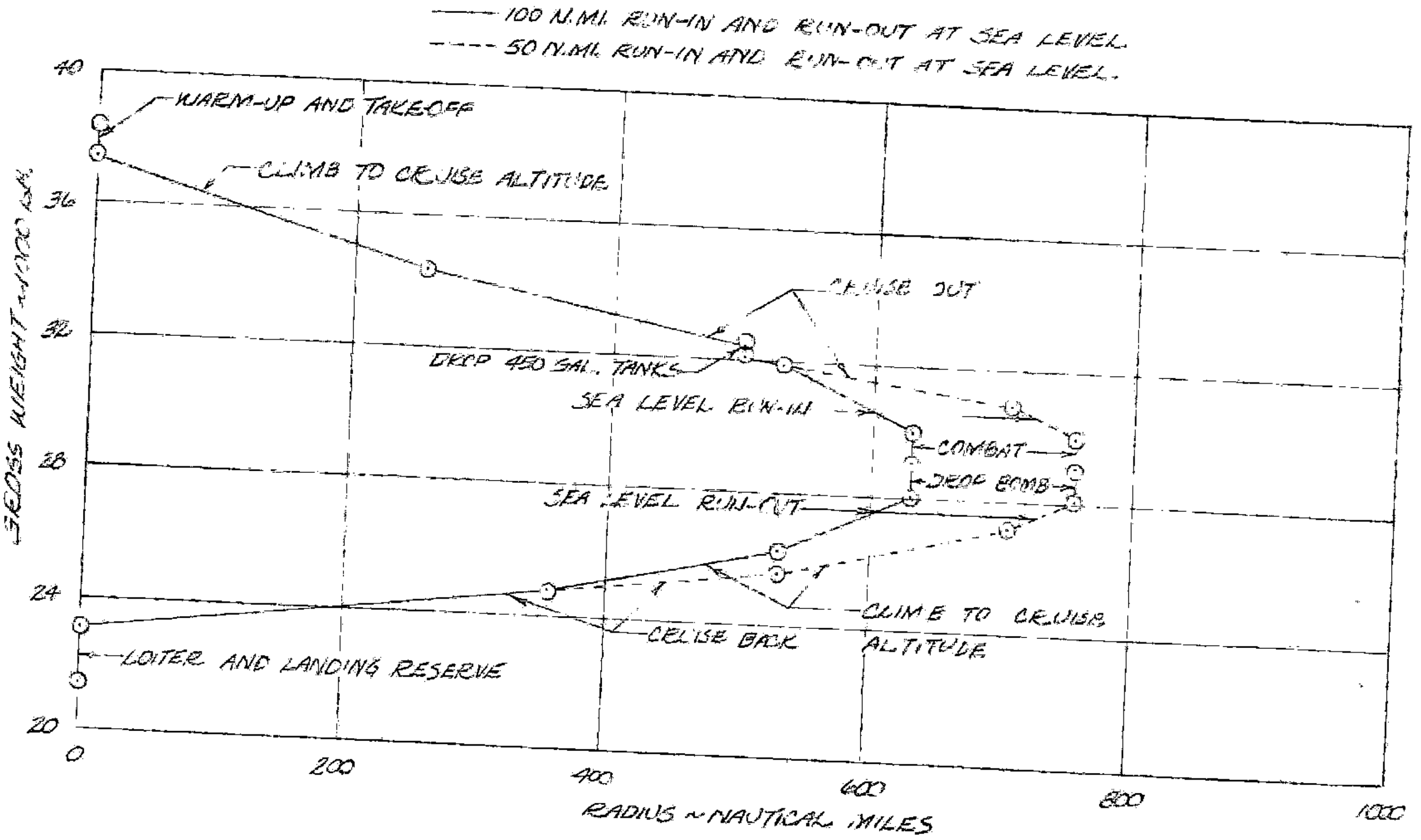


[redacted]

F-102 STRIKE ATTACK AIRCRAFT

HIGH-LOW-HIGH STRIKE MISSION

2 - 450 GALLON EXTERNAL FUEL TANKS AND 2-SIDEWINDERS EXTERNAL
1-1000 LB. BOMB AND 24-2.75 IN. ROCKETS INTERNAL



[redacted]

F-102A STRIKE ATTACK AIRCRAFT

Low-Low-Low Strike Mission

100 Nautical Mile Run-in and Run-out

1. Warm-up, taxi, T. O. and accelerate to cruise speed.
2. Cruise out at S. L. and drop tanks as indicated, M = 0.45.
3. Accelerate to Mach number = 0.75.
4. 100 N. Mi. run-in (M = 0.75), drop tanks as indicated, and run out (M = 0.85). Elapsed time from T. O. to combat = 1.2 hour.
5. Cruise back, M = 0.85.
6. Loiter for 15 minutes at S. L., M = 0.3 .
7. Land with reserve of 5% initial fuel load (total mission time = 2.6 hour.)

50 Nautical Mile - Run-in and Run-out

The radius with a 50 N. Mi. run-in and out is 418 N. Mi.; elapsed time from T. O. to combat is 1.3 hr. and total mission time is 2.8 hr.

[redacted]

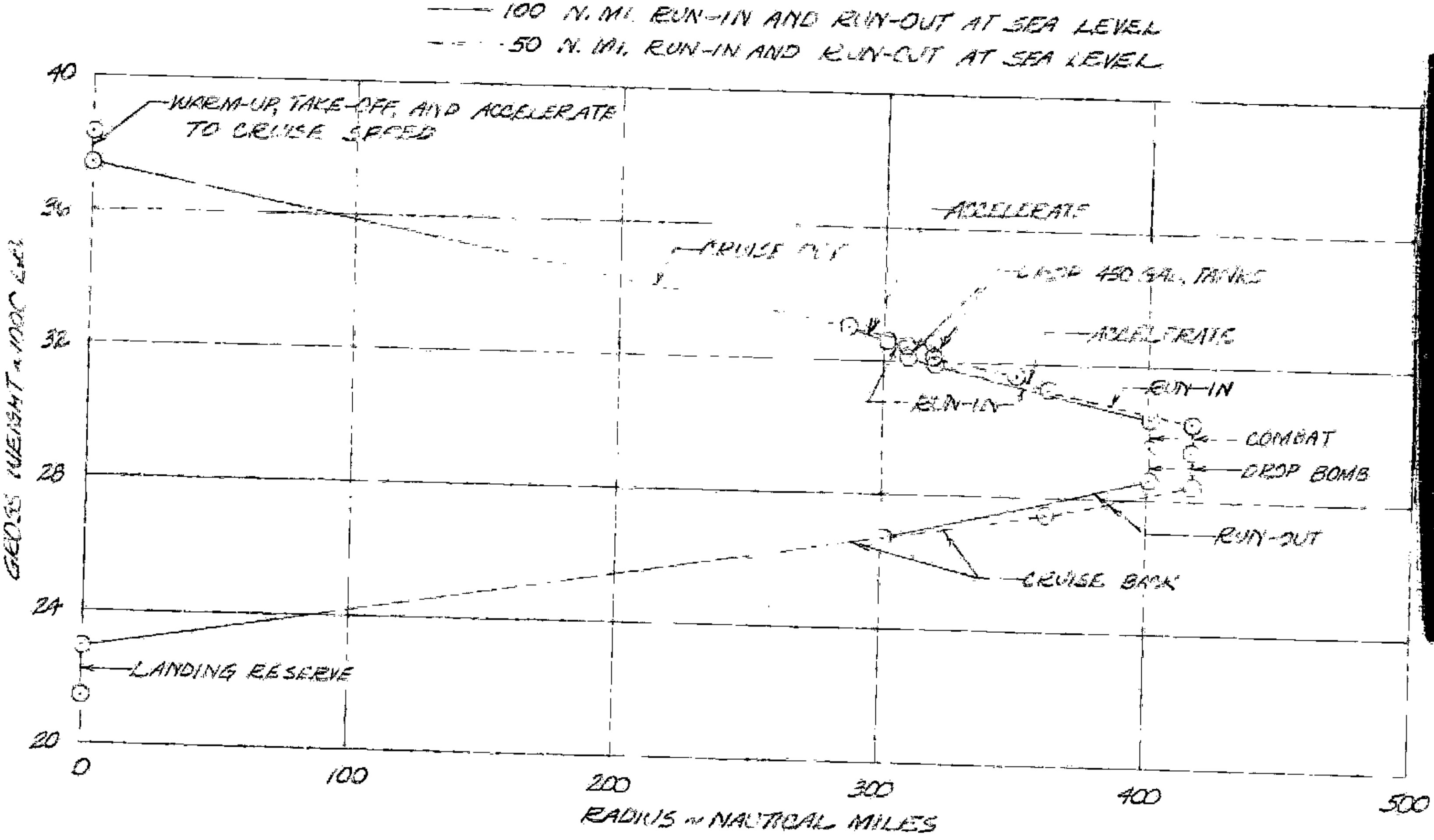
F-102 STRIKE ATTACK AIRCRAFT

LOW-LOW-LOW STRIKE MISSION

2-450 GALLON EXTERNAL FUEL TANKS AND 2-SIDEWINDERS EXTERNAL
1-1000 LB. BOMB AND 24-2.75 IN. ROCKETS INTERNAL

[redacted]

[redacted]



[redacted]

SECTION II - PERFORMANCE BASIS

Except for the effects of changes in basic engine thrust (JT3C-24 vs. J57-P-23) and the use of wing leading edge slots in lieu of wing fences, the performance basis for the Strike Attack aircraft is the same as currently used for the production F-102A. This basis has been substantiated by flight results obtained in the USAF Phase IV performance evaluation of the F-102A airplane. * The agreement between predictions and flight test results is demonstrated in the plots of high speed and rate of climb versus altitude on page 22. (A comprehensive comparison between prediction and flight test is given in reference 4) 22

The only area of performance in which the prediction method used herein does not amply represent the capability of the final production F-102A configuration (Case XX wing) is in the high altitude regime where the flight data indicate a ceiling increase of between 2,500 feet and 4,500 feet. ** A ceiling increase of 2,500 feet, over and above that calculated by the prediction method, has thereby been included in the performance quotes of the Strike Attack aircraft. (See page 23.) 23

Drag Basis

Drag polar curves for the basic F-102A configuration, based upon predictions as noted above and developed in reference 5, are shown on page 24. Drag increments are shown on page 25 for the several configuration changes incorporated into the Strike Attack aircraft. These are listed below. 24 25

a. Removal of wing fences and addition of wing leading edge slots. (Drag change determined from flight tests of the F-106A.)
b. Use of 450 gallon external tanks. (Since tank is similar in shape to the 230 gallon tank tested on the F-102A the drag coefficient increment was increased by the frontal area ratio of the tanks.)

* Airplane with and without two 230 gallon external fuel tanks (References 1, 2, and 3).

** Basic prediction method was developed originally for the Case X wing. The performance improvement results from the change to the Case XX wing (change in wing camber from 6.34% semispan to 15% semispan and change in design lift coefficient from 0.166 to 0.3).

[redacted]

c. Addition of external Sidewinder missiles (drag increment based on wind tunnel tests of a similar installation under wing of F-106A model.)

Installed Thrust and Fuel Flow

The basic thrust and fuel flow data of the JT3C-24 engine were obtained from P and W Spec. T-1435, dated 6 January 1959. Installation losses were derived in the same manner as developed for the basic F-102A (J57-P-23 engine) in reference 6. Thrust and fuel flow data with installation losses taken into account are shown on pages 26 through 33.

26-33

References:

1. USAF Air Force Flight Test Center Report AFFTC-TR-56-23, "F-102A Phase IV Performance Test" January 1957.

2. Addendum I to AFFTC Report AFFTC-TR-56-23, "F-102A Phase IV Performance Test with Two Standard Class II 230 gallon Pylon Tanks" July 1957.

3. USAF Air Force Flight Test Center Report AFFTC-TC-TR-57-25, "F-102A Case XX Wing Phase IV Performance" November 1957.

4. Convair Aerodynamics Memorandum A-8-249, "Analysis of F-102A Phase IV Performance Test Results", Revised July 1958.

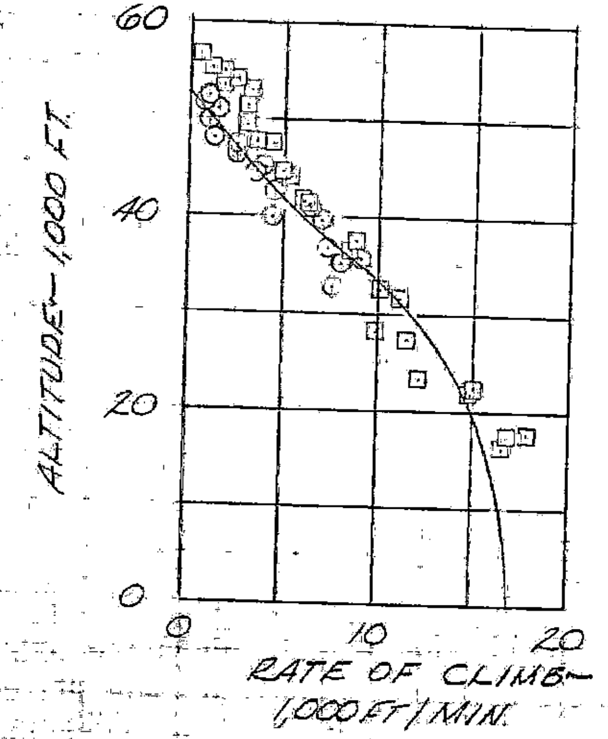
5. Convair Report ZA-8-012," Performance Data Report for the F-102A Flight Handbook Appendix I", Revised October 1958.

6. Convair Report ZJ-8-012, "Installed Thrust Available, Fuel Flow and Engine Air Inlet Performances of the F-102A and TF-102A airplane" May 1955.

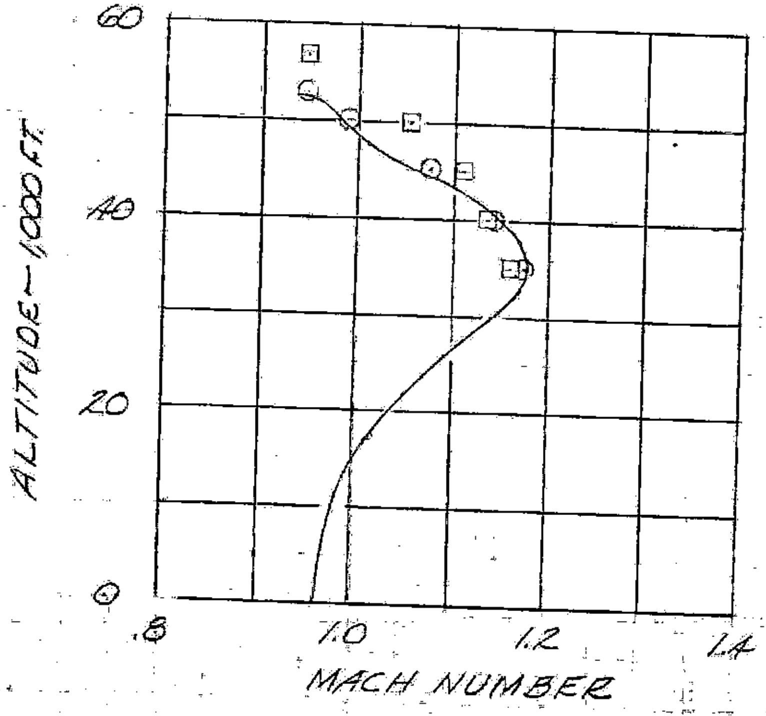
[redacted]

F-102A PERFORMANCE COMPARISON
FLIGHT TEST VS PREDICTIONS

——— F-102A PREDICTION
- ⊙ F-102A PHASE IV FLT. TEST, CASE X WING
- ⊡ F-102A PHASE IV FLT. TEST, CASE XX WING (FINAL PRODUCTION CONFIGURATION)



CLIMB CHARACTERISTICS

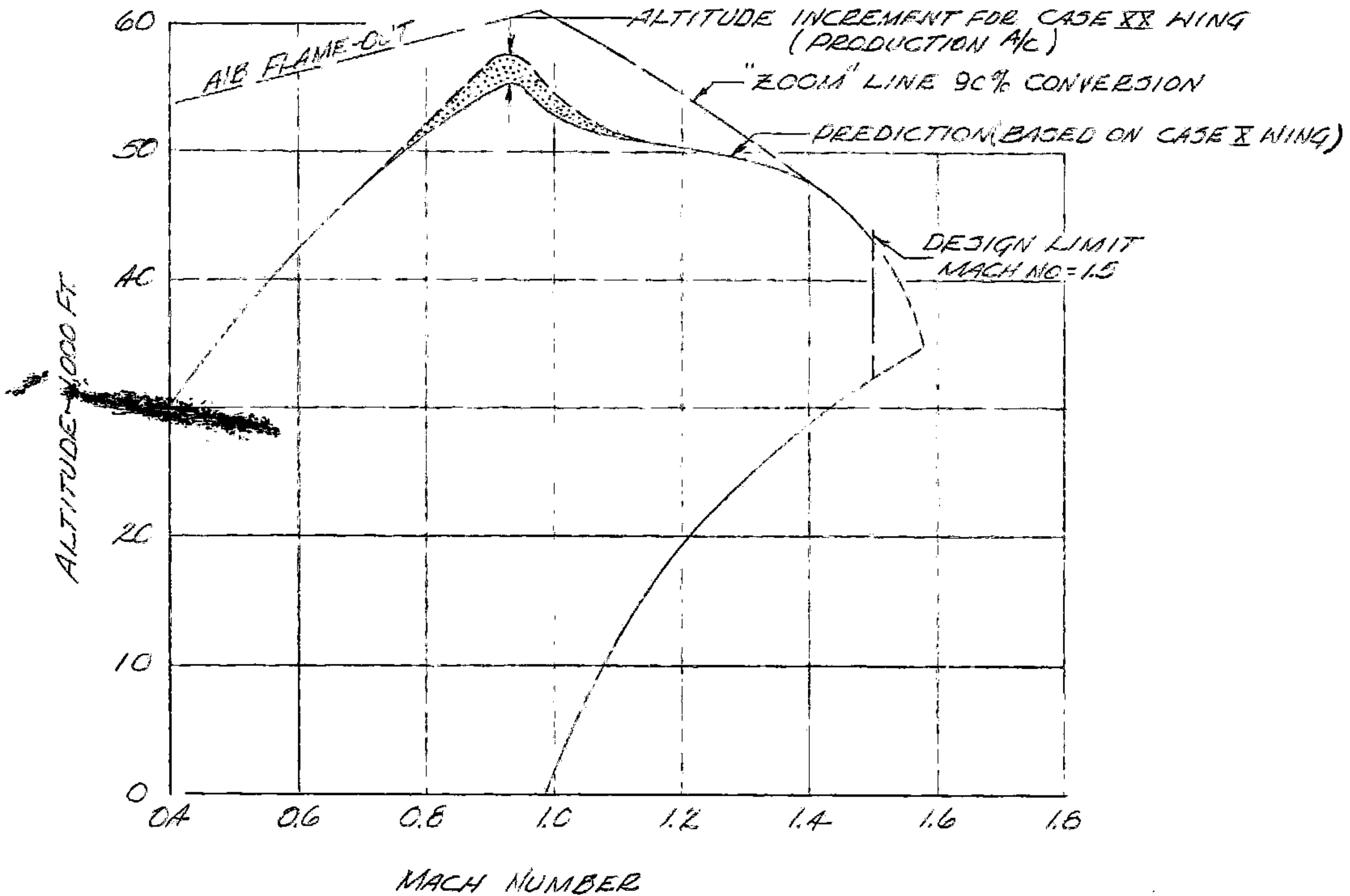


HIGH SPEED CHARACTERISTICS

[redacted]

F-102 STRIKE ATTACK AIRCRAFT

ALTITUDE-SPEED ENVELOPE



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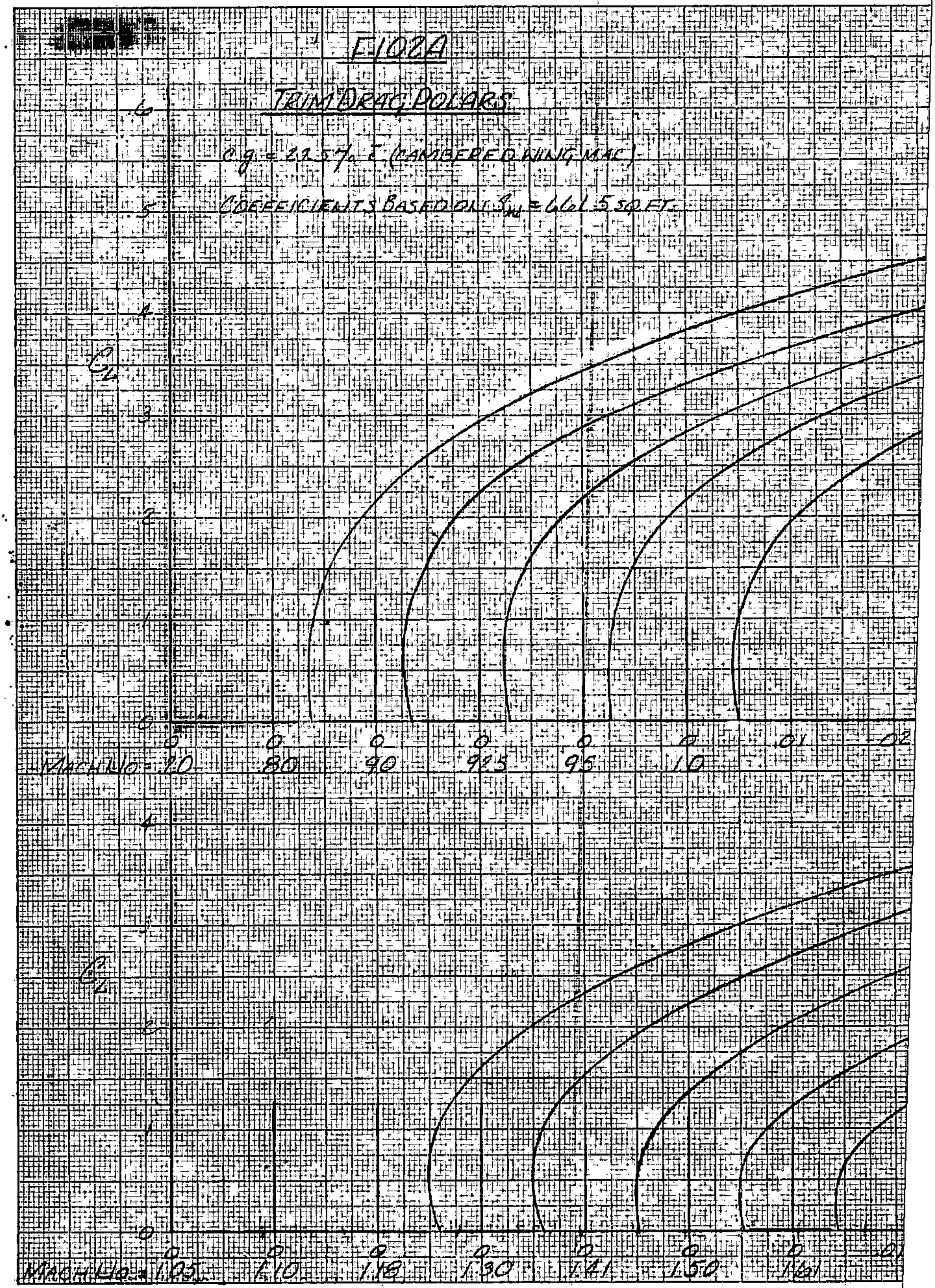


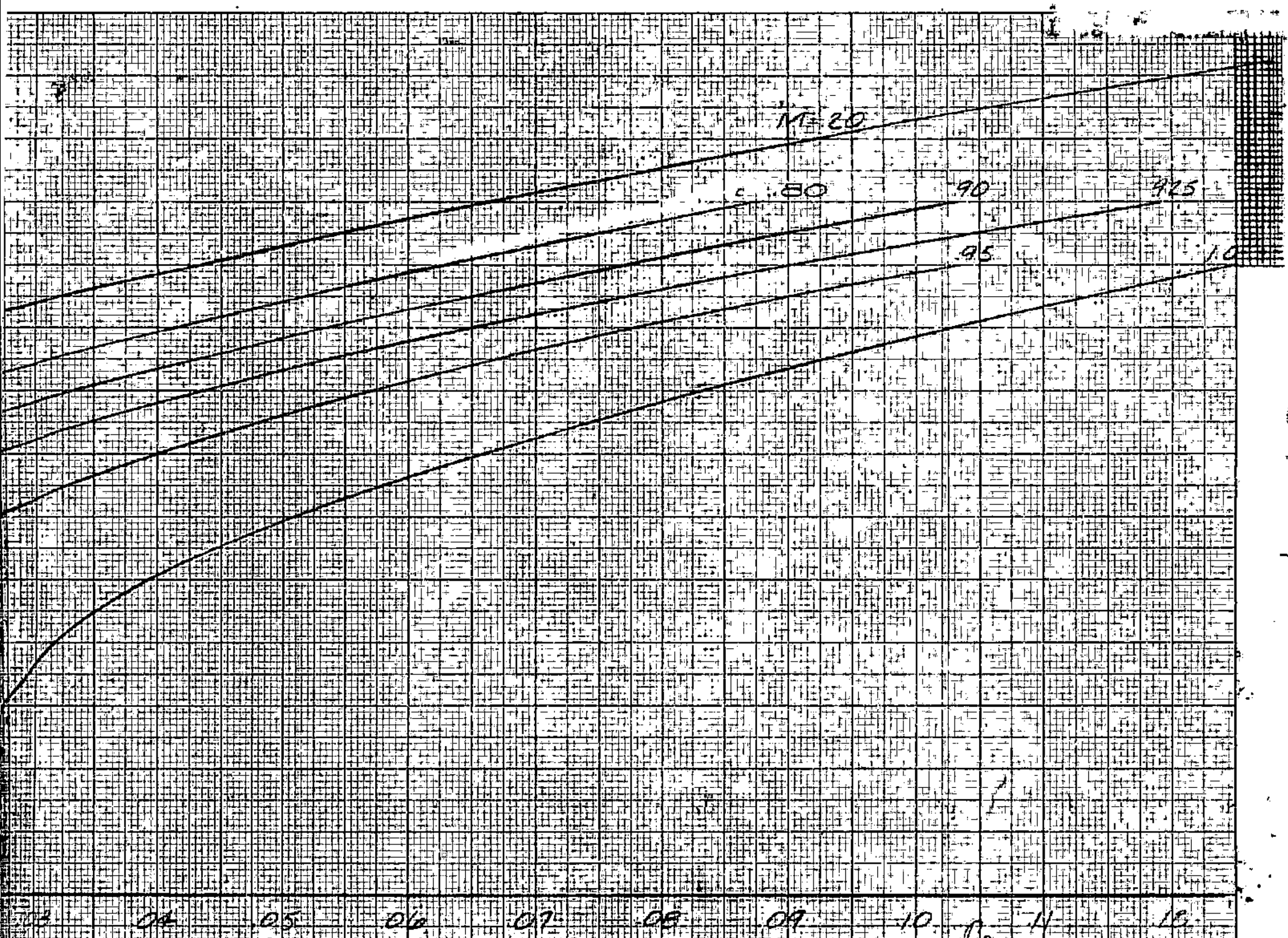
F-102A

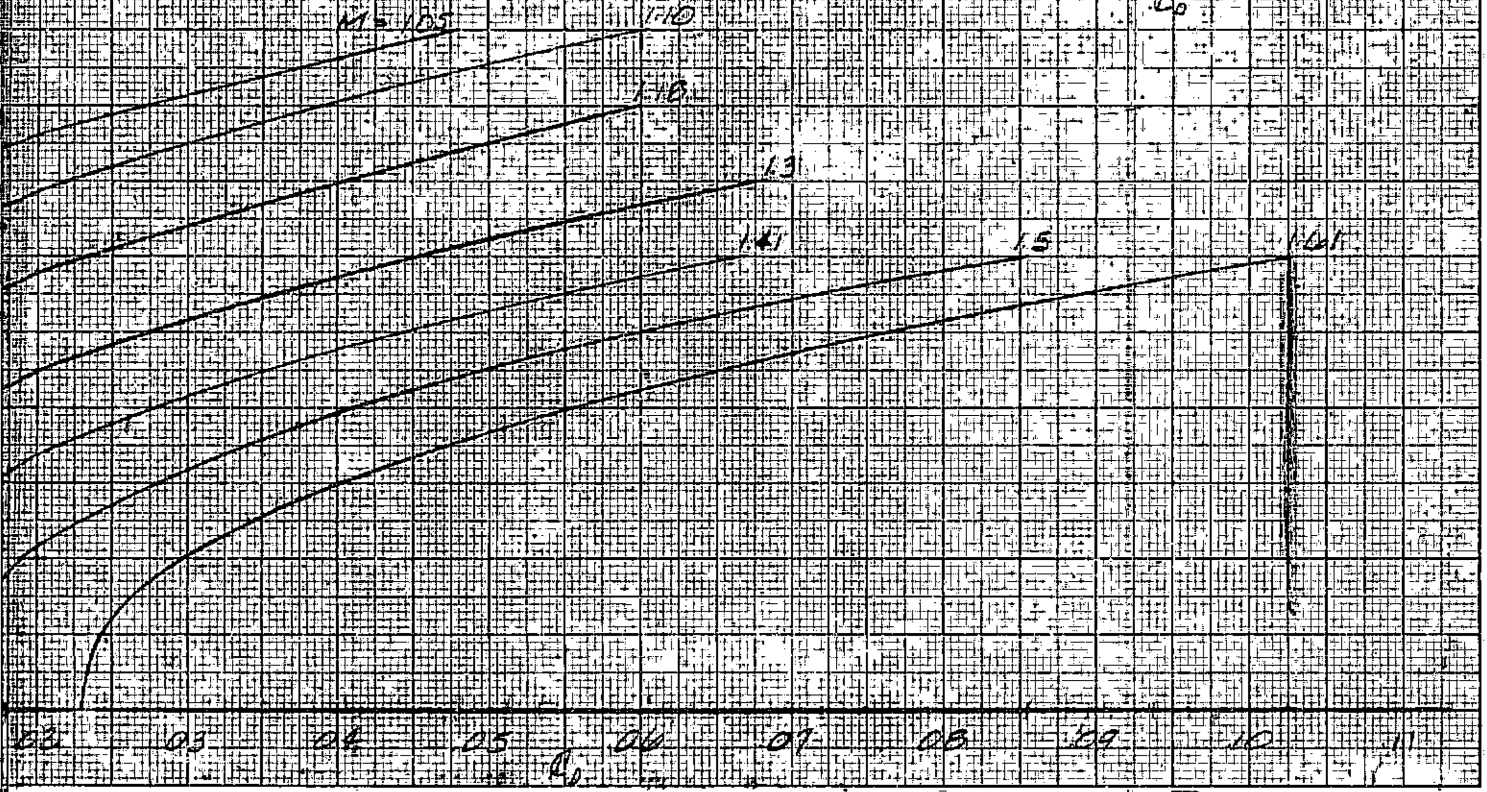
TRIM DRAG POLARS

c.g. = 22.5% c̄ (CAMBERED WING MAC)

COEFFICIENTS BASED ON S_W = 661.5 SQ.FT.

K+E 10 X 10 TO THE CM. 359T-14LG
KEUFFEL & ESSER CO. MADE IN U.S.A.
ALBANENE ®

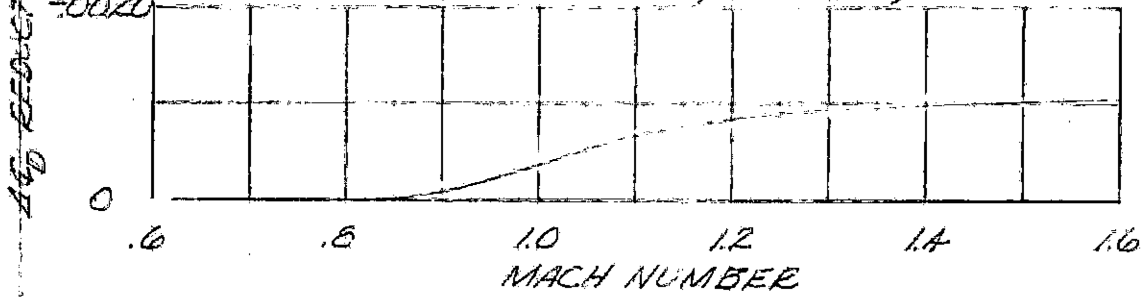

M=.20
.80
.90
.925
.95
1.0
.03
.04
.05
.06
.07
.08
.09
.10
C_D
.11
.12


M=1.05
1.10
1.16
1.3
1.41
1.5
1.61
.02
.03
.04
.05
C_D
.06
.07
.08
.09
.10
.11

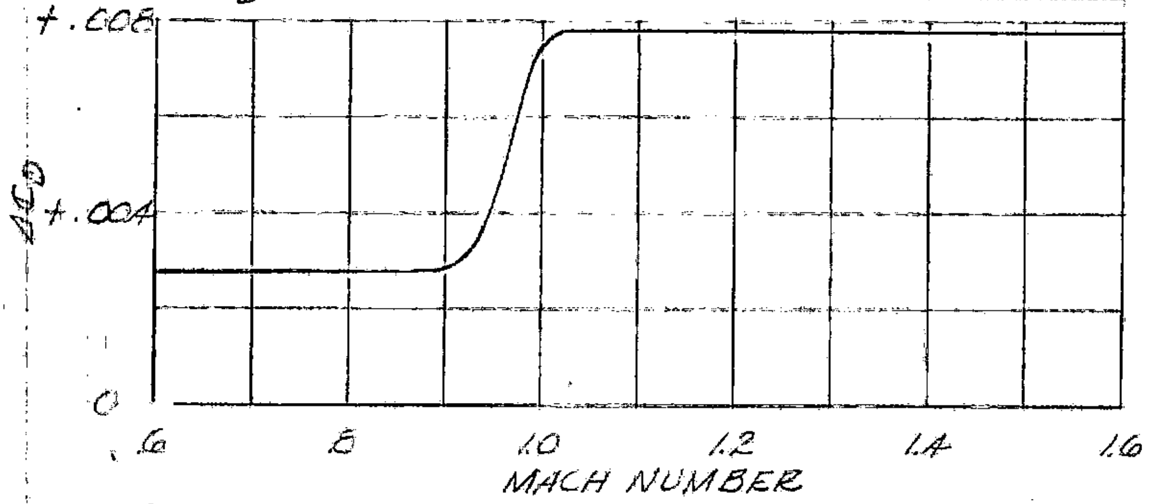
[redacted]

F-102 STRIKE ATTACK AIRCRAFT

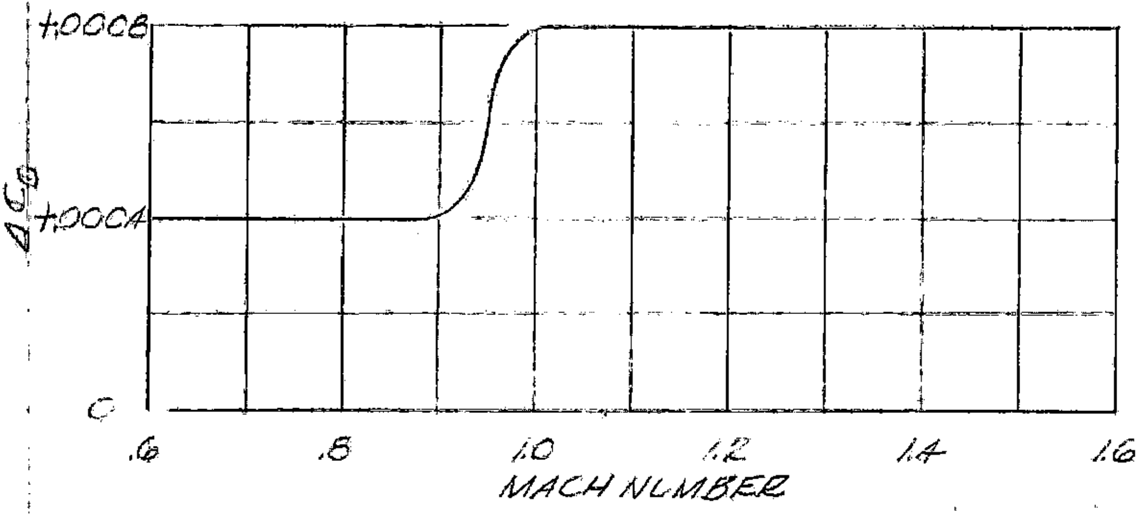
ΔC_D REDUCTION FOR WING LEADING EDGE SLOTS
(IN LIEU OF WING FENCES)



ΔC_D FOR TWO-450 GALLON EXTERNAL FUEL TANKS



ΔC_D FOR TWO SIDEWINDERS MOUNTED UNDER WING



[redacted]

NET THRUST AVAILABLE VS MACH NUMBER

AIRPLANE - F-102

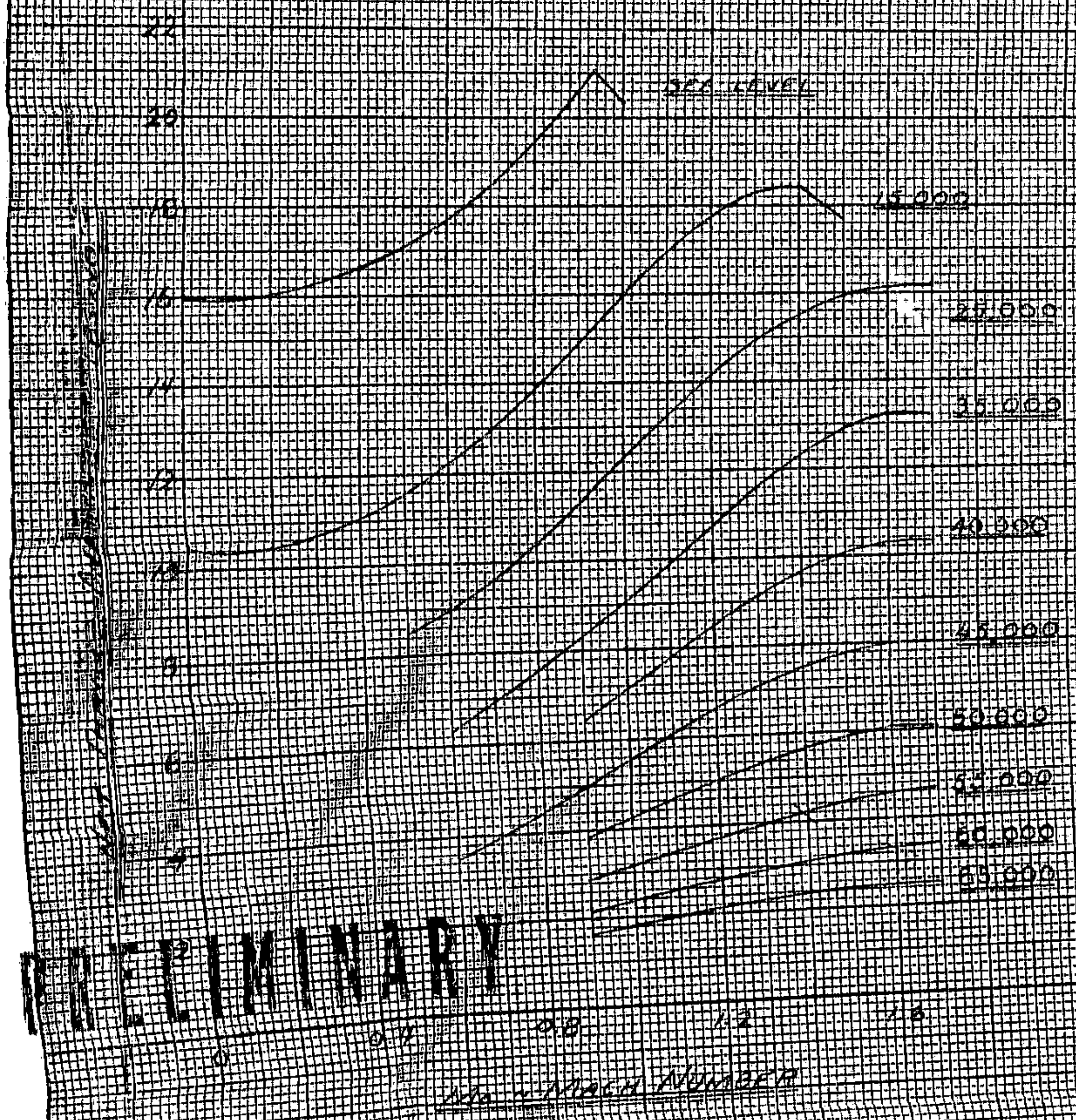
ENGINE - J75C-24

P&W DATA T-1435, 1-13
DATED 1-6-59

INSTALLATION DATA
CSD REPORT ZJ-8-012
DATED 5-1-55

STANDARD ATMOSPHERIC CONDITIONS

MAXIMUM AFTERBURNING POWER



PRELIMINARY

6-19-59 ABD

[redacted]

NET THRUST AVAILABLE VS MACH NUMBER

AIRPLANE ~ F-102

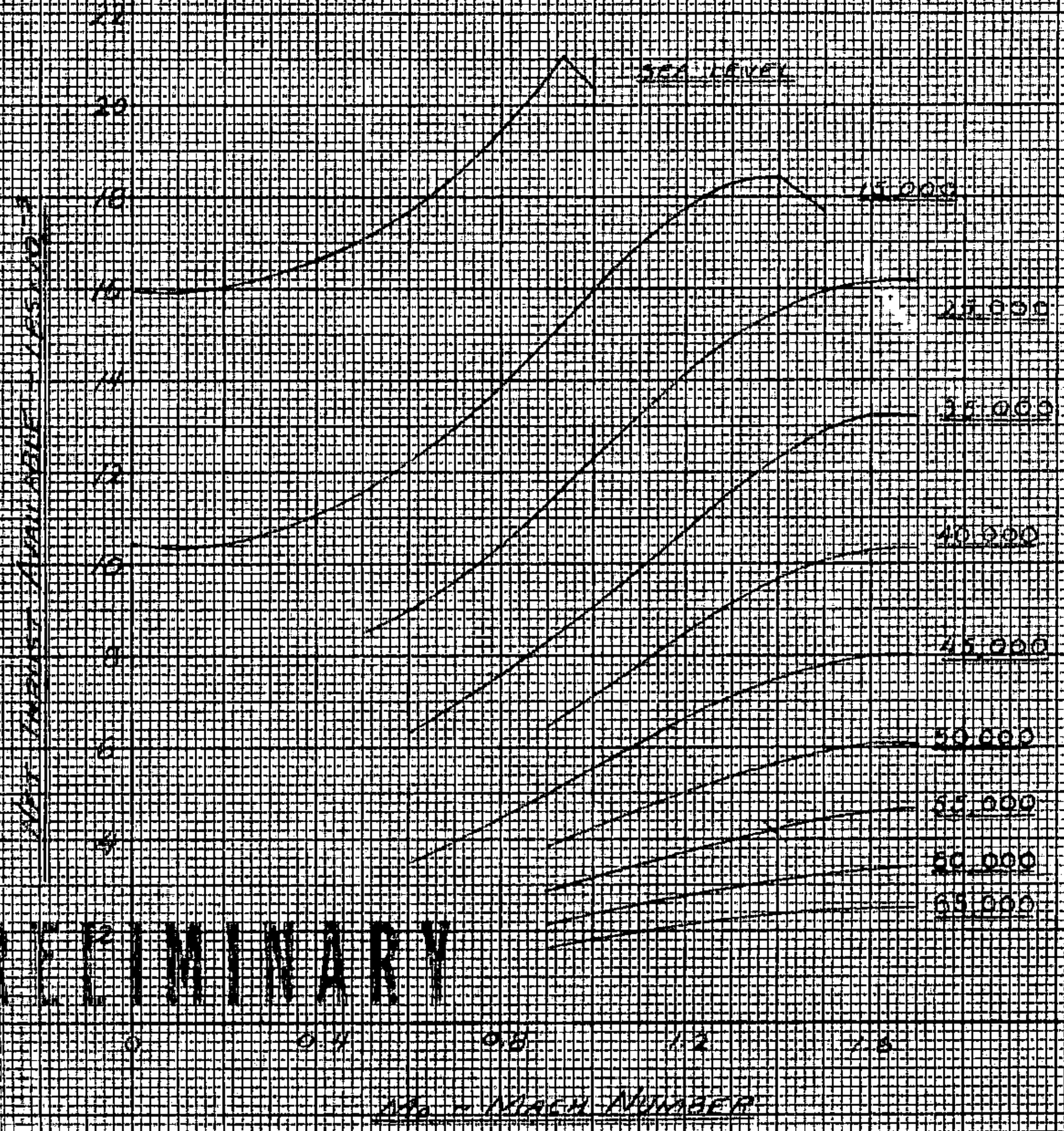
P&W DATA: T-1435, 1-13
DATED 1-6-59

ENGINE ~ JT3C-24

INSTALLATION DATA
OSD REPORT ZJ-8-012
DATED 5-4-55

STANDARD ATMOSPHERIC CONDITIONS

MAXIMUM AFTERBURNING POWER



PRELIMINARY

[redacted]

2-10-59 R.D.

K&E KEUFFEL & ESSER CO. ALBANENE ® 10 X 10 TO THE 1/2 INCH 359T-11G MADE IN U.S.A.

[redacted]

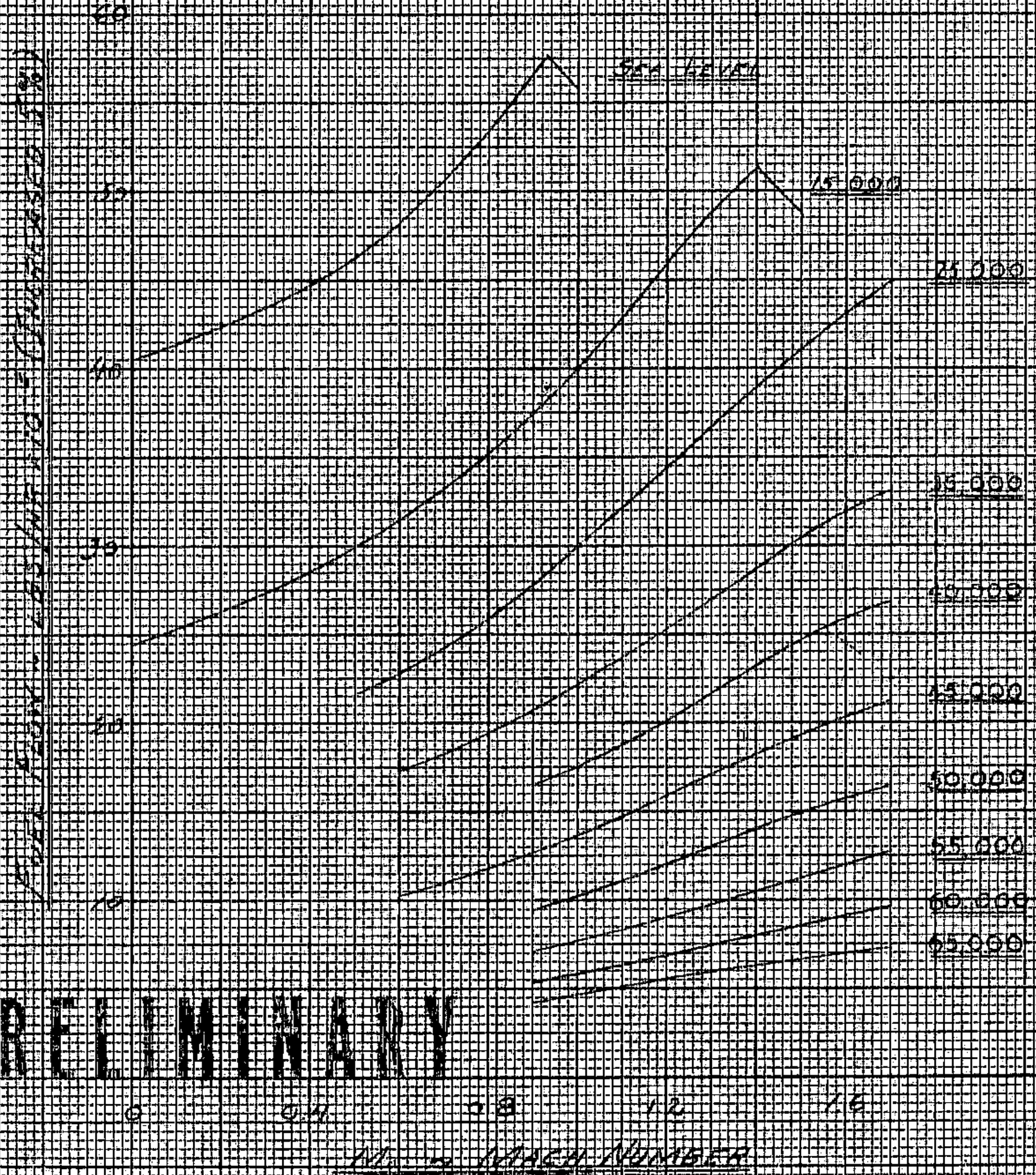
FUEL FLOW VS MACH NUMBER

AIRPLANE - F-102
ENGINE - JT3C-24

P&W DATA: T-1435.1-13
DATED 1-6-59
INSTALLATION DATA:
CSD REPORT ZJ-8-012
DATED 5-4-55

STANDARD ATMOSPHERIC CONDITIONS

MAXIMUM AFTERBURNING POWER



PRELIMINARY

[redacted]

6-10-59 RD

K+E KEUFFEL & ESSER CO. 10 X 10 TO THE ½ INCH ALBANENE® 359T-11G

[redacted]

NET THRUST AVAILABLE VS MACH NUMBER

AIRPLANE - F-102

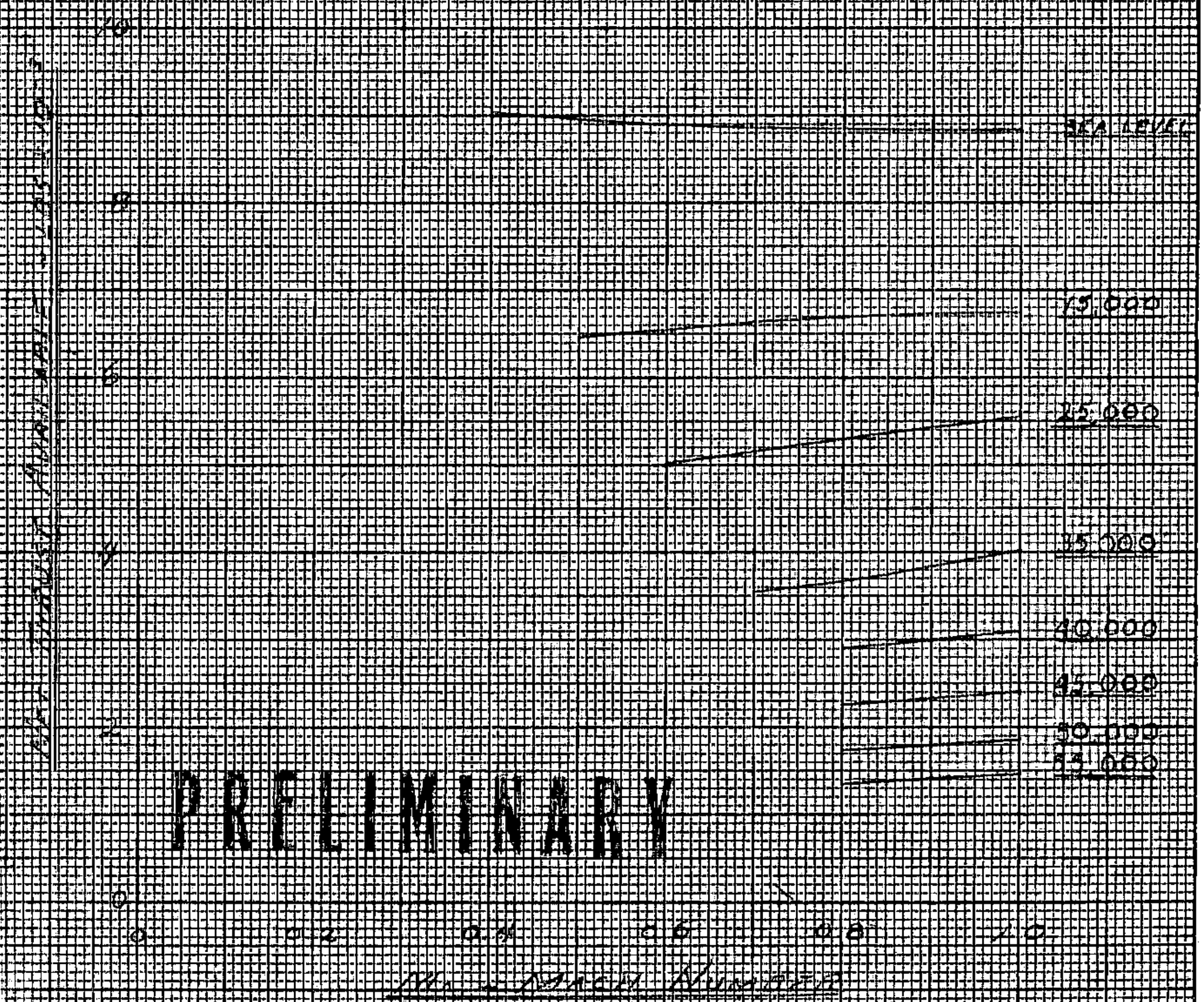
P-W DATA T-1435, 1-13
DATED 1-6-58

ENGINE - JT3C-24

STANDARD ATMOSPHERIC CONDITIONS

MILITARY POWER

INSTALLATION DATA
CSD REPORT ZU-8-012
DATED 5-4-55



PRELIMINARY

[redacted]

6-11-58 R.O.

[redacted]

FUEL FLOW VS MACH NUMBER

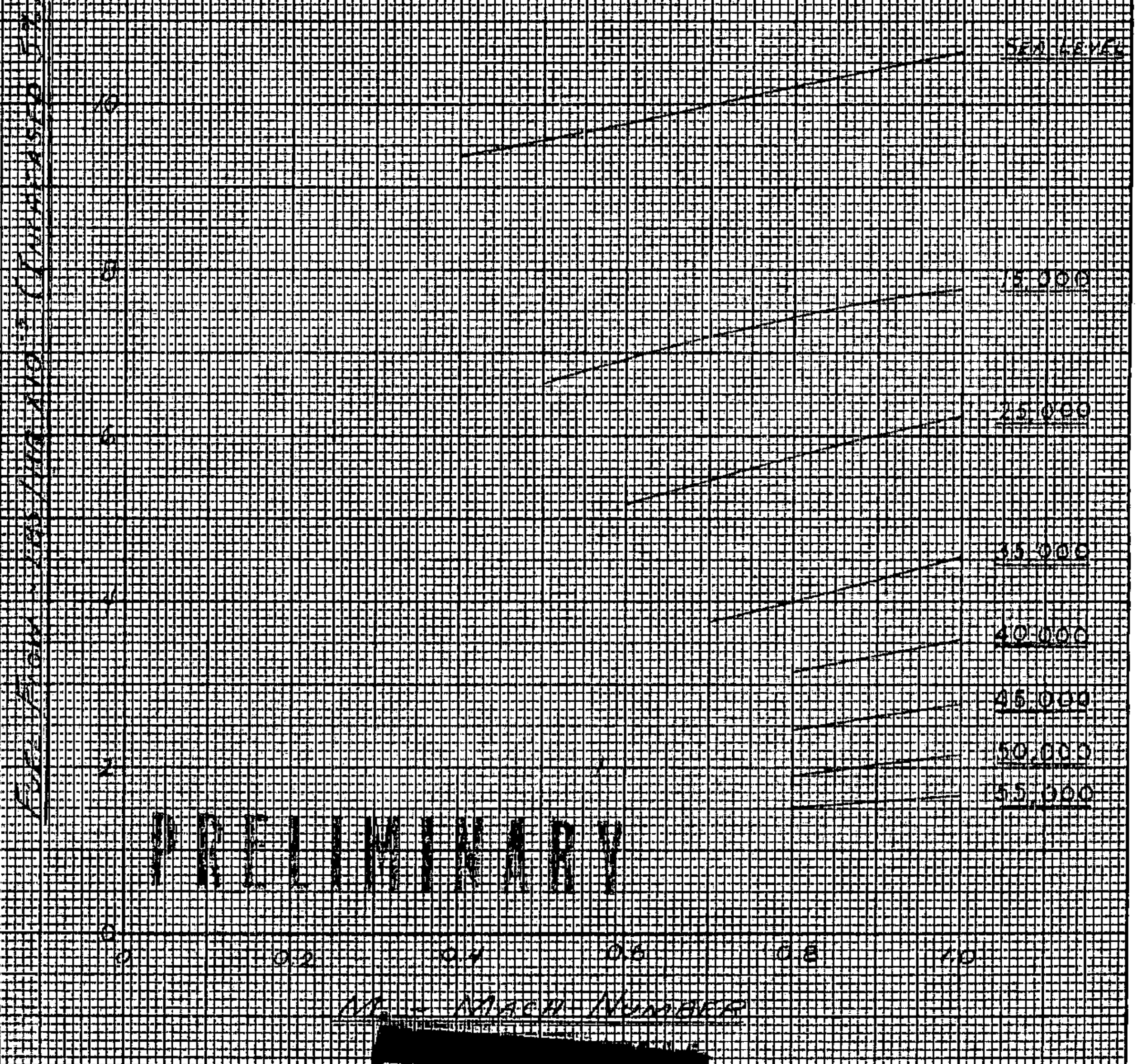
AIRPLANE - F-102
P&W DATA T-1435, 1-13
DATED 1-6-59

ENGINE - JT3C-24

STANDARD ATMOSPHERIC CONDITION

MILITARY POWER

INSTALLATION DATA
CSD REPORT ZJ-8-012
DATED 5-4-55



PRELIMINARY

[redacted]

5-11-59 R.D.

K+E 10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO.
ALBANENE
MADE IN U.S.A.

[redacted]

SPECIFIC FUEL CONSUMPTION
VS
PERCENT OF MILITARY THRUST

AIRPLANE - F-102
ENGINE - JT3C-24

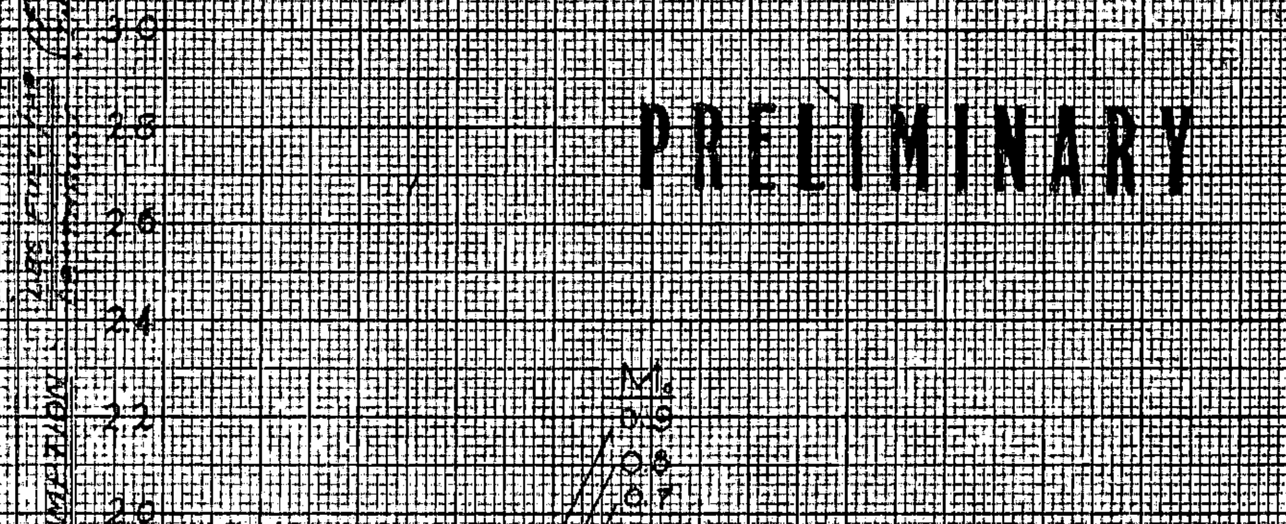
P&W DATA - T-1435 1-13
DATED 1-6-50

STANDARD ATMOSPHERIC CONDITIONS

ALTITUDE ~ SEA LEVEL

INSTALLATION DATA
CSD REPORT ZJ-8-012
DATED 5-4-55

PRELIMINARY



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K+E KEUFFEL & ESSER CO. ALBANENE ® 10 X 10 TO THE ½ INCH 359T-11G

[redacted]

SPECIFIC FUEL CONSUMPTION
VS
PERCENT OF MILITARY THRUST

AIRPLANE - F102
ENGINE - JT3C-24

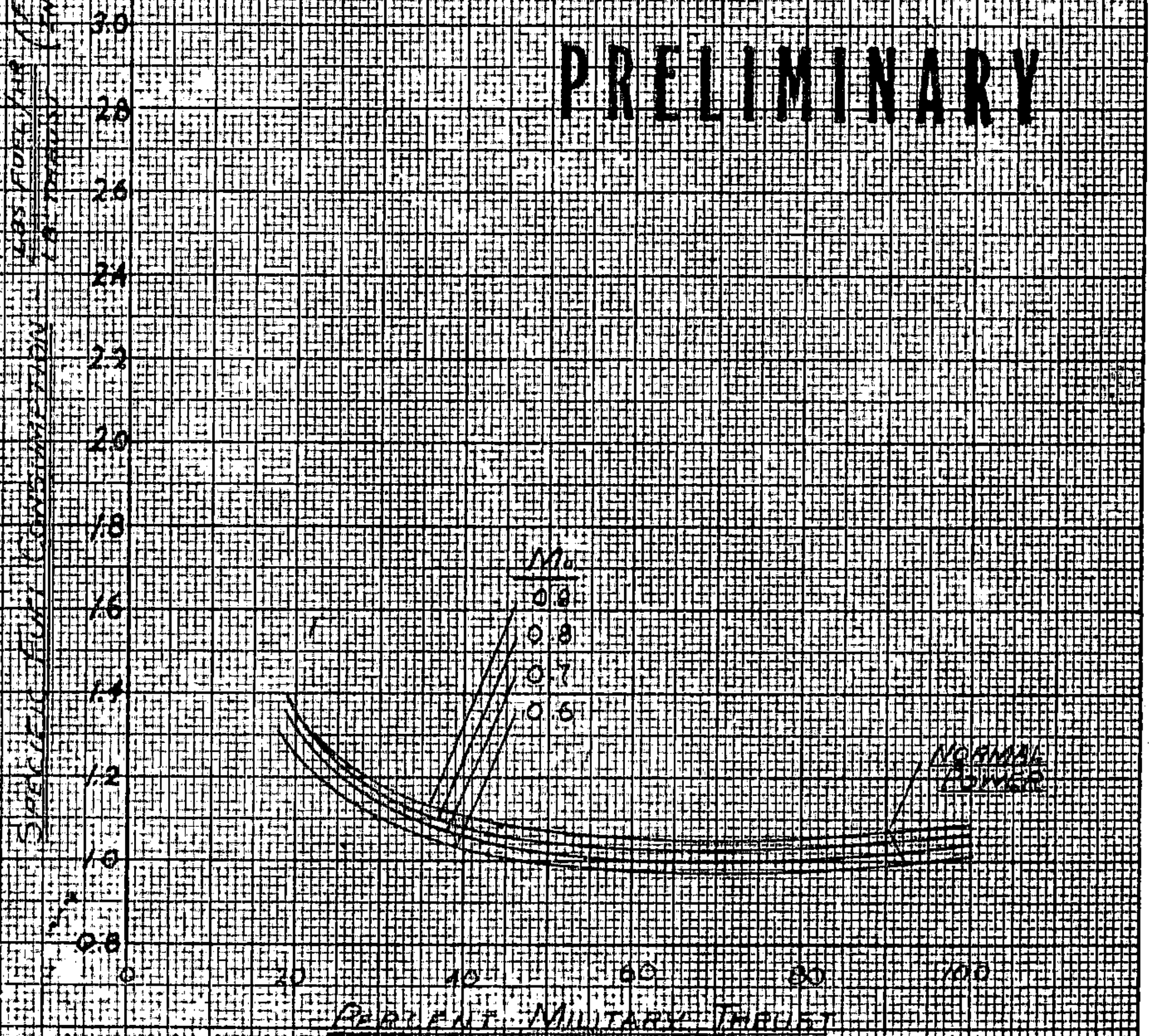
P-W DATA: T-1435 1-13
DATED 1-6-59

STANDARD ATMOSPHERIC CONDITIONS

ALTITUDE - 35,000 FT.

INSTALLATION DATA
OSD REPORT ZU-8-012
DATED 5-4-55

PRELIMINARY



PERCENT MILITARY THRUST

[redacted]

6-12-59 RO.

K&E 10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO.

[redacted]

SPECIFIC FUEL CONSUMPTION
VS
PERCENT OF MILITARY THRUST

AIRPLANE - F-102
ENGINE ~ JT3C-24

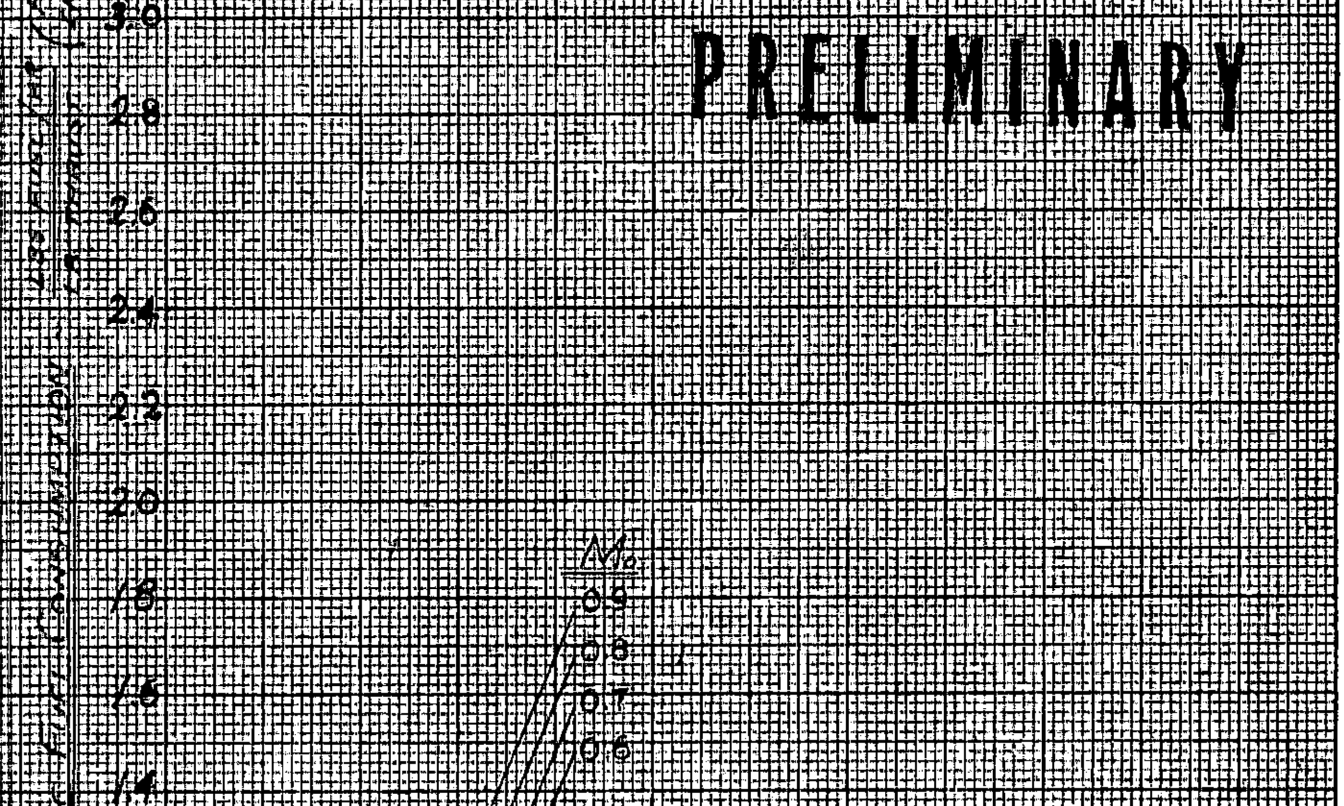
P+W DATA: T-1435, 1-13
DATED 1-6-59

STANDARD ATMOSPHERIC CONDITIONS

ALTITUDE ~ 40,000 FT.

INSTALLATION DATA:
CSD REPORT ZJ-8-012
DATED 5-4-55

PRELIMINARY



PERCENT MILITARY THRUST

[redacted]

6-12-59

[redacted]

SPECIFIC FUEL CONSUMPTION
VS
PERCENT OF MILITARY THRUST

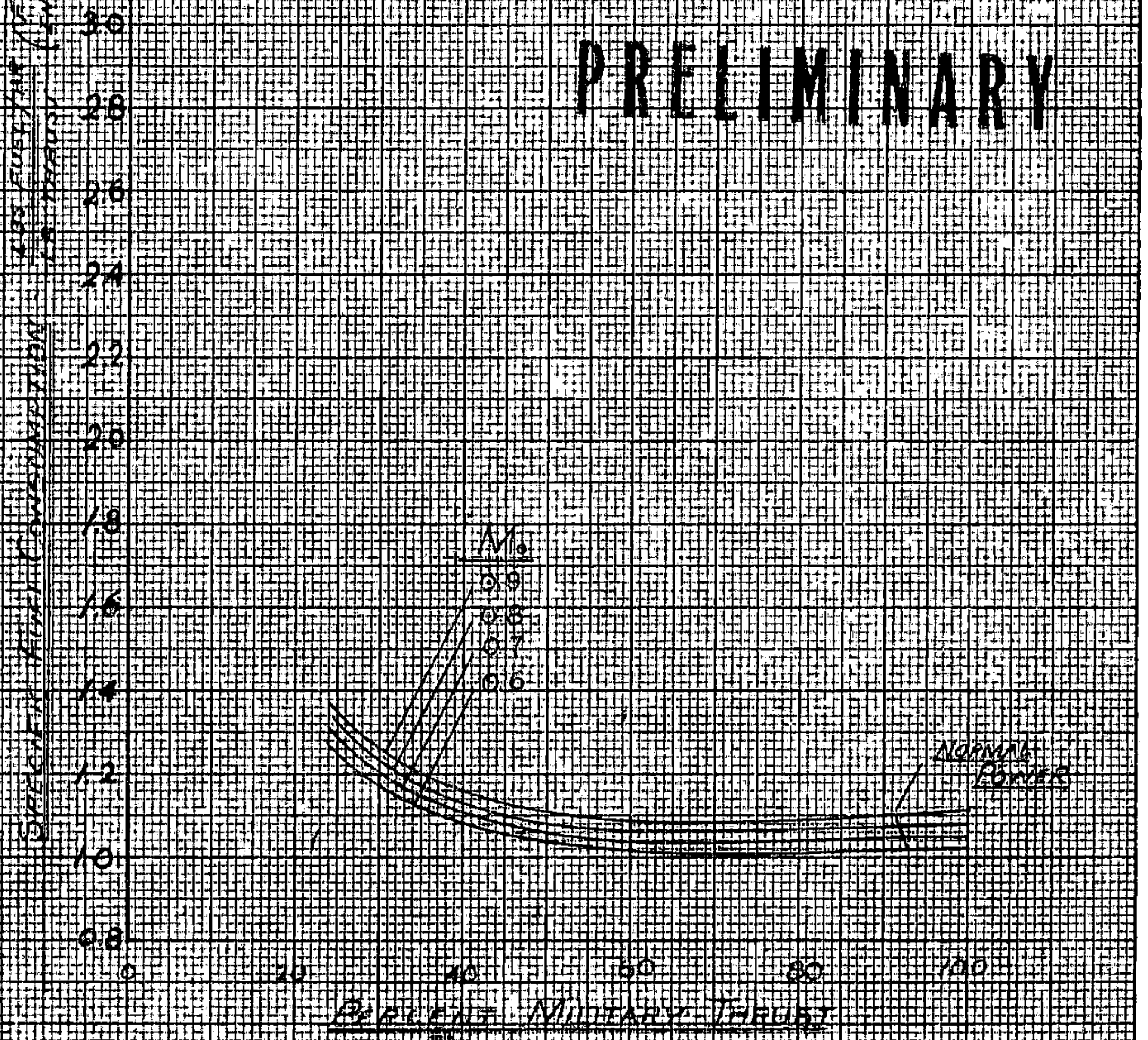
AIRPLANE - FV02
ENGINE - JT3C-24

P&W DATA: T-1435, 1-13
DATED 1-6-59

STANDARD ATMOSPHERIC CONDITIONS

ALTITUDE - 46,000 FT

INSTALLATION DATA:
CSD REPORT ZJ-8-012
DATED 5-4-55



PRELIMINARY

[redacted]

6-12-59

K+E KEUFFEL & ESSER CO.
ALBANENE ®
10 X 10 TO THE ½ INCH
359T-11G
MADE IN U.S.A.

[redacted]

UNCLASSIFIED

SECTION III - FATIGUE LIFE

This section of the report is devoted to a brief estimate of the fatigue life of the F-102 in the strike attack role. It was requested that an estimate of the F-102 fatigue life in performing strike attack missions be made. It was further requested that particular emphasis be placed upon gust loads between M = .8 and .9.

Using the missions of Convair Report No. ZP59-08002, F-102 Strike Attack Aircraft, equally distributed, together with actual gust frequency determinations, as outlined in Convair Memo Report DA-PD-108, (enclosed as an appendix) and with arbitrary maneuver loads, also therein outlined, a loading spectrum for these particular missions was obtained.

A critical structural member (a wing spar) on which fatigue tests had been run, thus affording a good estimate of fatigue strength, was selected.

The above load spectrum when applied to this member by two analytical methods yielded over 2,000 hours in each case.

It is, therefore, believed that the F-102 Strike Attack aircraft will have a fatigue life of over 2,000 hours when performing these particular strike attack missions.

UNCLASSIFIED

[redacted]

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

DA-PD-108

25 May 1959

Title: ESTIMATED NORMAL AND LATERAL FATIGUE SPECTRUMS FOR F-102 STRIKE ATTACK AIRCRAFT

References: (1). ZP-59-08002, "F-102 Strike Attack Aircraft," dated May 1959.

(2). NACA TN 4332, "An Approach to the Problem of Estimating Severe and Repeated Gust Loads for Missile Operations," dated September 1958.

(3). DA-8-382, "Elasticized Stability and Control Aerodynamic Coefficients for F-102A Large Tail Aircraft," dated 13 August 1956.

Subject Summary:

Figures 1 and 2 herein present rational estimates of normal and lateral fatigue spectrums for the F-102 airplane performing the strike attack missions specified in reference (1). [Structures Group has reportedly estimated (preliminary type) that when the structure of the F-102A airplane is rationally modified to account for the increased gross weight configurations specified in Reference (1), the airplane will have structural integrity for approximately 2000 hours of flight associated with the spectrums presented in Figures 1 and 2.]

Discussion:

The fatigue spectrums presented in Figures 1 and 2 herein are based on the following major assumptions:

(a). Airplane will fly equal number of each type "strike-mission" specified on pages 13 and 14 of Reference (1).

(b). Airplane will average two take-off's and landings per "strike-mission". (i.e., fifty percent of take-offs and landings will not be associated with basic "strike-missions".)

(c). Airplane will acquire the cumulative gust history presented in Figure 6 of Reference (2) in both the normal and lateral directions during its flight in each of the assumed "Mach number-altitude" segments of its mission profiles.

(d). Airplane will perform (i) three maneuvers to one-half limit design per mission and (ii) one limit design maneuver per 1000 hours of flight.

Discussion (Cont'd):

(e). Airplane fatigue spectrum per 1000 hours of flight will be composed of the gust spectrum obtained from assumptions (a) and (c) combined with a maneuver spectrum based on assumptions (a) and (d). The maneuver spectrum will be developed by a "gust-application" distribution of the maneuvers up to one-half limit design with a linear extrapolation to the single application of limit design maneuver. In addition, the airplane will experience the take-offs and landings resulting from assumptions (a) and (b).

The following procedural outline was used to obtain the estimated fatigue spectrums presented in Figures 1 and 2 in accordance with the major assumptions specified above:

(I.) Divided each of the "strike-attack" mission profiles specified on pages 13 and 14 of reference (1) into Mach number (M)-altitude (h) segments correspond to $M \leq .6$ and $M \geq .85$ at each of the following altitudes: $h = 0 \rightarrow 2$, 20, 35, and 45 thousand feet.

(II.) Tabulated nautical miles travelled and time spent in each segment established in (I) and ratioed total sum of segments to obtain segment subtotals of mileage per 1000 hours of flight.

(III.) Calculated the incremental normal load factor and vertical tail load per unit derived gust velocity for each of the segments established in (I). These slopes were established by using a standard sharp-edge-gust formula and an average wing loading of 44.3 pounds per square foot with the applicable aerodynamic coefficients from reference (3).

(IV.) Using the mileage flown in each segment as determined in (II) and the cumulative frequency per mile specified in Figure 6 of Reference (1), determined the number of occurrences, in each segment, of derived gust velocity in multiples of ten up to sixty feet per second. Established associated load levels by using the slopes determined in (III).

(V). Compiled the number of occurrences determined in (IV) into small intervals of load level and obtained an accumulative total of load level vs number of applications. It was assumed that each encounter with a given level of derived gust velocity would produce three peaks with a geometrical decay. (Based on minimum structural damping available in the subsonic regime).

DA-PD-108

Discussion (Cont'd):

(VI.) Estimated number of applications of one-half limit design maneuver per 1000 hours of flight by using the time total sum of segments obtained in (II.), and assumption (d).

(VII.) Constructed fatigue spectrums presented in Figures 1 an 2 using data from (V.) and (VI.) in accord with assumptions (d.) and (e.).

Figure 1 presents an estimated normal fatigue spectrum in terms of number of applications, per 1000 hours of flight, of a given level of incremental limit normal load factor ($\Delta n_z = \frac{\text{Total Normal Force}}{\text{Gross Weight}} - 1$). Since the normal fatigue spectrum is predominantly established by the assumed maneuvers, it must be designated by two curves which reflect the difference in magnitude of the positive ($n_z = 7.0$) and negative ($n_z = -3.0$) design levels.

Figure 2 presents an estimated lateral fatigue spectrum in terms of number of applications, per 1000 hours of flight, of a given level of vertical tail load ($\pm L_{tail} \sim$ pounds). One curve is sufficient to designate the spectrum since the positive and negative lateral design levels are equal. The difference in basic shape between the lateral and normal fatigue spectrums is attributable to the fact that the design maneuver loads for the vertical tail are much smaller than the design gust loads. (See assumption (d) and preceding paragraph.).

Prepared by ______________________
L. G. Barbee

Approved by ______________________
J. Weisman

cc:

L. V. Clements	6-143
T. M. Hemphill	6-109
Dynamic Files	6-106

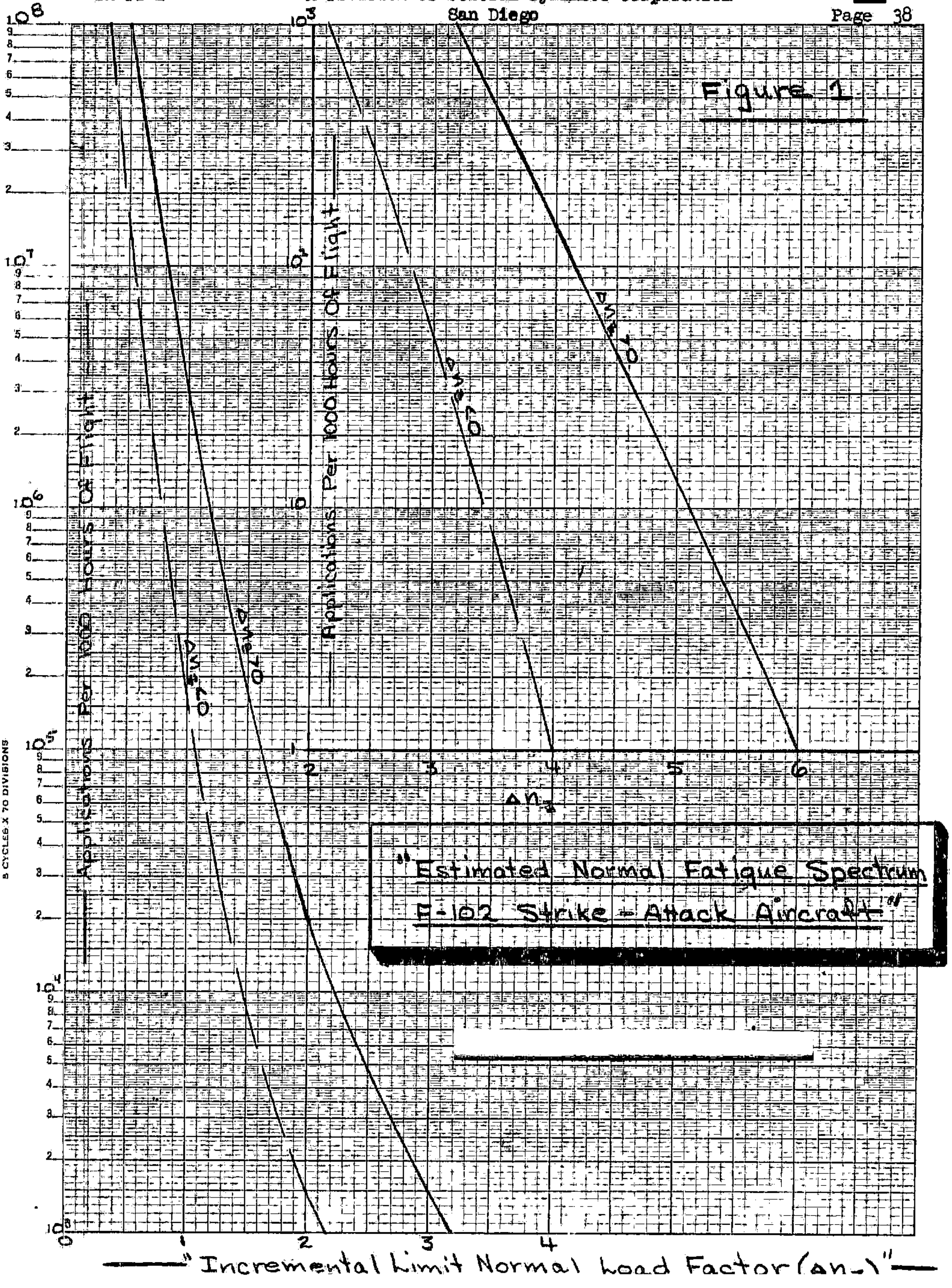


Figure 1

"Estimated Normal Fatigue Spectrum
F-102 Strike-Attack Aircraft"

K&E SEMI-LOGARITHMIC 359-91
KEUFFEL & ESSER CO. MADE IN U.S.A.
5 CYCLES X 70 DIVISIONS

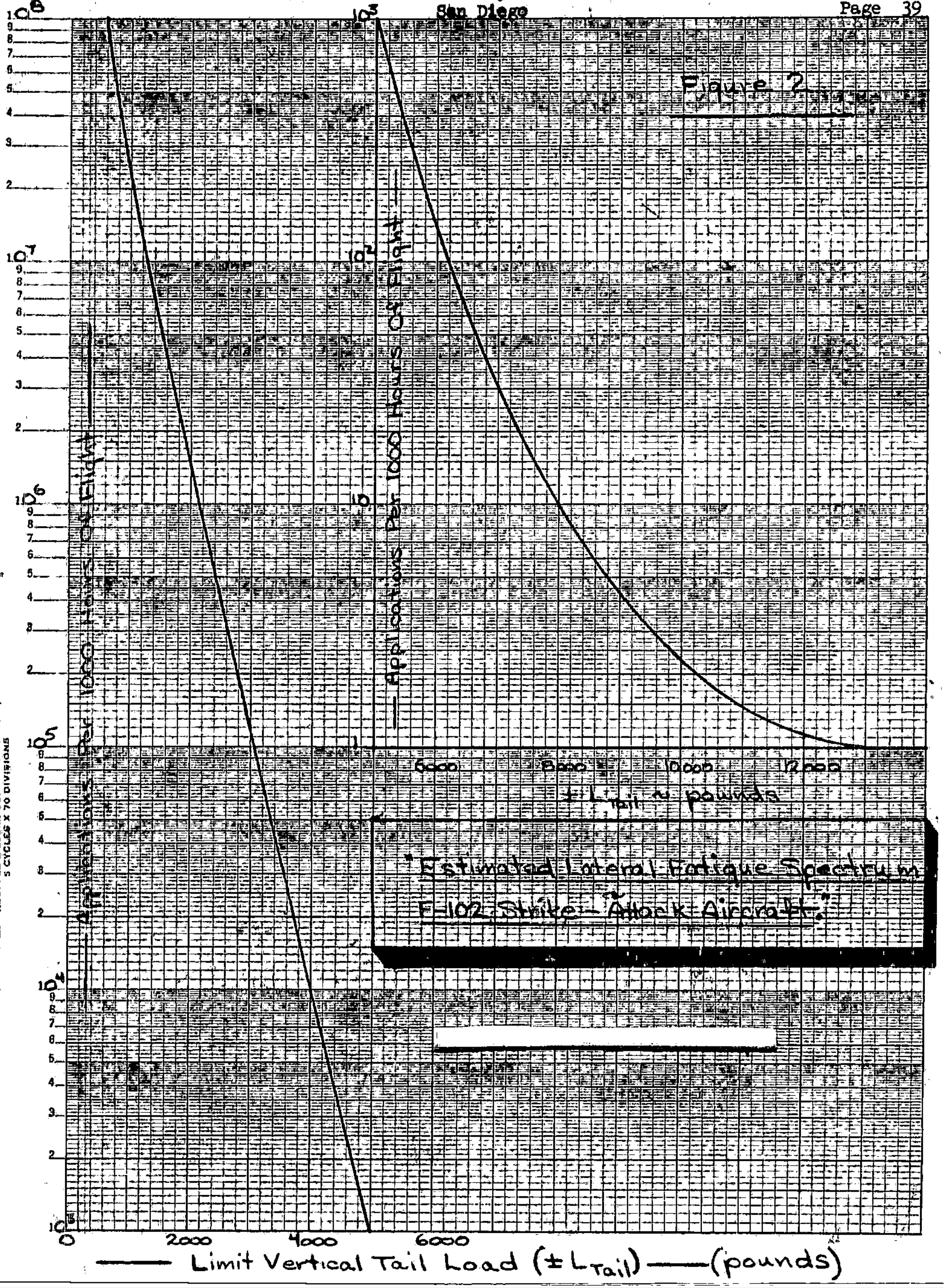


Figure 2

"Estimated Lateral Fatigue Spectrum
F-102 Strike-Attack Aircraft."

K&E SEMI-LOGARITHMIC 359-91
KEUFFEL & ESSER CO. MADE IN U.S.A.
5 CYCLES X 70 DIVISIONS

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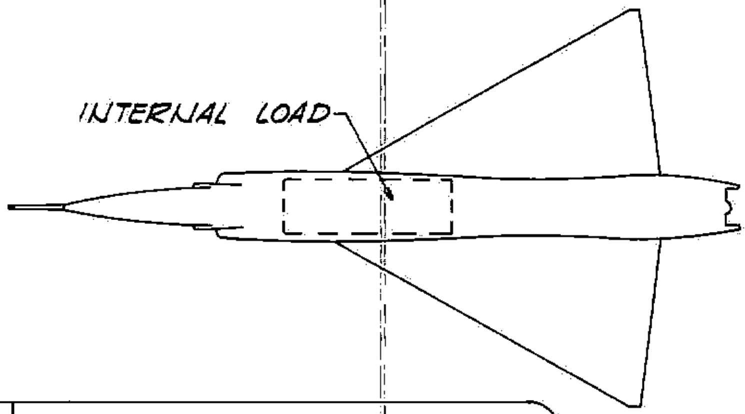
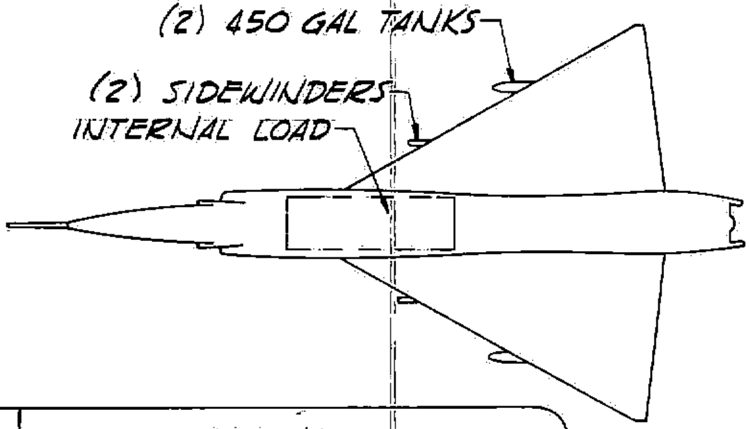
F 102 STRIKE ATT

TYPICAL

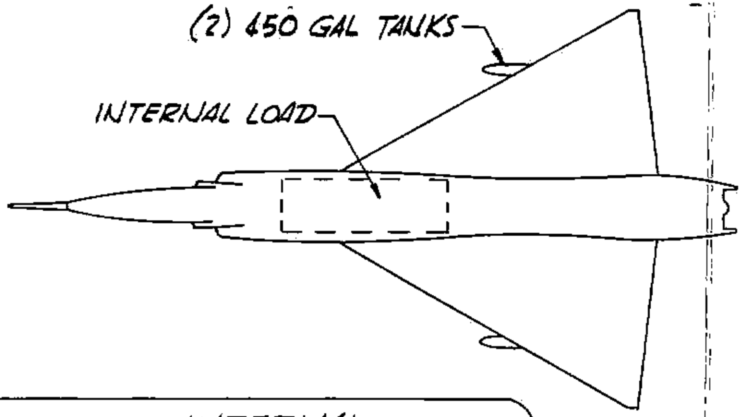


	EXTERNAL	INTERNAL
1		1000 LB BOMB IN BAY (1) 2.75" ROCKETS IN DOORS (24)
2		2000 LB STORE IN BAY (1) 2.75" ROCKETS IN DOORS (24)

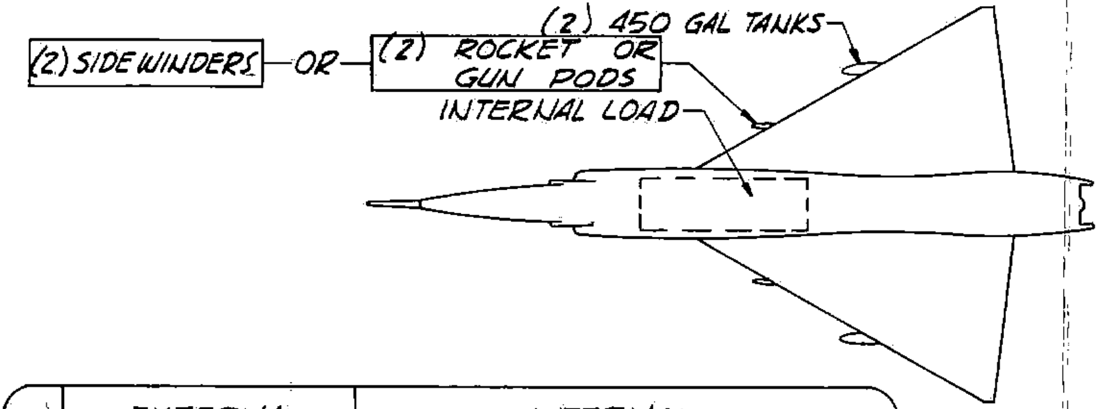


	EXTERNAL	INTERNAL
5	450 GAL TANKS (2) SIDEWINDER (2)	1000 LB BOMB IN BAY (1) 2.75" ROCKETS IN DOORS (24)
6	450 GAL TANKS (2) SIDEWINDER (2)	150 GAL TANK IN BAY (1) 2.75" ROCKETS IN DOORS (24)

ACK AIRCRAFT
LOADINGS



	EXTERNAL	INTERNAL
3	450 GAL TANKS(2) (FERRY CONDITION)	150 GAL TANK IN BAY (1)
4	450 GAL TANKS(2)	CAMERA IN BAY (2) 150 GAL TANK IN BAY (1) 2.75" ROCKETS IN DOORS(24)



	EXTERNAL	INTERNAL
7	450 GAL TANKS(2) SIDEWINDER (2)	CAMERA IN BAY (2) 150 GAL TANK IN BAY (1) 2.75" ROCKETS IN DOORS (24)
8	450 GAL TANKS(2) ROCKET OR GUN PODS (2)	1000 LB BOMB (1) 2.75" ROCKETS IN DOORS (24)

[redacted]

Crowlmeyer 6-4-59

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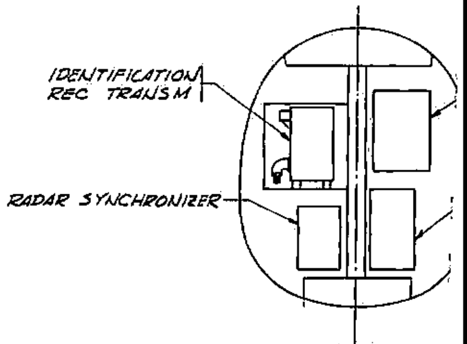
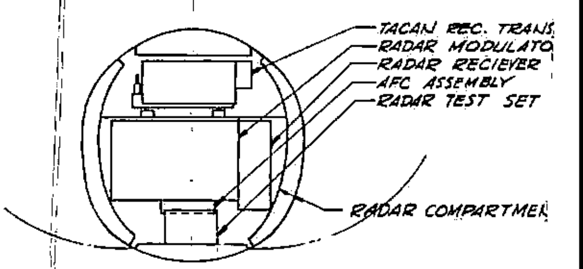
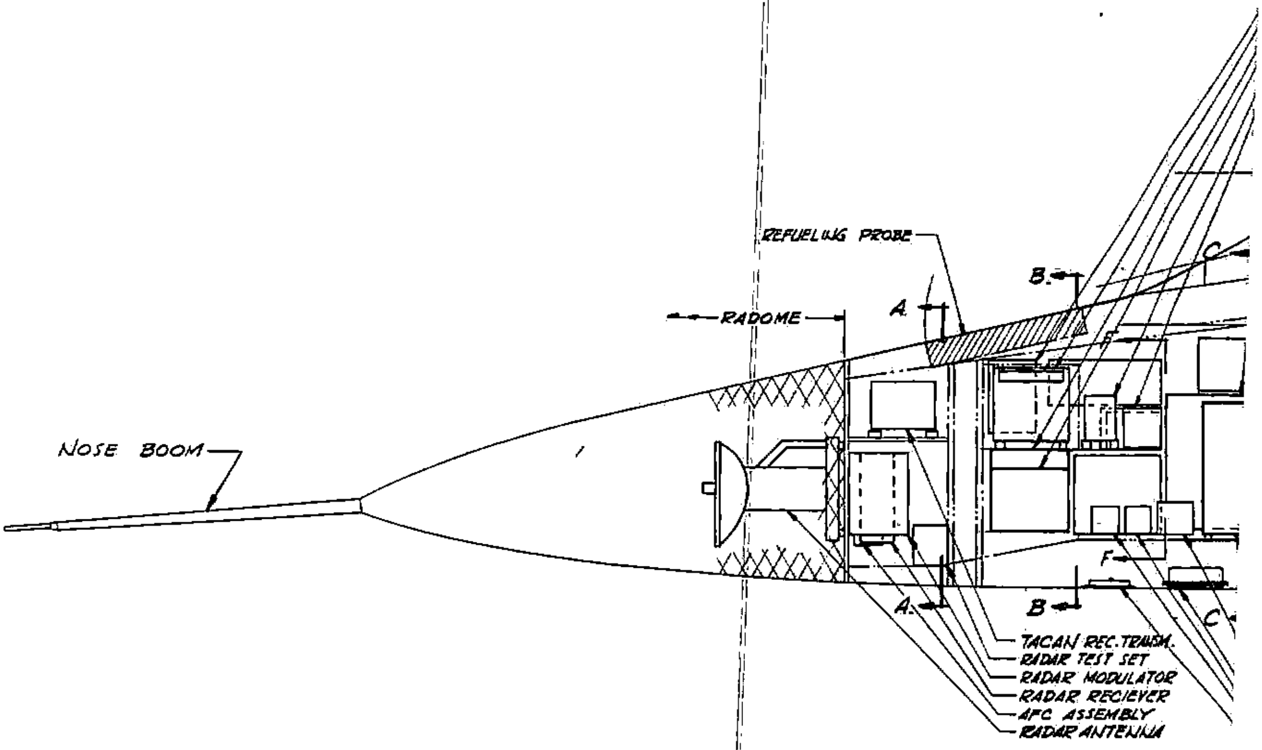
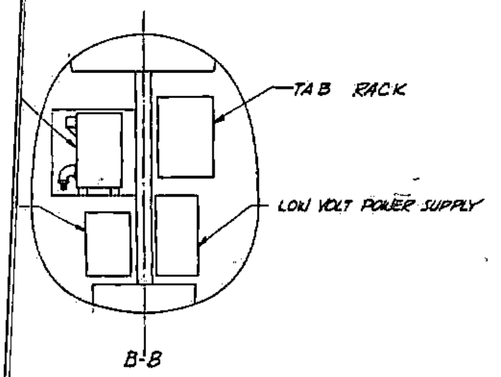


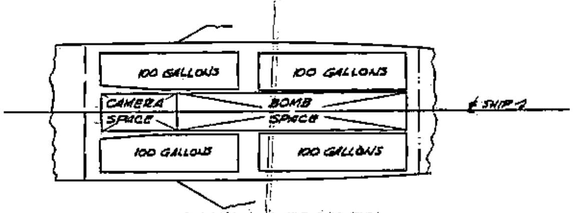
B-B



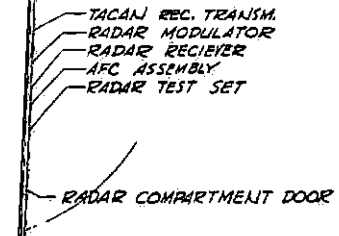
A-A

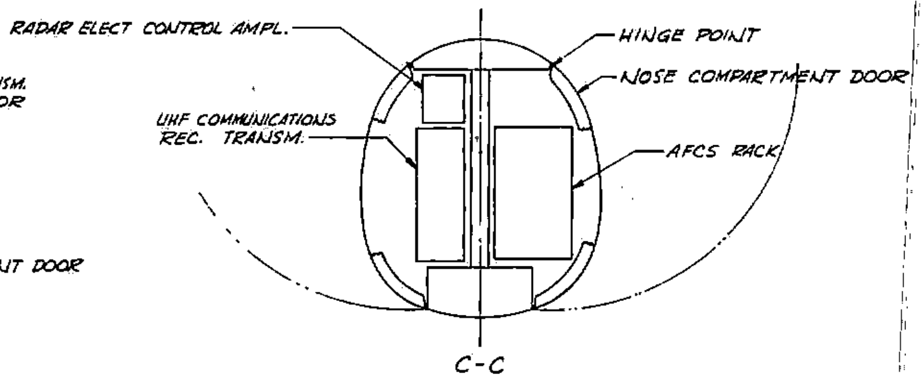


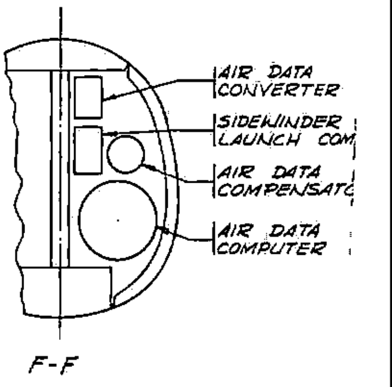


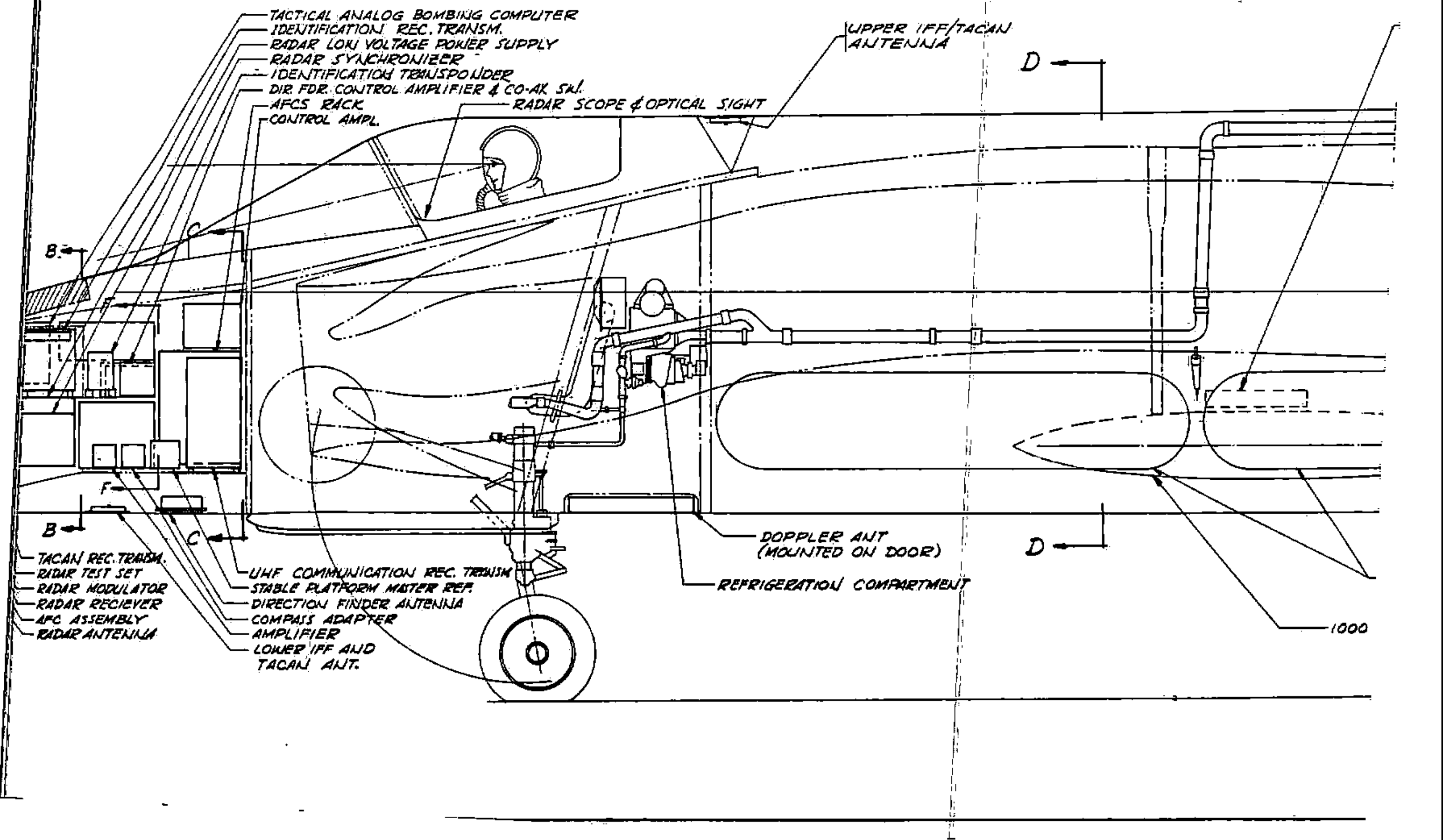


PLAN VIEW- MISSILE BAY AREA NO SCALE

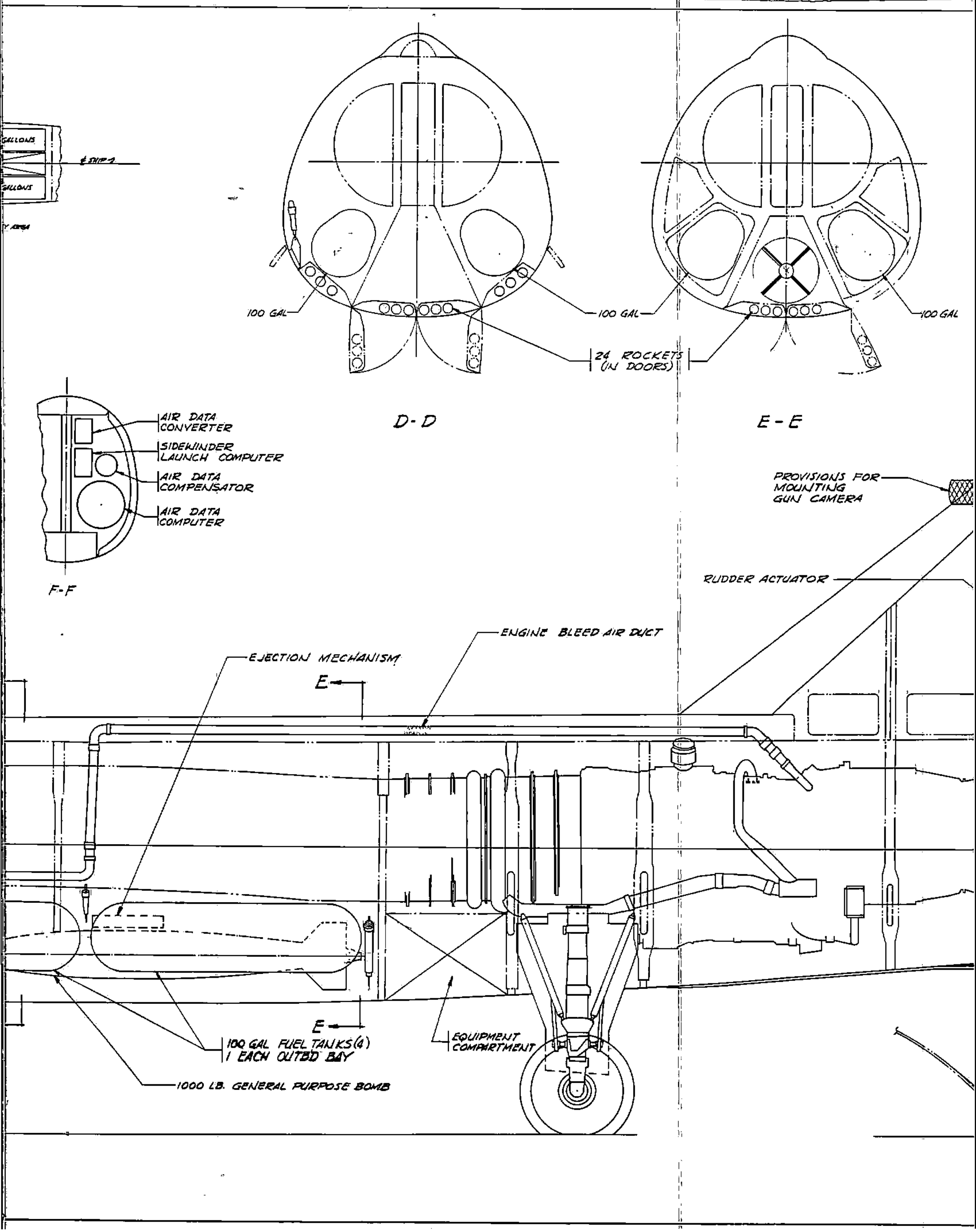


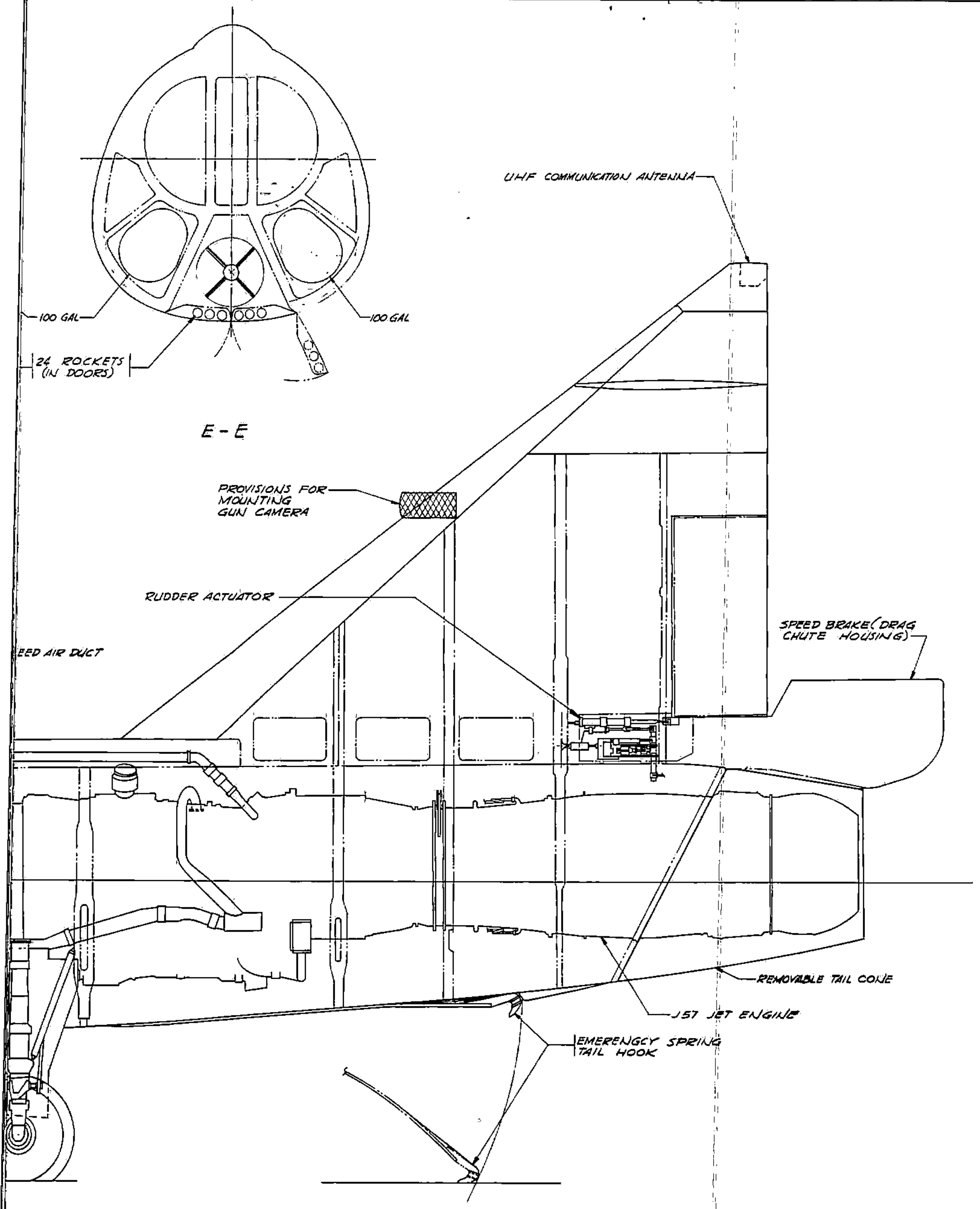




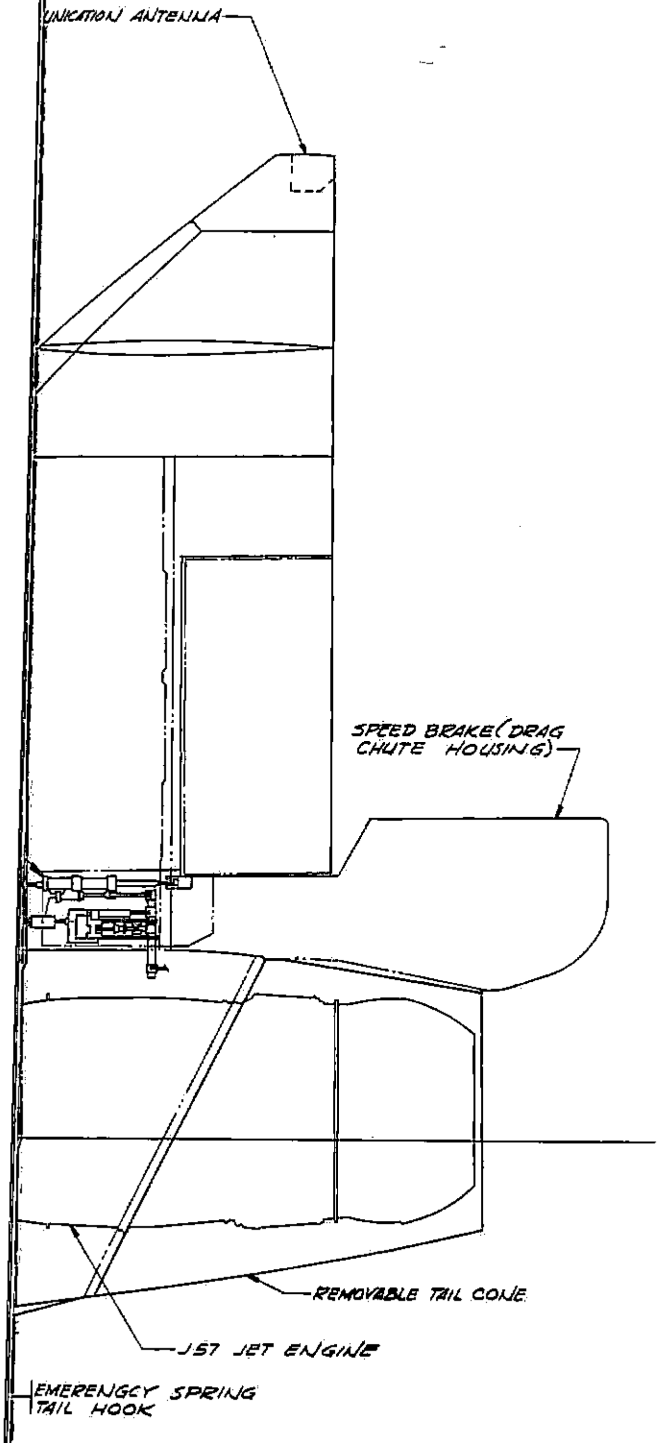


TACTICAL ANALOG BOMBING COMPUTER
IDENTIFICATION REC. TRANSM.
RADAR LOW VOLTAGE POWER SUPPLY
RADAR SYNCHRONIZER
IDENTIFICATION TRANSPONDER
DIR. FDR. CONTROL AMPLIFIER & CO-AX. SW.
AFCS RACK
CONTROL AMPL.
RADAR SCOPE & OPTICAL SIGHT
UPPER IFF/TACAN ANTENNA
D
B
C
F
F
B
C
D
TACAN REC. TRANSM.
RADAR TEST SET
RADAR MODULATOR
RADAR RECIEVER
AFC ASSEMBLY
RADAR ANTENNA
UHF COMMUNICATION REC. TRANSM
STABLE PLATFORM MASTER REF.
DIRECTION FINDER ANTENNA
COMPASS ADAPTER
AMPLIFIER
LOWER IFF AND TACAN ANT.
DOPPLER ANT (MOUNTED ON DOOR)
REFRIGERATION COMPARTMENT
1000


GALLONS
℄ SHIP
GALLONS
Y AREA
100 GAL
100 GAL
24 ROCKETS (IN DOORS)
100 GAL
100 GAL
D-D
E-E
AIR DATA CONVERTER
SIDEWINDER LAUNCH COMPUTER
AIR DATA COMPENSATOR
AIR DATA COMPUTER
F-F
PROVISIONS FOR MOUNTING GUN CAMERA
RUDDER ACTUATOR
ENGINE BLEED AIR DUCT
EJECTION MECHANISM
E
E
100 GAL FUEL TANKS (4) 1 EACH OUTBD BAY
EQUIPMENT COMPARTMENT
1000 LB. GENERAL PURPOSE BOMB



100 GAL
100 GAL
24 ROCKETS (IN DOORS)
E - E
UHF COMMUNICATION ANTENNA
PROVISIONS FOR MOUNTING GUN CAMERA
RUDDER ACTUATOR
EED AIR DUCT
SPEED BRAKE (DRAG CHUTE HOUSING)
REMOVABLE TAIL CONE
J57 JET ENGINE
EMERENGCY SPRING TAIL HOOK



[redacted]

CONVAIR

PRELIMINARY DESIGN DRAWING

INBOARD PROFILE
F102A STRIKE ATTACK

CROWHOVER CHECKED APPROVED SCALE 1/10 DATE 6-10-59

CONVAIR–SAN DIEGO, CALIFORNIA SD 5908003
A DIVISION OF GENERAL DYNAMICS CORPORATION

ZP-8-044

ZP-8-044

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-044
DATE 16 August 1956
MODEL

~~CONFIDENTIAL~~

UNCLASSIFIED

TITLE

INSTALLATION OF TWO GAR-X MISSILES IN THE F-106A AND F-106B AIRPLANES

ZP-8-044

16 August 1956

PREPARED BY R. C. Lingley
R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY E R Tuttle for
A. R. Lambert

Classification Canceled
USAF 1st Ind. (AFMDC-3E/
Maj. Fristoe/MT/71843) 15AU660
Authorized by: Date
6-17-68
Reclassified by Dept. Date

NO. OF PAGES
NO. OF DIAGRAMS

REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED

UNCLASSIFIED

FORM 1612 A-4

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INTRODUCTION

This preliminary study was made by Air Force request to install two GAR-X missiles in the F-106A and F-106B airplanes in place of one MB-1 rocket and four Falcon missiles.

Three possible GAR-X missile configurations have been proposed by Hughes Aircraft Company:

1. 133.50" long - 625 lbs. (Reference HAC Drawing SP 30865)
2. 113.90" long - 607 lbs. (Reference HAC Drawing SP 30876)
3. 113.90" long - 563 lbs. (Reference HAC Drawing SP 30875)

It is assumed that ultimately only one of the above configurations will be produced; hence, no provisions for interchangeability have been considered.

The anticipated operational date of the GAR-X is mid 1959 which would be compatible with the F-106A and F-106B production schedules. It would be feasible to incorporate the required changes on a retrofit basis for those airplanes that might not be picked up on the production line.

This study assumes that the weapons will be displaced by an ejection system (one at a time for a "two-pass" capability). Convair has had success with a similar system for ejection of the MB-1 rocket in tests conducted at Holloman Air Development Center. These tests have shown excellent ejection trajectories at speeds up to Mach 1.10 and altitudes up to 40,000 feet. If certain limitations on the weapon preclude the use of an ejection system, then further studies will be required to develop a method for displacing the weapon.

Similarity of the F-106A and F-106B airplanes eliminates the necessity of separate studies, hence, the changes for both airplanes are included in this report.

All provisions for the MB-1 rocket and four Falcon missiles will be removed from the airplanes.

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SUMMARY

Conversion of the F-106A and F-106B airplanes to carry two GAR-X weapons can be accomplished with varying degrees of complexity depending upon the particular GAR-X weapon selected.

Two long (133.50") GAR-X weapons can be installed in the existing missile bay of either airplane with minor structural changes in the area of the ejection mechanism and revisions to the missile bay doors and mechanism. The existing electronic components in the forward-end of the F-106B missile bay must be repackaged in the same general area to clear the noses of the two GAR-X weapons.

Two short (113.90") GAR-X weapons will be more difficult to install since their folded geometry is larger, and the fixed portion of the wings has a greater span. This will necessitate a local revision to the missile bay roof. In addition, new missile bay doors must be designed with indentations toclear the wings and fins of the weapon. The fixed portion of the wings will require a local blister on the outside of the doors. The hinge at the folding portion of the doors must be cutout for approximately 45" to clear the lower folded portion of the wings and fins of the weapon. The missile bay longeron must be reinforced for approximately 30" because of the hinge cutout and the indentation of the doors adjacent to the longeron. The geometry of the door mechanism must be revised to provide greater door travel in order to clear the two lower wings of the weapon which must be opened prior to ejection. The baffle between the two weapons must be cut out locally to clear the unfolded portion of the short GAR-X wings.

This study indicates that the 133.50" long weapons would be preferred by Convair due to the relatively minor structural changes required.

The changes required to install the short weapons may be of even greater consequence than this brief study indicates. Past studies have shown that it will be very difficult to revise the door mechanism within the space available to provide greater door travel. The required cutouts in the doors remove such a great amount of door structure that it might be necessary to bulge the external lines of the airplane to provide for adequate door structure.

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SUMMARY (Continued)

With the proposed armament load the airplane gross weight will be reduced by a varying amount (depending on which of the three GAR-X weapons is finally selected) and there will be a slight aft C.G. shift which will remain within acceptable limits.

There will be no performance degradation when carrying two GAR-X weapons due to the basic Convair concept of installing all armament internally.

Recommendation:

If the fixed wing portions of the short GAR-X can be folded (or eliminated), the extent of the modifications to the airplane structure will be reduced.

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DISCUSSION

The following brief description of changes required to install two GAR-X weapons in the F-106A or F-106B airplanes is based on an ejection system for displacing the weapons.

Further studies must be made to determine the precise detail changes required:

A. General

1. Remove the existing center line ejection mechanism and provisions for the MB-1 rocket.
2. Remove the existing displacement gear and provisions for the (4) Falcon missiles.
3. Remove the missile bay doors.
4. Remove the door actuating mechanism (6 places).
5. Revise the electronic missile auxiliary components to provide for (2) GAR-X in lieu of (1) MB-1 and (4) Falcons. Hughes Aircraft has advised that approximately four units will be changed but that space provision in the existing racks are adequate.
6. Re-package electronic components in the forward end of the missile bay. (F-106B only).
7. Revise electrical system as required to accommodate the two GAR-X installations.

B. Installation of Long GAR-X (Reference Drawing SD-56-08059)

1. Install two 133.50" GAR-X weapons in aft end of missile bay.
2. Revise structure to provide for the relocated ejection mechanism and associated ejection loads.

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3. Add baffle between weapons to decrease turbulence in bay.

4. Add new missile bay doors and actuating mechanism from station 253 to 412 and revise pneumatic system accordingly.

5. Add partition across missile bay at station 253.

6. Add structure to fair over forward end of bay station 216.50 to 253.

7. Re-package electronic components from the existing missile bay rack into this compartment (For F-106B only).

C. Installation of Short GAR-X (Reference Drawing SD-56-08060)

1. Install two 113.50" GAR-X weapons in aft end of missile bay.

2. Revise structure to provide for the relocated ejection mechanism and associated ejection loads.

3. Add baffle between weapons to decrease turbulance in bay.

4. Add new missile bay doors and actuating mechanism from station 253 to 412 and revise pneumatic system accordingly.

5. Add partition across missile bay at station 253.

6. Add structure to fair over forward end of bay station 216.50 to 253.

7. Re-package electronic components from the existing missile bay rack into this compartment (For F-106B only).

8. Revise roof of missile bay locally, to clear the fixed portion of the GAR-X wings.

9. Revise geometry of missile bay doors to provide greater travel.

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10. Cut out approximately 45" of hinge at folding portion of doors to clear wings and fins of the weapon.

11. Provide cutouts in doors to clear wings and fins of the weapon.

12. Provide local blisters on outside of doors to clear fixed wing portion of the weapon.

13. Reinforce missile bay longeron from Stations 310 to 340.

14. Cut out baffle in center of bay to clear fixed wing portion of weapon.

Following is the weight summaries and C. G. curves for both the F-106A and F-106B airplanes with any of the three proposed GAR-X weapons installed.

Drawings SD-56-08059 and SD-56-08060 show installations of the long and short weapons in either the F-106A or F-106B airplanes.

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PRELIMINARY WEIGHT STUDY

A. COMPARATIVE GROUP WEIGHT STATEMENTS - F-106A with (2) 133.5" Long GAR-X In lieu of (1) MB-1 Type Rocket + (4) GAR-3/-4 Missiles

		(1) MB-1 Type Rocket +(4) GAR-3/-4 Falcons	(2) GAR-X (133.5")
Wing Group		3274	3274
Tail Group		636	636
Body Group		4654	4840
Alighting Gear Group		1179	1179
Surface Controls Group		437	437
Engine Section		37	37
Propulsion Group		(8000)	(8000)
Engine Installation		5911	5911
Air Induction System		888	888
Exhaust System		264	264
Cooling System		46	46
Lubricating System		50	50
Fuel System		750	750
Engine Controls		26	26
Starting System		65	65
Fixed Equipment Group		(4918)	(4416)
Instruments and Navigational Equipment Group		134	134
Hydraulic and Pneumatic Group		381	353
Electrical Group		535	535
Electronics Group		2522	2502
Armament Group		684	230
Frunishings and Equipment Group		263	263
Air Conditioning and Anti-Icing Group		346	346
Auxiliary Gear Group		53	53
WEIGHT EMPTY		23135	22819
Pilot, Survival Kit, and Parachute		270	270
Fuel - Unusable (JP-4)		239	239
Fuel - Usable (JP-4)	1240 Gals.	8060	8060
Oil - Trapped and Engine		60	60
MB-1 Type Rocket	(1)	815	---
GAR-X Weapon @ 625	(2)	---	1250
Ejector Cartridges		1	2
GAR-3/-4 Missile	(4)	540	---
NORMAL TAKEOFF GROSS WEIGHT		33120*	32700

* Convair Status dated 15 August 1956,
Plus 20% Span Wing Camber

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PRELIMINARY WEIGHT STUDY
(Cont'd.)

B. F-106A with (2) 113.9" Long GAR-X (29.91" Fixed Wing Diameter)

	Weight
Normal Take-off Gross Weight F-106A with (2) 133.5" Long GAR-X @ 625	32700
Less	
GAR-X (133.5" Long) (2)	-1250
Plus	
Missile Bay Structural Revisions	+ 46
Door Structure and Mechanism Changes	+ 50
GAR-X (113.9" Long, 29.91" Fixed Wing Diameter) (2)	+1214
Normal Take-off Gross Weight - F-106A with (2) @ 607 113.9" Long GAR-X (29.91" Fixed Wing Diameter)	32760

C. F-106A with (2) 113.9" Long GAR-X (26.89" Fixed Wing Diameter)

	Weight
Normal Take-off Gross Weight F-106A with (2) 133.5" Long GAR-X	32700
Less	
GAR-X (133.5" Long) (2)	-1250
Plus	
Missile Bay Structural Revisions	+ 40
Door Structure and Mechanism Changes	+ 44
GAR-X (113.9" Long, 26.89" Fixed Wing Diameter) (2)	+1126
Normal Take-off Gross Weight - F-106A with (2) @ 563 GAR-X (113.9" Long, 26.89" Fixed Wing Diameter)	32660

NOTE: The weight studies above do not include any allowance for possible missile bay temperature environmental problems associated with the GAR-X weapons.

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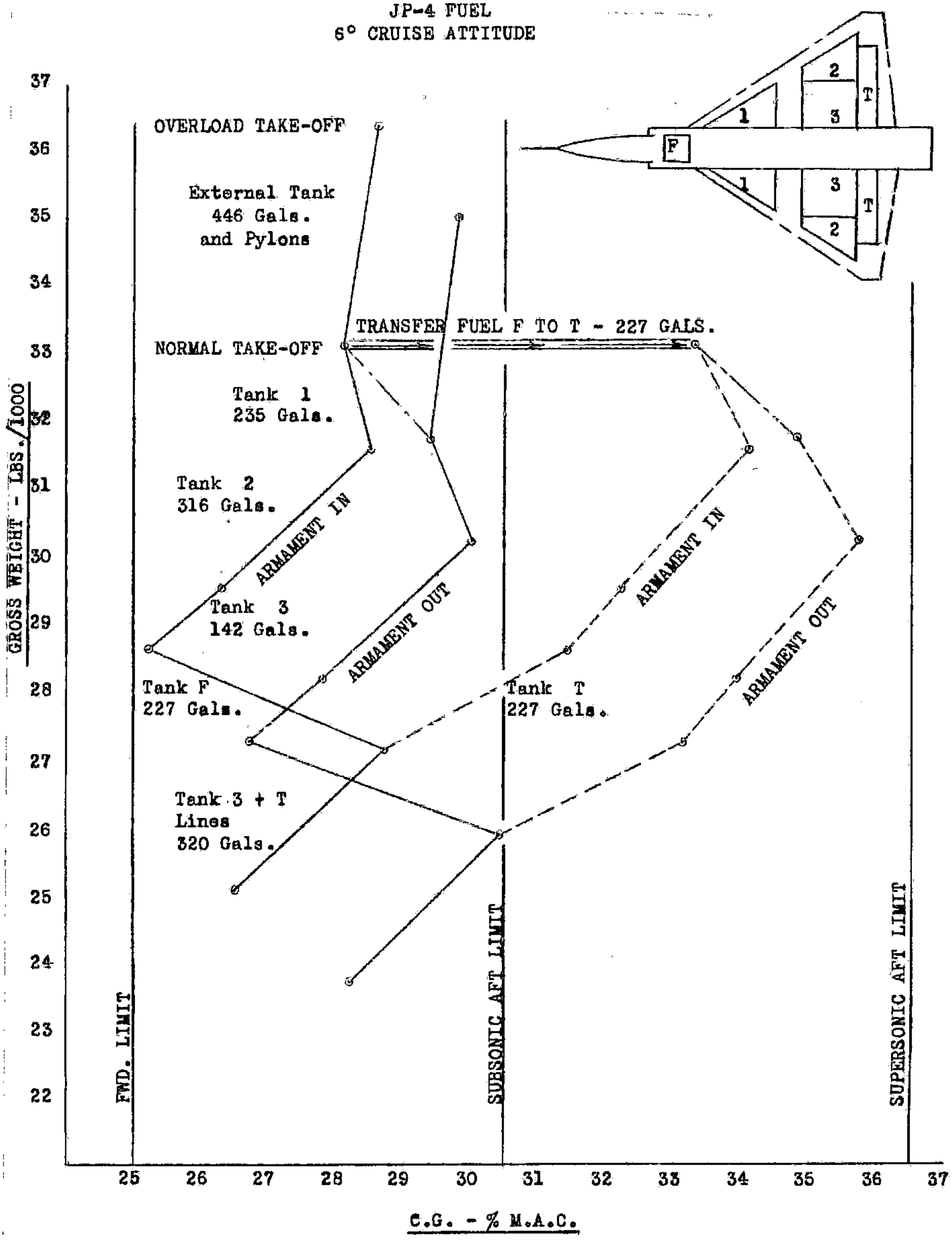
F-106A WITH (1) MB-1 TYPE ROCKET & (4) GAR -3/-4 MISSILES

CENTER OF GRAVITY ENVELOPE
JP-4 FUEL
6° CRUISE ATTITUDE



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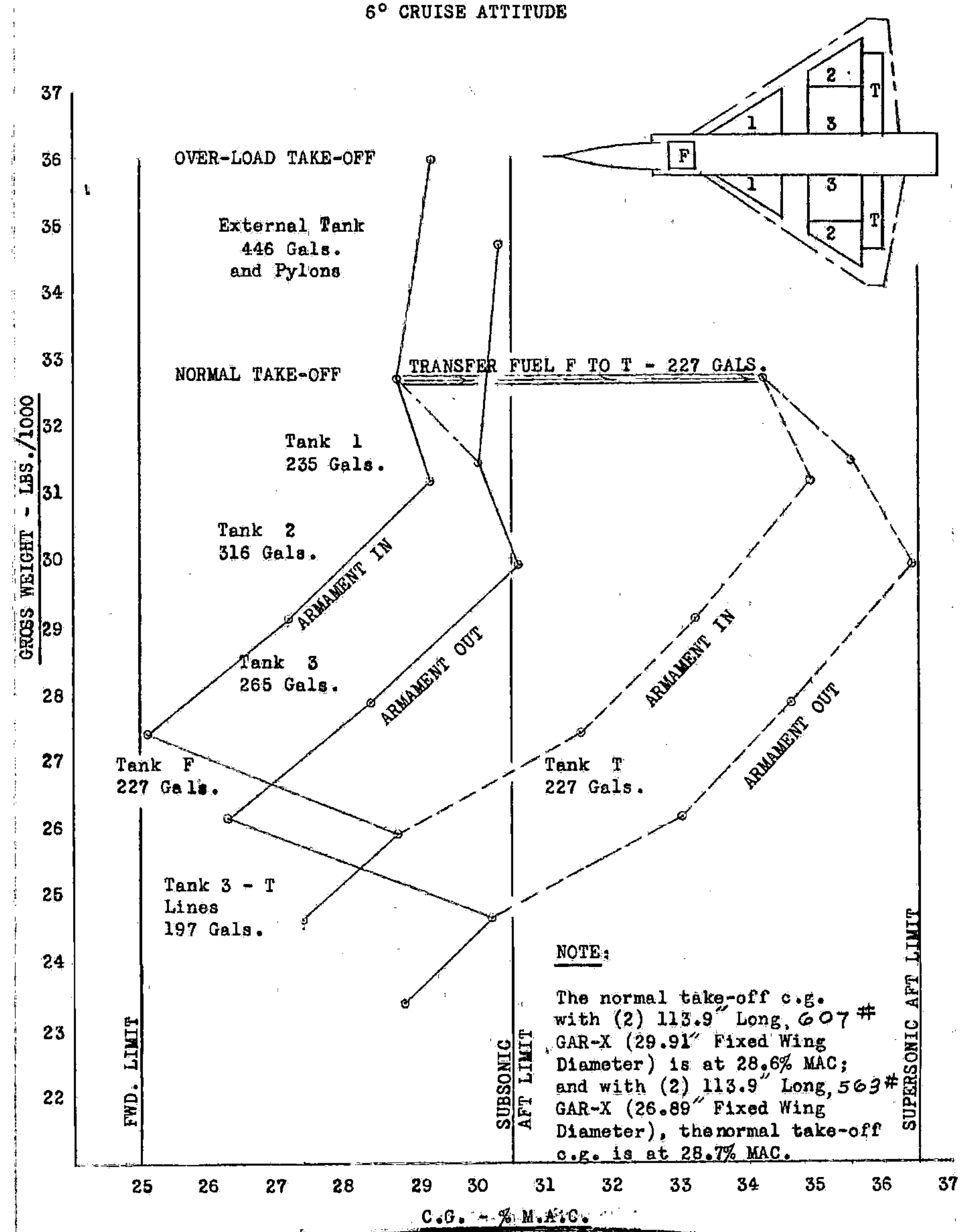
F-106A WITH (2) 133.5" LONG GAR-X WEAPONS @ 625

CENTER OF GRAVITY ENVELOPE
JP-4 FUEL
6° CRUISE ATTITUDE



NOTE:

The normal take-off c.g. with (2) 113.9" Long, 607 # GAR-X (29.91" Fixed Wing Diameter) is at 28.6% MAC; and with (2) 113.9" Long, 563 # GAR-X (26.89" Fixed Wing Diameter), the normal take-off c.g. is at 28.7% MAC.

[redacted]

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PRELIMINARY WEIGHT STUDY

A. COMPARATIVE GROUP WEIGHT STATEMENTS - F-106B with (2) 133.5" Long GAR-X In lieu of (1) MB-1 Type Rocket +(4) GAR-3/-4 Missiles

		(1) MB-1 Type Rocket +(4) GAR-3/-4 Missiles	(2) GAR-X (133.5")
Wing Group		3274	3274
Tail Group		636	636
Body Group		5213	5346
Alighting Gear Group		1195	1195
Surface Controls Group		480	480
Engine Section		37	37
Propulsion Group		(8033)	(8033)
Engine Installation		5911	5911
Air Induction System		889	889
Exhaust System		264	264
Cooling System		46	46
Lubricating System		51	51
Fuel System		770	770
Engine Controls		36	36
Starting System		66	66
Fixed Equipment Group		(5527)	(4948)
Instruments and Navigational Equipment Group		167	167
Hydraulic and Pneumatic Group		382	354
Electrical Group		558	558
Electronics Group		2752	2732
Armament Group		762	231
Furnishings and Equipment Group		458	458
Air Conditioning and Anti-Icing Group		394	394
Auxiliary Gear Group		54	54
WEIGHT EMPTY		24395	23949
Pilots, Survival Kits, and Parachutes		540	540
Fuel - Unusable (JP-4)		239	239
Fuel - Usable (JP-4)	1240 Gals.	8060	8060
Oil - Trapped and Engine		60	60
MB-1 Type Rocket	(1)	815	---
GAR-X Weapon @ 625	(2)	---	1250
Ejector Cartridges		1	2
GAR-3/-4 Missile	(4)	540	---
NORMAL TAKE-OFF GROSS WEIGHT		34650 *	34100

* Convair Preliminary Status dated 15 August 1956, Plus 20% Span Wing Camber

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PRELIMINARY WEIGHT STUDY
(Cont'd.)

B. F-106B with (2) 113.9" Long GAR-X (29.91" Fixed Wing Diameter)

		Weight
Normal Take-off Gross Weight F-106B with (2) 133.5" Long GAR-X @ 625		34100
Less		
GAR-X (133.5" Long)	(2)	-1250
Plus		
Missile Bay Structural Revisions		+ 46
Door Structure and Mechanism Changes		+ 50
GAR-X (113.9" Long, 29.91" Fixed Wing Diameter((2)		+1214
Normal Take-off Gross Weight @ 607 F-106B with (2) 113.9" Long GAR-X (29.91" Fixed Wing Diameter)		34160

C. F-106B with (2) 113.9" Long GAR-X (26.89" Fixed Wing Diameter)

		Weight
Normal Take-off Gross Weight F-106B with (2) 133.5" Long GAR-X		34100
Less		
GAR-X (133.5" Long)	(2)	-1250
Plus		
Missile Bay Structural Revisions		+ 40
Door Structure and Mechanism Changes		+ 44
GAR-X (113.9" Long, 26.89" Fixed Wing Diameter) (2)		+1126
Normal Take-off Gross Weight @ 563 F-106B with (2) 113.9" Long GAR-X (26.89" Fixed Wing Diameter)		34060

NOTE: The weight studies above do not include any allowance for possible missile bay temperature environmental problems associated with the GAR-X weapons.

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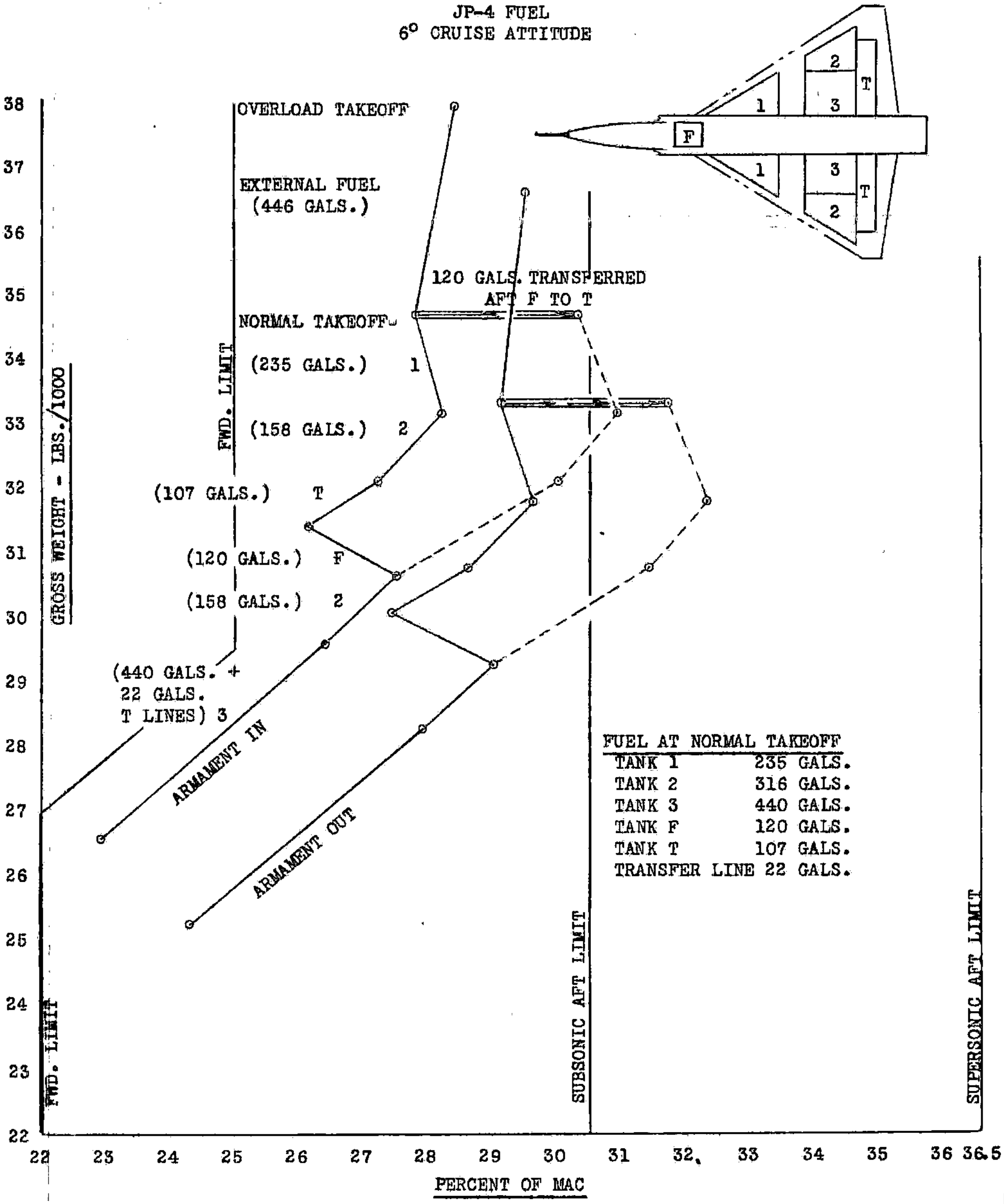
F-106B WITH (1) MB-1 TYPE ROCKET + (4) GAR-3/-4 MISSILES

PRELIMINARY CENTER OF GRAVITY ENVELOPE
TWO PILOTS
JP-4 FUEL
6° CRUISE ATTITUDE



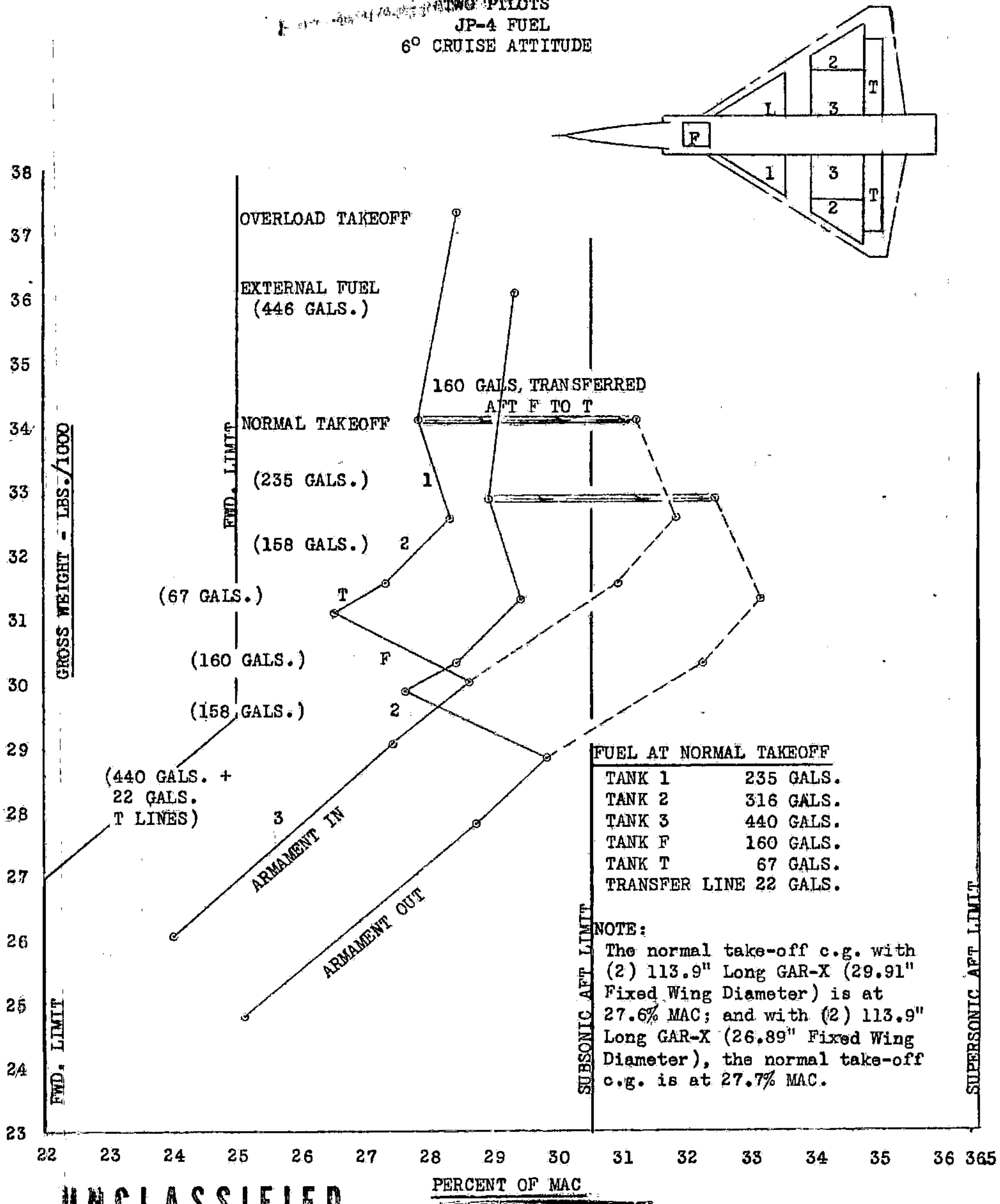
FUEL AT NORMAL TAKEOFF

TANK 1	235 GALS.
TANK 2	316 GALS.
TANK 3	440 GALS.
TANK F	120 GALS.
TANK T	107 GALS.
TRANSFER LINE	22 GALS.

[redacted]

F-106B WITH (2) 133.5" LONG GAR-X @ 625

PRELIMINARY CENTER OF GRAVITY ENVELOPE
TWO PILOTS
JP-4 FUEL
6° CRUISE ATTITUDE



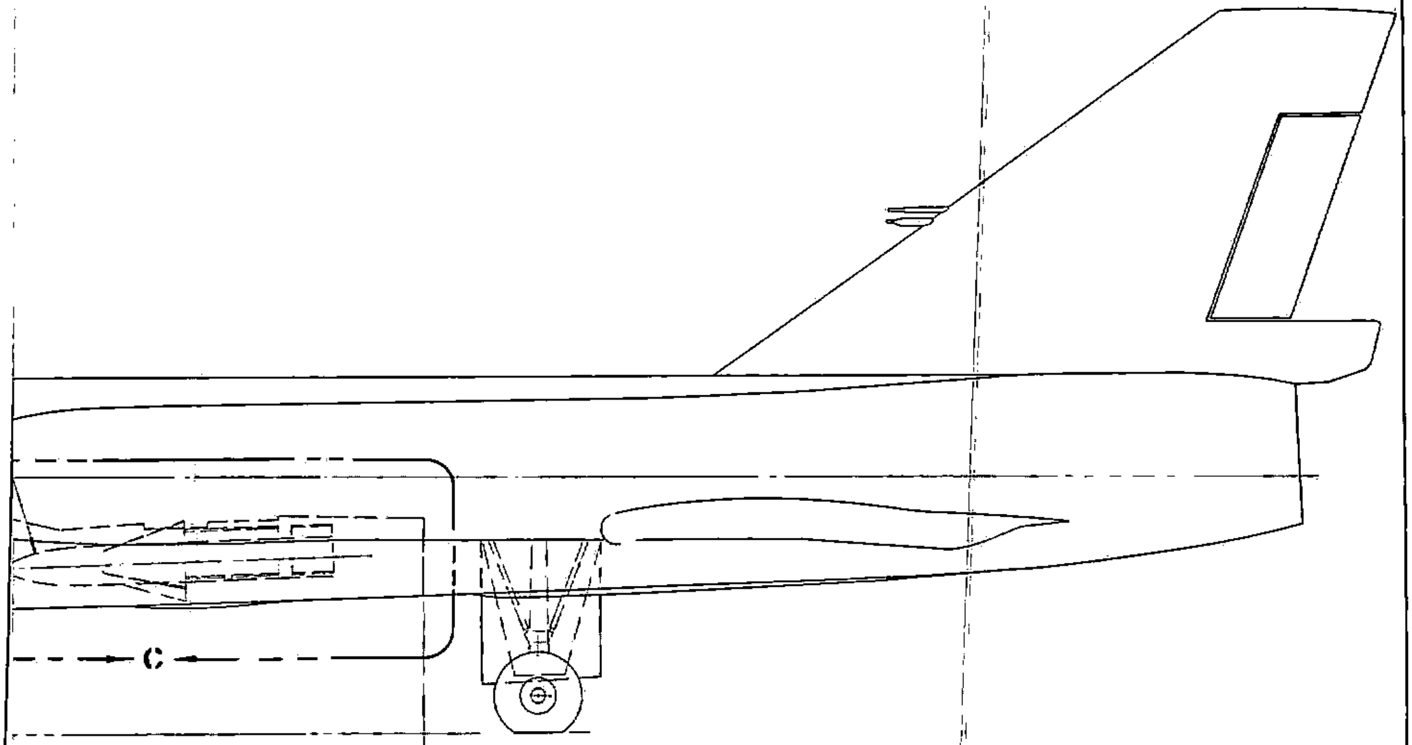
FUEL AT NORMAL TAKEOFF

TANK 1	235 GALS.
TANK 2	316 GALS.
TANK 3	440 GALS.
TANK F	160 GALS.
TANK T	67 GALS.
TRANSFER LINE	22 GALS.

NOTE: The normal take-off c.g. with (2) 113.9" Long GAR-X (29.91" Fixed Wing Diameter) is at 27.6% MAC; and with (2) 113.9" Long GAR-X (26.89" Fixed Wing Diameter), the normal take-off c.g. is at 27.7% MAC.

copys of regs to this Report

ZP-8-044
page 13



SIDE VIEW
TWENTIETH SCALE

NS (10·0 STROKE)

GAR-X MISSILE (606·5") WITH
29·91 FIXED WING DIA (HUGHES
DWG. NO. SP 30876 SHOWN STOWED
GAR-X MISSILE (562·5") WITH 26·89 FIXED
WING DIA.(HUGHES DWG NO. SP 30875)
NCE
UT

T BE EXTENDED
IING

SEE DWG 5D-56-08059 FOR INST'L
OF 133·5" GAR-X

[redacted]

CONVAIR
PRELIMINARY DESIGN DRAWING
STUDY - INSTL. OF (2) GAR-X
MISSILES (113·9" LONG)(2 WING TYPES)
IN F-106 A AIRPLANE

KRUNIAK	CHECKED	APPROVED	SCALE TENTH NOTED	DATE 8/16/56

CONVAIR—SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION
5D-56-08060

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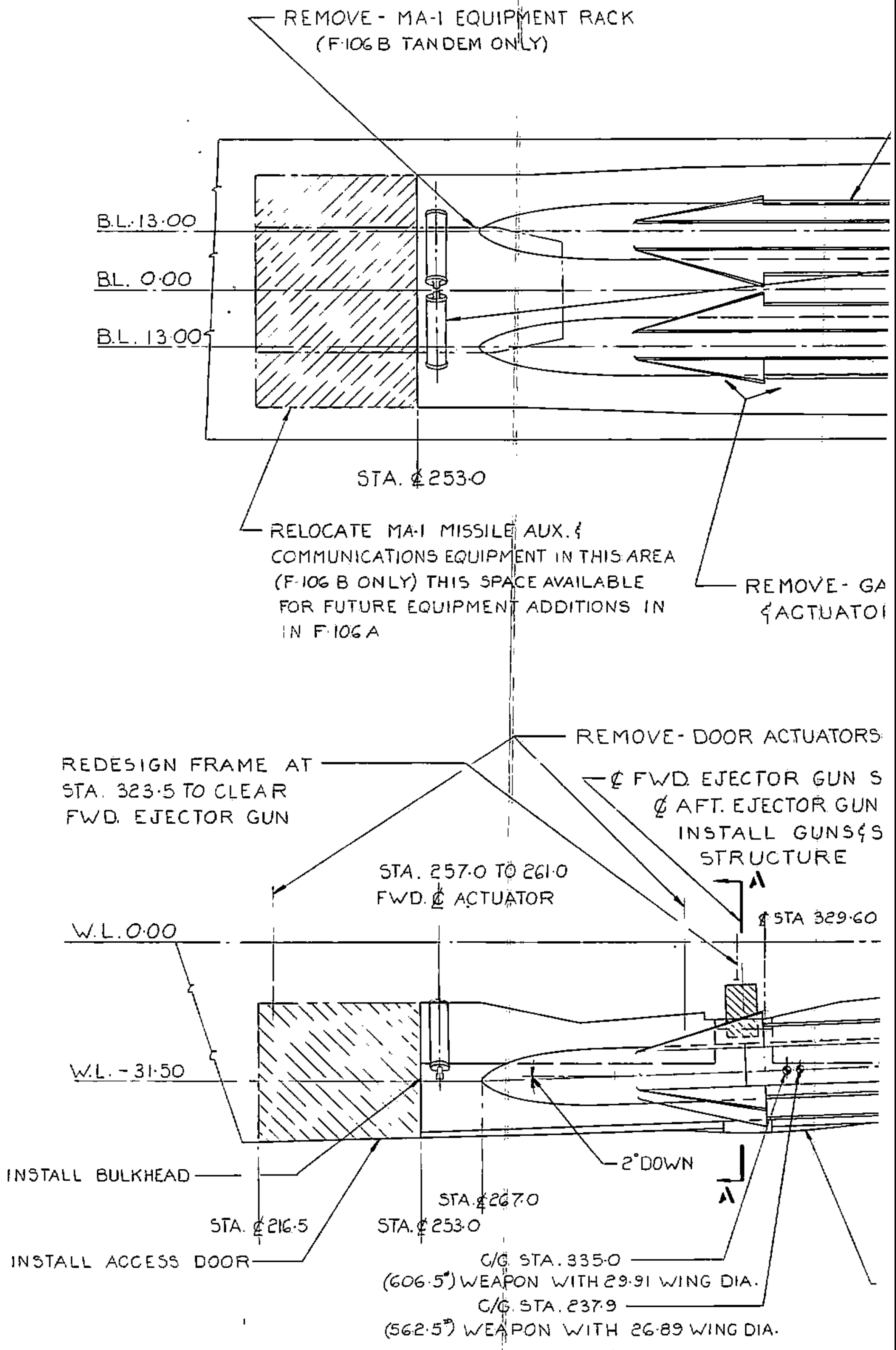

VIEW C

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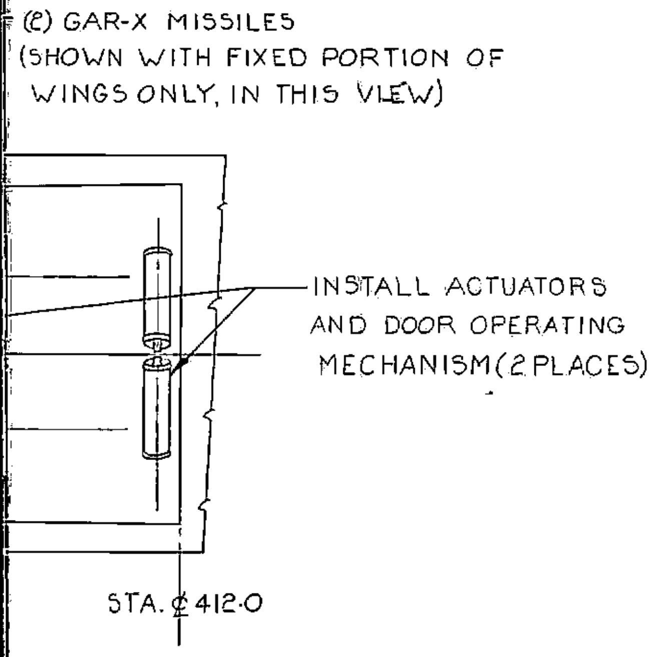
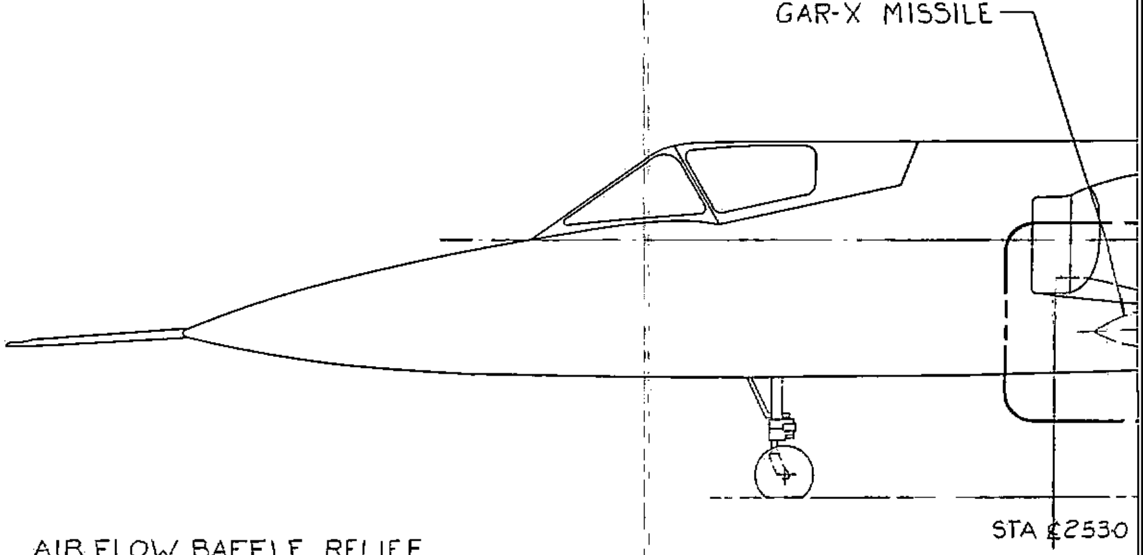
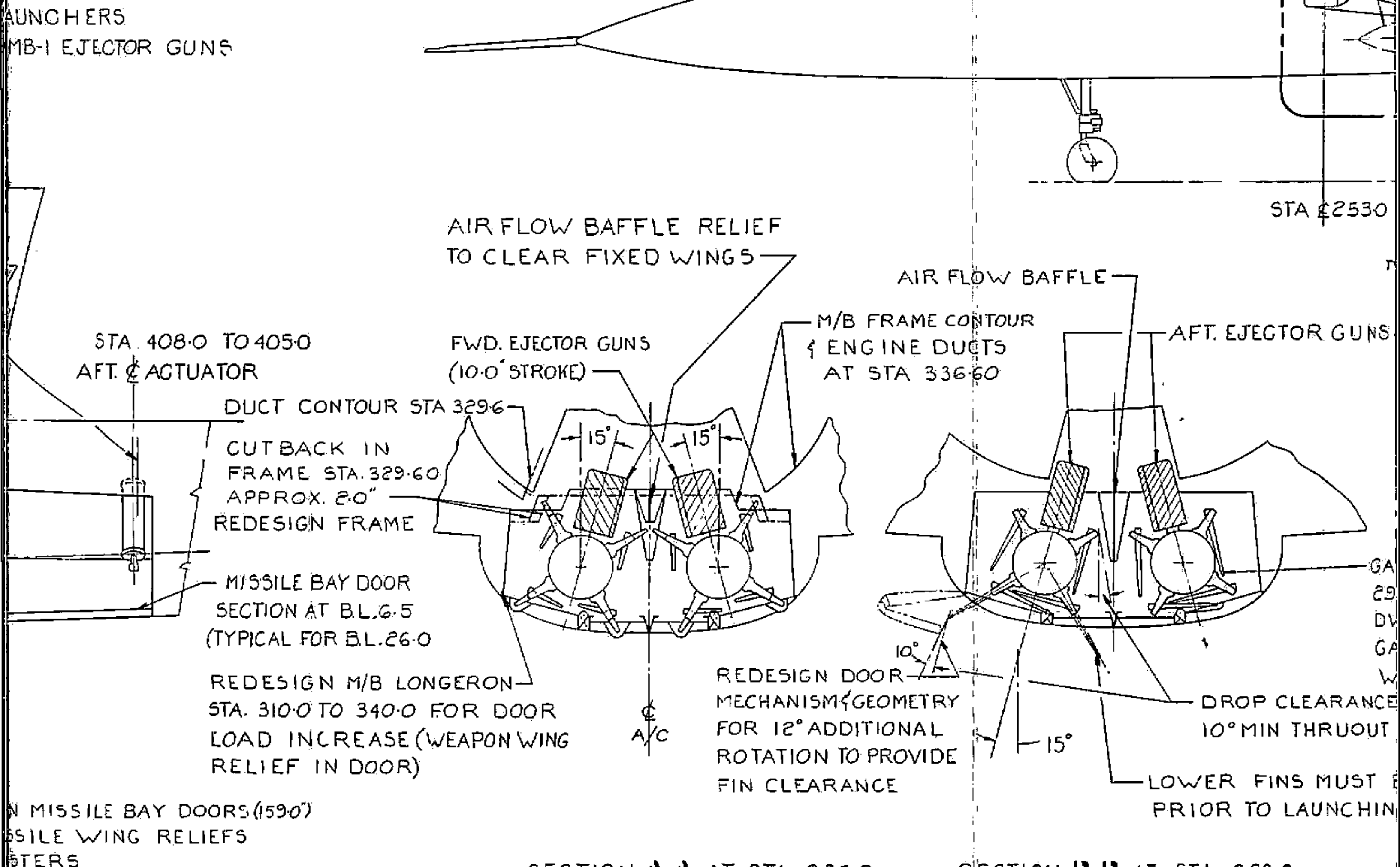
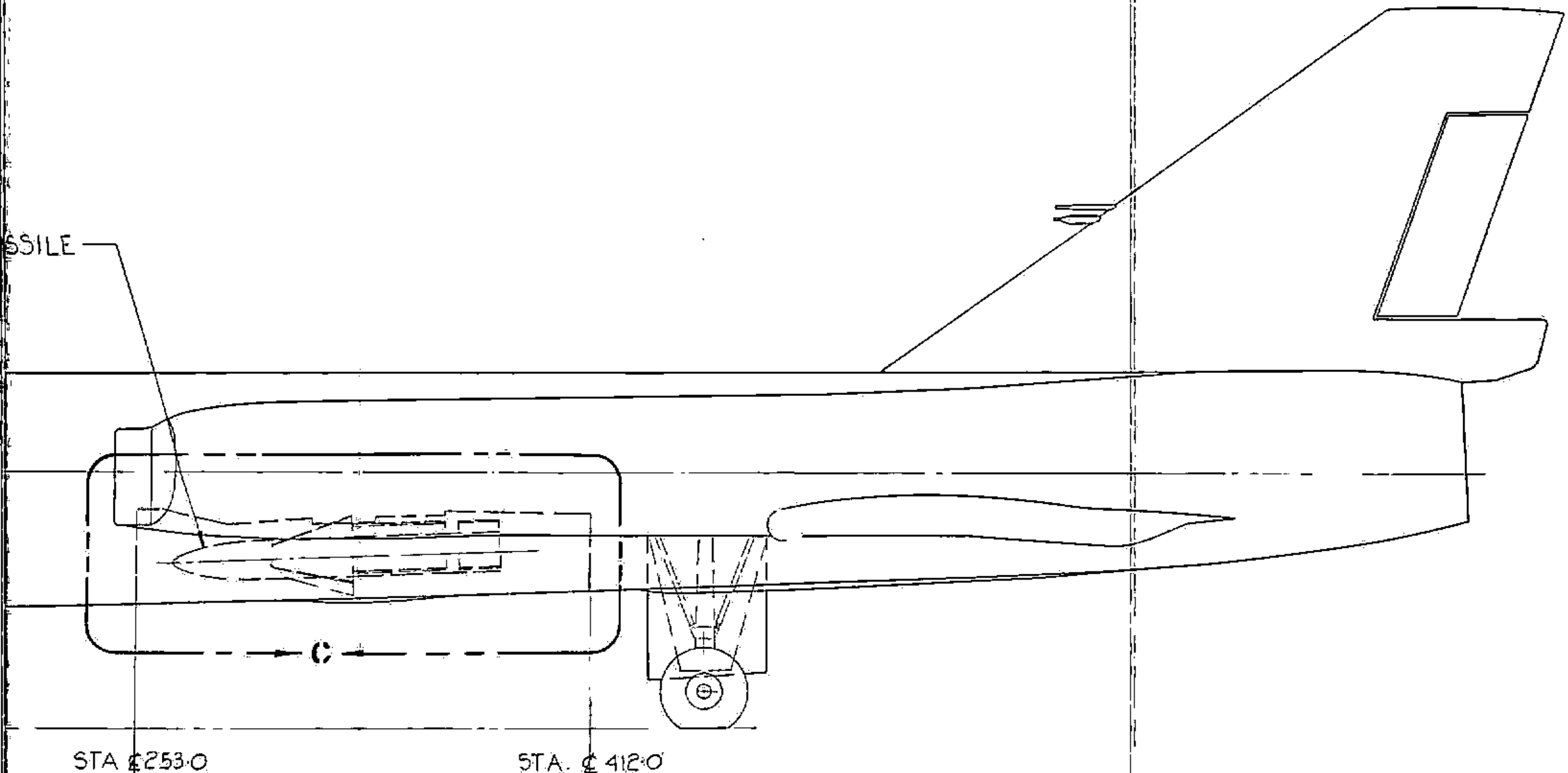






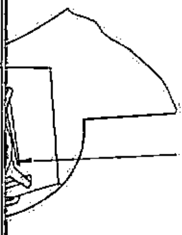
SECTION **A-A** AT STA. 326.0
LOOKING FWD.

SECTION **B-B** AT STA. 369.0
LOOKING FWD.



SIDE VIEW
TWENTIETH SCALE

EJECTOR GUNS (10.0 STROKE)



GAR-X MISSILE (606.5") WITH
29.91 FIXED WING DIA (HUGHES
DWG. NO. SP 30876 SHOWN STOWED
GAR-X MISSILE (562.5") WITH 26.89 FIXED
WING DIA. (HUGHES DWG. NO. SP 30875)

ROP CLEARANCE
0° MIN THRUOUT

ER FINS MUST BE EXTENDED
R TO LAUNCHING

369.0
D.

SEE DWG SD-56-08059 FOR INST'L
OF 133.5" GAR-X

[redacted]

CONVAIR

PRELIMINARY DESIGN DRAWING

STUDY - INSTL. OF (2) GAR-X
MISSILES (113.9" LONG) (2 WING TYPES)
IN F-106 A AIRPLANE

Kosniak	CHECKED	APPROVED	SCALE TENTH / NOTED	DATE 8/16/56

CONVAIR - SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION
SD-56-08060

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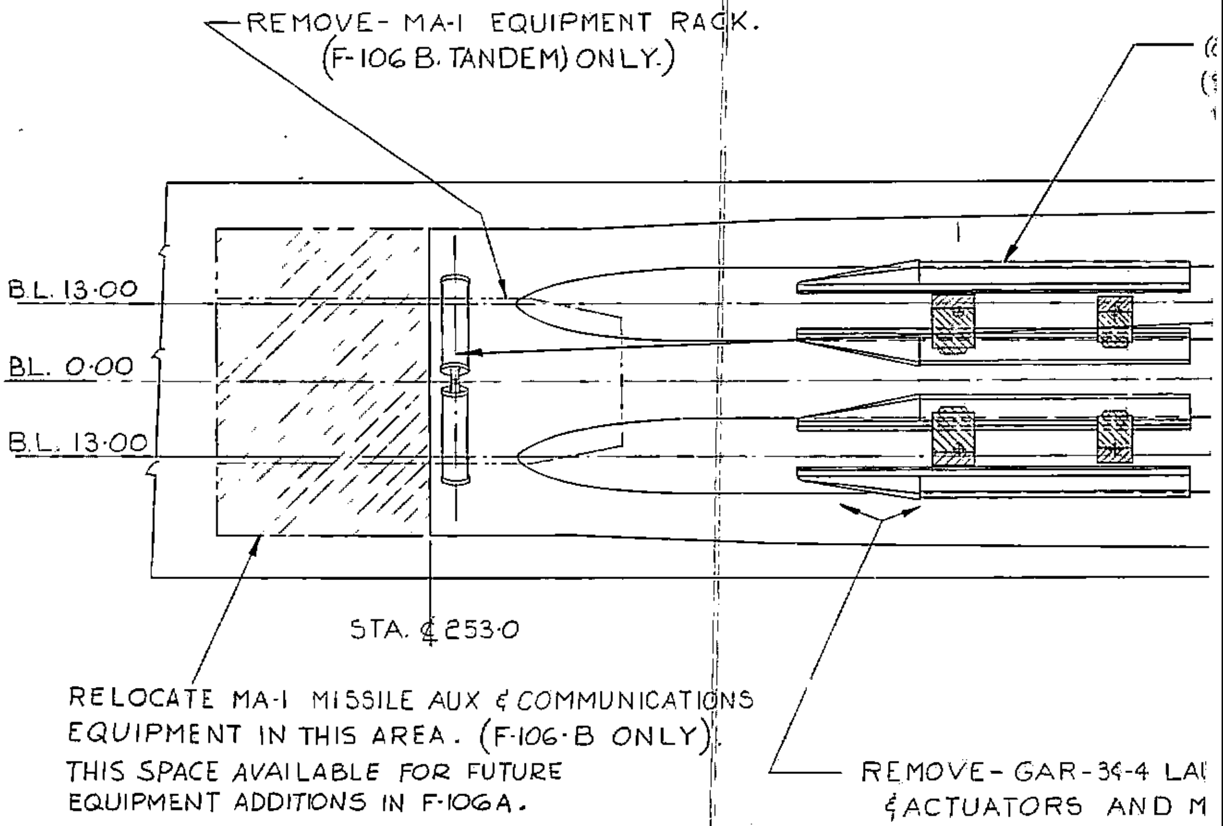
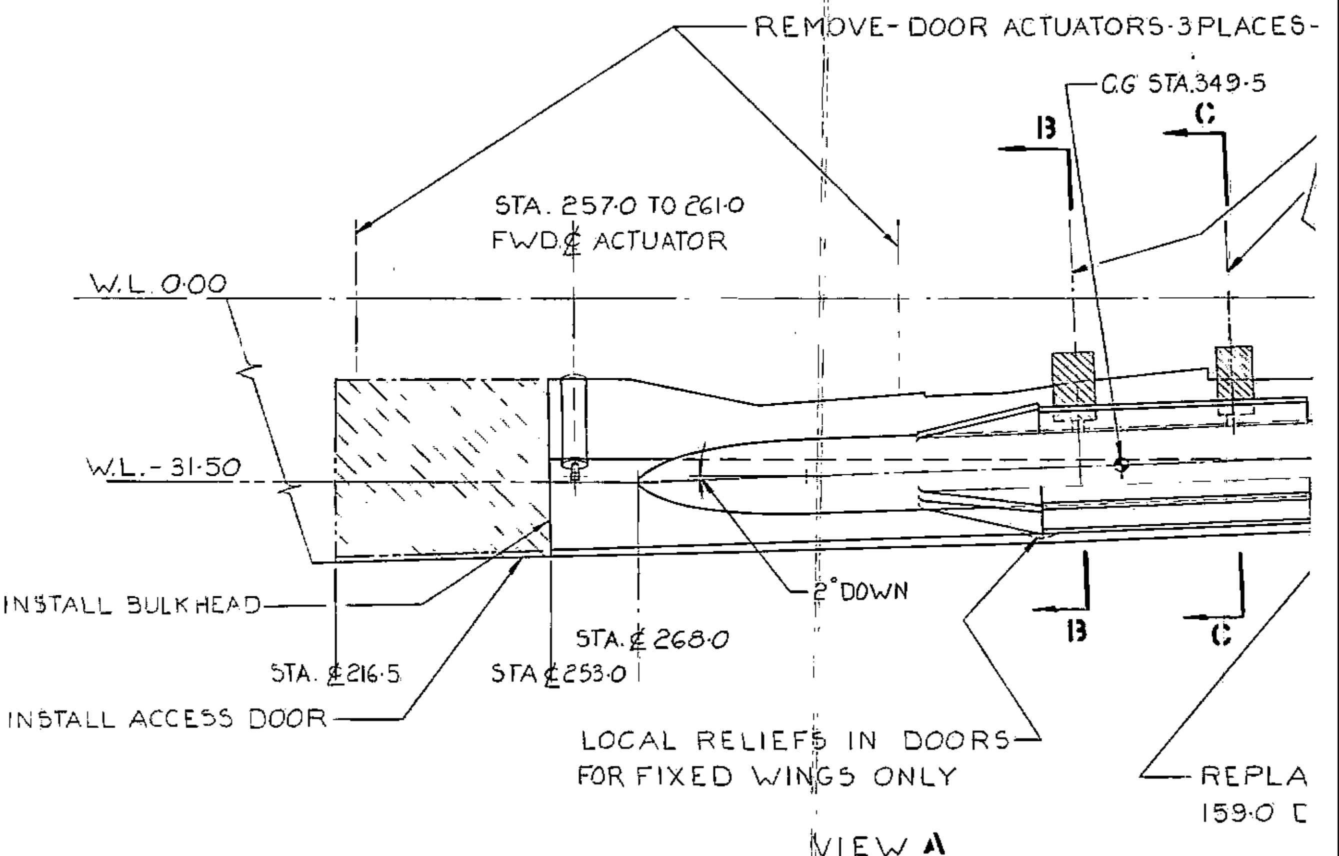
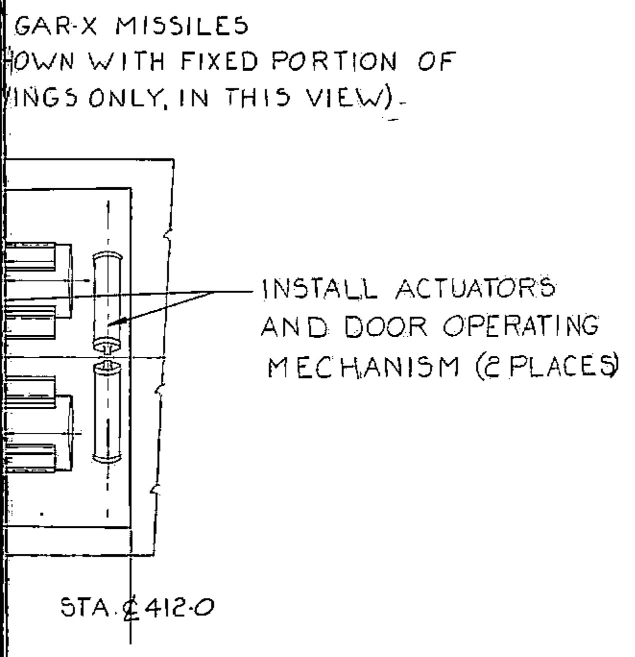


REMOVE- MA-I EQUIPMENT RACK. (F-106 B. TANDEM) ONLY.)
B.L. 13·00
B.L. 0·00
B.L. 13·00
STA. ₵ 253·0
RELOCATE MA-I MISSILE AUX & COMMUNICATIONS EQUIPMENT IN THIS AREA. (F-106-B ONLY). THIS SPACE AVAILABLE FOR FUTURE EQUIPMENT ADDITIONS IN F-106A.
REMOVE- GAR-3&-4 LA
&ACTUATORS AND M

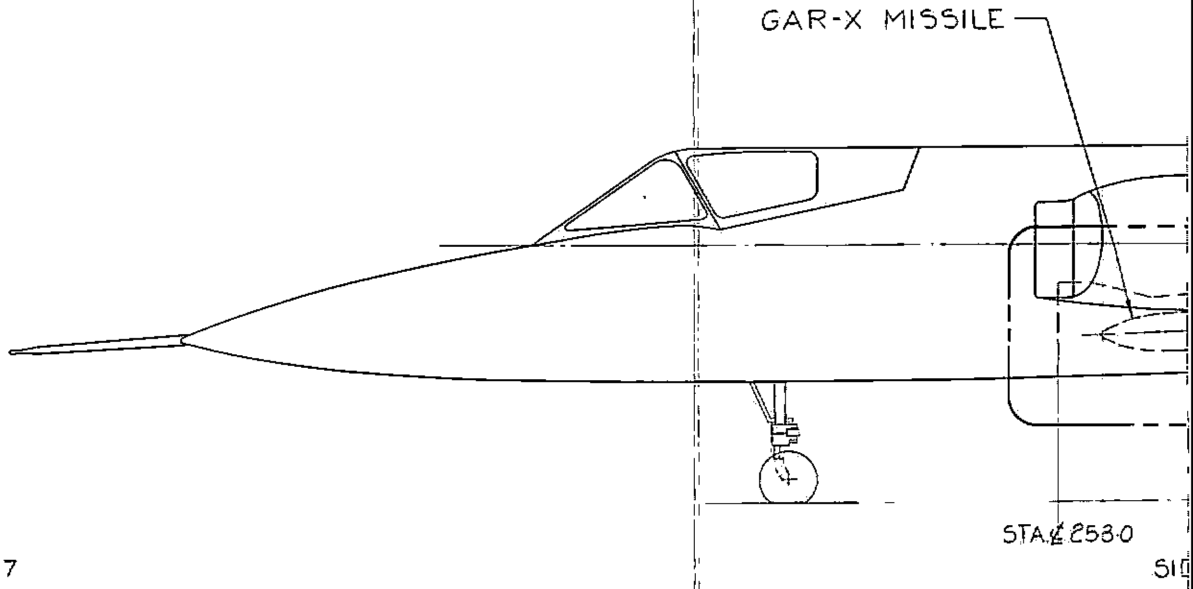
PLAN VIEW

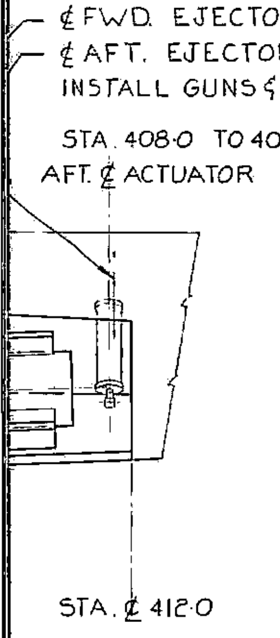


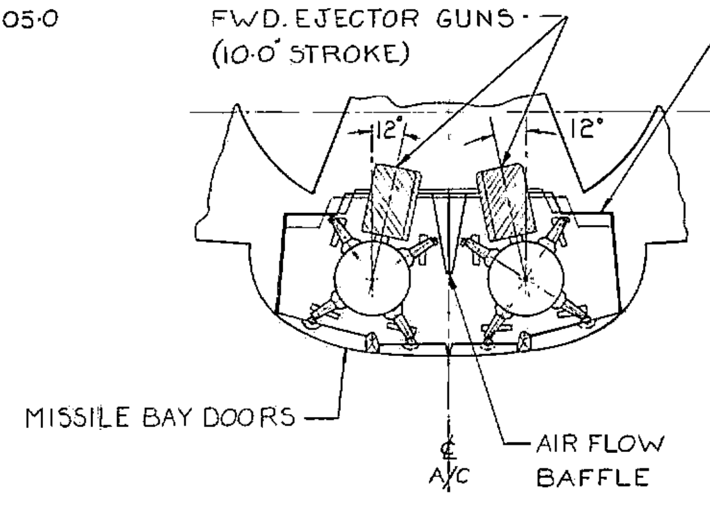
VIEW A



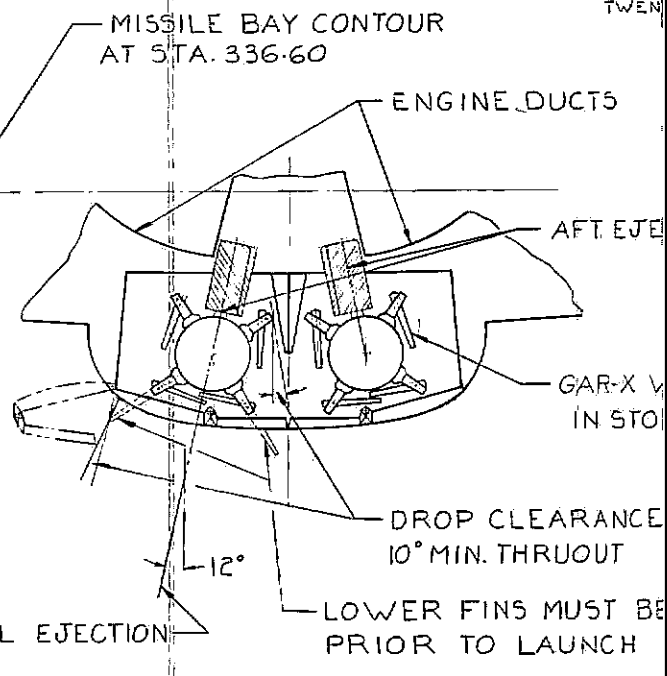
CHERS
EJECTOR GUNS







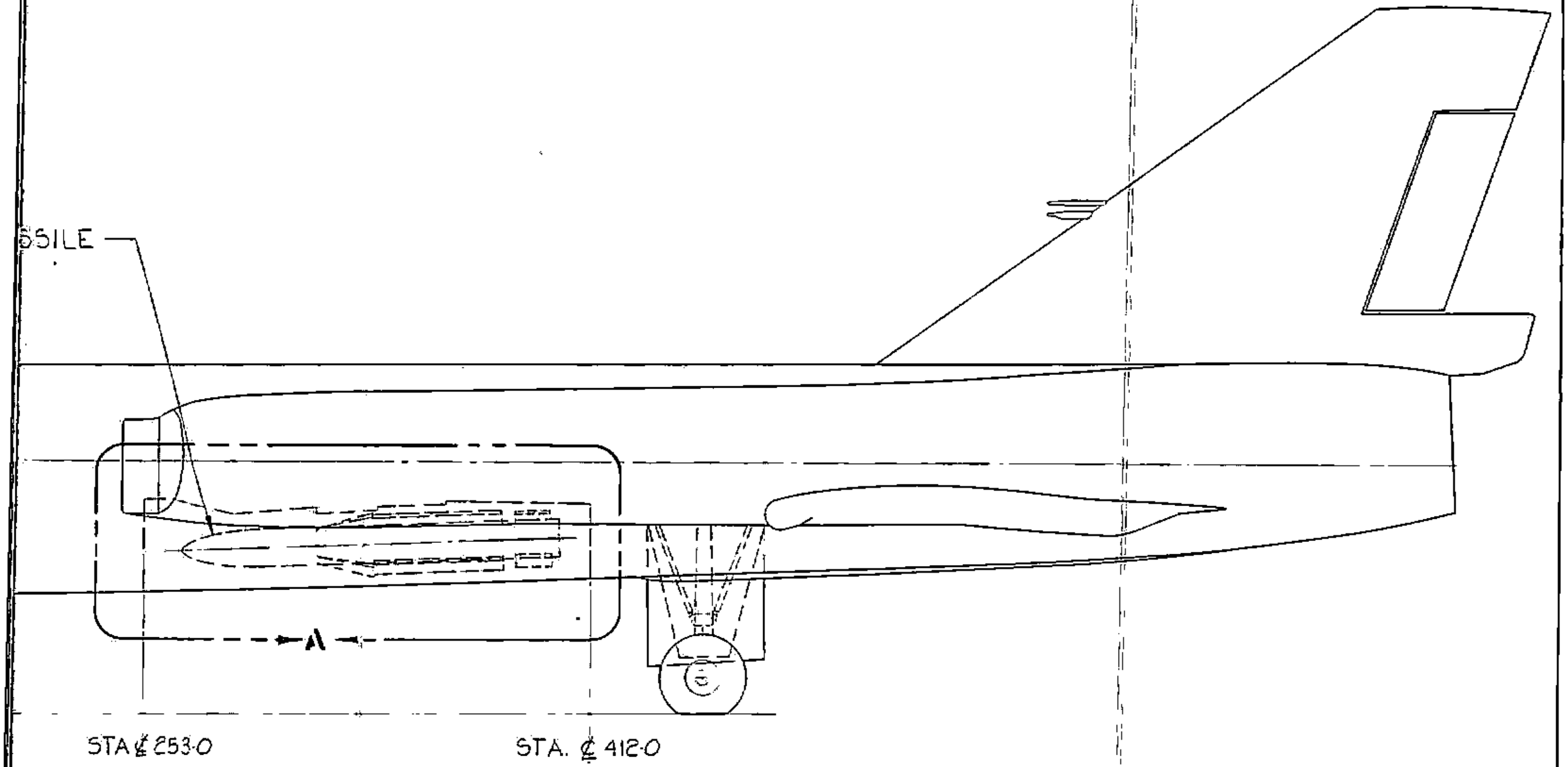
SECTION B-B AT STA. 342.7
LOOKING FWD.



SECTION C-C AT STA. 369.0
LOOKING FWD.

-195.5 DOOR WITH
R

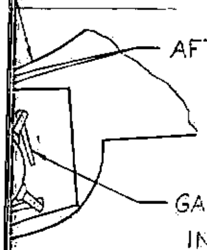
Z-P-8-044
Page 12



SIDE VIEW
TWENTIETH SCALE

OUR

ENGINE DUCTS



SEE DWG SD-56-08060 FOR
INSTL. OF 113.90" GAR-X

DROP CLEARANCE
0° MIN. THRUOUT

ER FINS MUST BE EXTENDED
OR TO LAUNCH

A. 369.0
D.

CONVAIR
PRELIMINARY DESIGN DRAWING
STUDY - INSTL. OF (2) GAR-X
MISSILES (133.5" LONG)
IN F-106-A AIRPLANE

DR Kozinak	CHECKED		SCALE TENTH (NOTED)	DATE 8/9/56

CONVAIR - SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION
SD-56-08059

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A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

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[redacted]

REPORT ZP-8-057
DATE 21 February 1958
MODEL F-102A

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by: Dept. Date
L. G. Avritt 130-1 4-4-67

Classification Changed to:
UNCLASSIFIED
Author 12-12-66
D EX.
Reclass : Dept. Date
130-1 4-4-67

TITLE

STUDY

INSTALLATION OF GAR-1Y WEAPONS IN THE F-102A AIRPLANE

ZP-8-057

PREPARED BY R. E. Dietz
R. E. Dietz

GROUP Armament and Preliminary Design

REFERENCE APPROVED R R Wahler
R. R. Wahler,

CHECKED BY R. C. Lingley
R. C. Lingley

APPROVED BY A. R. Lambert
A. R. Lambert
Chief, Pre-Design

NO. OF PAGES 8
NO. OF DIAGRAMS 2

REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED
1.	5/1/58		Firing sequence change	
2.	5/13/58		Chg. Page 2 as noted (A)	2

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FORM 1612 A-4

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INTRODUCTION

This study shows changes required to install two GAR-1Y missiles in the center bay of the F-102A airplane. The four GAR-1D/-2 Falcons in the outboard bays will be retained, and (24) 2.75" Rockets will be removed.

The existing six Falcon capability will be retained with provisions to install either the GAR-1Y or Falcons in the center bay interchangeably.

The study is based upon a modification type change to be accomplished during an IRAN program at Convair's Palmdale facility, and/or by kits in the field.

A portion of this change is being accomplished on aircraft No. 1799 as a prototype for Hughes Aircraft Company. Early flight testing accomplished by Hughes will furnish Convair some data so that a limited amount of flight testing will be required of Convair's production configuration described herein.

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SUMMARY

The changes required to an F102A airplane to carry two GAR=1Y missiles and four Falcon missiles can be accomplished with no structural changes except for the center missile bay door redesign.

Minor revisions to the armament and related systems will be required.

The two pass airplane capability will be retained. The airplane will have the following interchangeable armament loading capability.

1. (2) GAR-1Y + (4) GAR-1D/-2, or
2. (6) GAR-1D/-2
   (2.75" rockets will be deleted)

The existing four Falcon missiles and their displacing gear in the outboard missile bays will be unchanged.

The existing displacing gear in the center bays will be unchanged except for substitution of larger diameter actuators, displacing gear drag members, cross beam and pivot pin with strengthened like members. A
Rockets and rocket provisions will be deleted from the center doors. Rocket tubes will be retained in outboard doors to minimize the change.

With the proposed armament load, the normal take-off gross weight will be 262 pounds less than the present tactical F102A. There will be a slight C.G. shift which will remain within acceptable limits.

For the extended range mission the take-off gross weight will be 136 pounds less than the current airplane. This includes seven dummy rockets (126 lbs.) carried in the outboard rocket tubes to maintain the C.G. within limits.

The changes in gross weight and C.G. will have a negligible effect on airplane performance.

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DESCRIPTION

To minimize changes to the pneumatic and electrical systems, it has been assumed that the following ground rules would apply:

1. Fire control system and the sequence of firing will not change.
2. For the GAR-1Y installation the crash loading requirement will be reduced to a value less than the presently established 8 g's.

The following changes must be made to provide for two GAR-1Y missiles in the F-102A airplane while retaining four Falcon missile provisions in the Forward and Aft Side Bays, and removing provisions for (12) 2.75" rockets in the center bay doors.

1. General (Reference Drawing 8-05547)
   - A. Replace (2) center bay Falcons with (2) GAR-1Y missiles.
   - B. Redesign center missile bay doors, retaining hinge points and actuator attachment as for original F-102A airplane. Relocate marker beacon antenna.
   - C. Remove center missile bay door 2.75" rocket capabilities.
   - D. Replace (2) center bay armament displacing gear actuators with (2) actuators of larger diameter for displacing heavier armament load.
   - E. Electrical change to provide special weapon indicator light plus control switch in cockpit.
   - F. Replace center bay displacing gear drag members, cross beam and pivot pin with strengthened like members due to increased drag of the GAR-1Y.
2. Armament Group
   - A. Redesign center missile bay doors to provide clearance for two GAR-1Y missiles.

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B. Revise armament installation to install two new center bay displacing gear actuators.

3. Electrical Group

A. Create modification drawings to revise:

(1) Schematic diagram

(2) Circuit diagrams

(3) Harnesses

(4) Connection diagrams

4. Pneumatics Group

A. Displacing cylinder specification

B. Control valve specification

C. Schematic diagram

Schedules and costs are shown under separate cover.

Weight summaries for normal and extended range missions are shown on the following pages.

A C.G. curve for the extended range mission is also shown.

NOTE: For training purposes, it will be possible to load and fire (12) 2.75" rockets in the outboard missile bay doors. (12) rockets weigh 216 lbs.

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F-102A WEIGHT SUMMARY WITH (2) GAR-1Y MISSILES IN CENTER BAY

(4) GAR-1D OR GAR-2 MISSILES IN SIDE BAYS - NO ROCKETS

	Weight
Wing Group	3,000
Tail Group	535
Body Group	(3,433)
Alighting Gear Groups	1,056
Surface Controls Group	413
Engine Section	39
Propulsion Group	(6,361)
Engine Installation	4,995
Air Induction System	693
Exhaust System	140
Cooling System	16
Lubricating System	24
Fuel System	394
Engine Controls	23
Starting System	76
Fixed Equipment Group	(4,345)
Instruments	141
Hydraulics and Pneumatics Group	319
Electrical Group	594
Electronics Group	2,174
Armament Group	554
Furnishings and Equipment Group	227
Air Conditioning and Anti-Icing Group	259
Auxiliary Gear Group	77
WEIGHT EMPTY	19,182

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F-102A WEIGHT SUMMARY (Continued)

		Weight
Useful Load - Normal Tactical Mission		(8,706)
Pilot, Survival Kit and Parachute		275
Fuel, Unusable (JP-4) 23.7 Gallons		154
Fuel, Usable (JP-4) 1085 Gallons		7,053
Oil, Trapped and Engine		62
Missiles		
GAR-1D and/or GAR-2 Side Bays	(4)	508
GAR-1Y Center Bay	(2)	432
Missile Launchers	(6)	222
NORMAL TAKE-OFF GROSS WEIGHT		27,888

Current F-102A Gross Weight is 28,150 Lbs. based on Report 31C dated 28 January 1958

USEFUL LOAD - Extended Range Mission		
Pilot, Survival Kit and Parachute		275
Fuel, Unusable (JP-4) 23.7 Gallons		154
Fuel, Usable (JP-4) Internal (1085 Gallons)		7,053
Fuel, External (JP-4) 430 Gallons, Tanks, and Pylons		3,126
Oil - Trapped and Engine		62
Missiles		
GAR-1D and/or GAR-2 - Side Bay	(4)	508
GAR-1Y Center Bay	(2)	432
Missile Launchers	(6)	222
Ballast - Dummy Rockets - Side Doors	(7)	126
EXTENDED RANGE MISSION TAKE-OFF GROSS WEIGHT		31,140

Current F-102A Extended Range Gross Weight is 31,276 Lbs.

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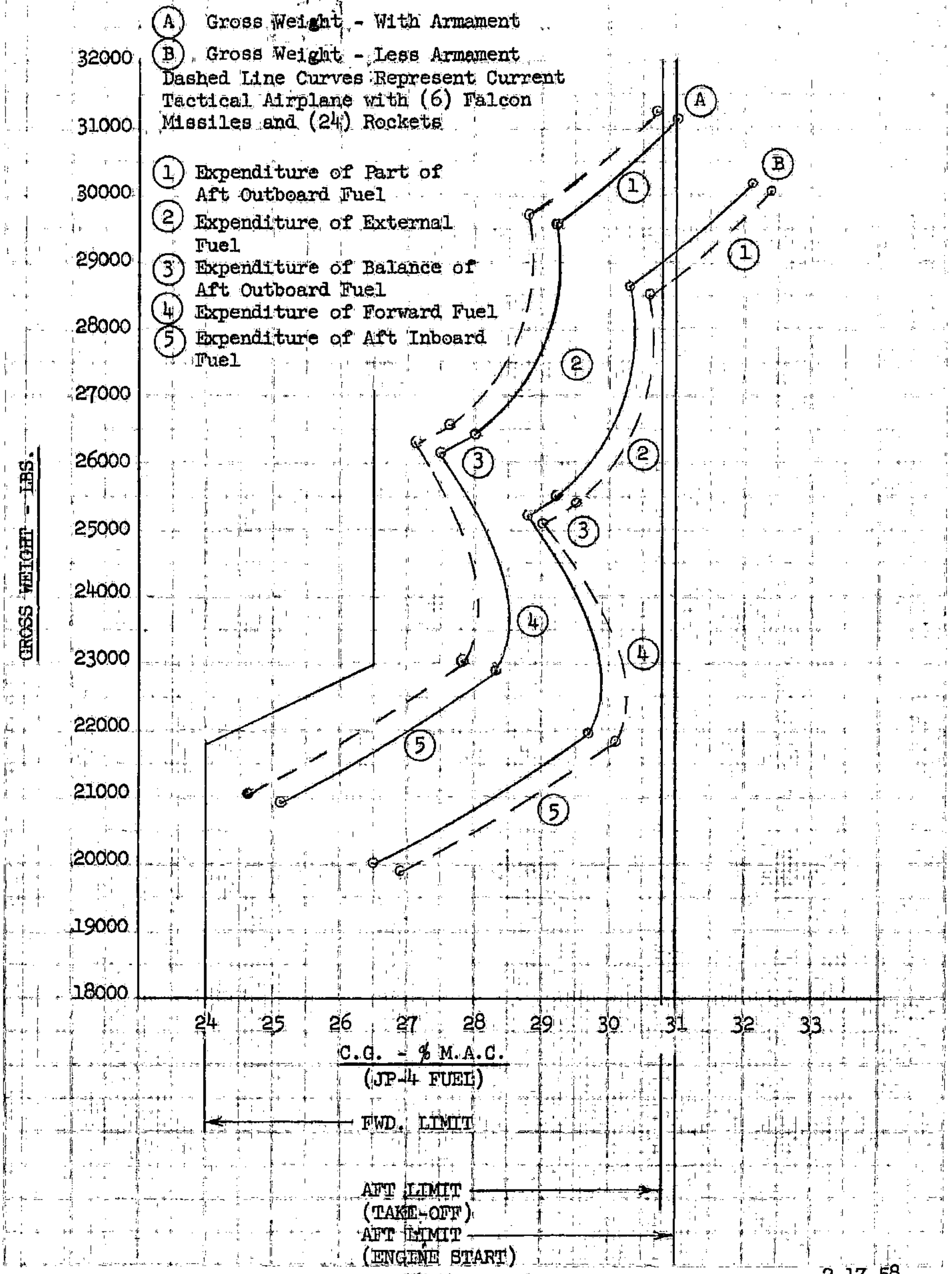
F-102A AIRPLANE WITH (2) GAR-1Y & (4) GAR-1D &/OR GAR-2 MISSILES - EXTENDED RANGE MISSION



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CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-053
DATE 9 July 1957
MODEL F-102A

UNCLASSIFIED

TITLE

" INFRARED AUGMENTATION FOR THE F-102A"

ZP-8-053

July 9, 1957

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UNCLASSIFIED
uthorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
eclassified by:
L. B. Avitt
Dept. 130-1 Date 4-4-67 [redacted]

Classificat Changed to:
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Auth -12-66
Recla
Dept. 130-1 Date 4-4 67

PREPARED BY F.L. Tripp
F.L. Tripp , P.D
E. Morrell
E. Morrell, O.A.

F.L. Tripp
Edited By F. L. Tripp

GROUP Preliminary Design

REFERENCE

APPROVED BY A. R. Lambert
A.R. Lambert

Group Approvals M. Kantor
M. Kantor , O.A.
R. E. Honer
R. Honer , E.E.L.

NO. OF PAGES 26

NO. OF DIAGRAMS 15

REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED
1	12/2/57	F. Tripp	Added Addendum #I to Report.	i-1 thru i-7
	"	"	Section 7.110, Para. 2	Page 16
	"	"	Section 7.120, Para. 3	Page 17
	"	"	Section 7.134 and 7.135	Page 20
	"	"	Figure 10	

FORM 1612 A-4

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TABLE OF CONTENTS

		Page
1.	Objectives of Infrared System Addition	4
2.	System Characteristics	8
3.	Installation	10
4.	Proposed Flight Test	13
5.	Schedule	15
6.	Recommendations	15
7.	System Descriptions	16
8.	Figures	17
9.	Addendum I	i-1
10.	Figures 1 and 2 of Addendum I	i-6

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ANALYSIS
PREPARED BY
CHECKED BY
REVISED BY

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
(SAN DIEGO)

PAGE i-1
REPORT NO. ZP-8-053
MODEL F-102A
DATE

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Modified Infrared Display Unit for Aerojet System

Addendum #1 to

Rep. ZP-8-053

Pages of ZP-8-053 effected:

Page 16, Section 7.110, Para. 2

Page 17, 18, Section 7.120, Para. 3

Page 20, Section 7.134 and 7.135

Figure 10

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INTRODUCTION

Additional investigations after submittal of Convair report ZP-8-053, "Infrared Augmentation for the F-102A" indicated that the infrared display unit, which was injected into the optical sight, protruded into the pilot escape ejection path. The protrusion was sufficient to cause serious injury to the pilot.

Aerojet was therefore requested to study possible relocation of the infrared display unit. Subsequent discussions with Aerojet formulated the following requirements:

1. Display mechanism located forward of the instrument panel.
2. Optical projection of the display onto a retracting screen such that visibility from the cockpit at landing was not restricted;
3. Combination of the acquisition and tracking display units into a single device. Aerojet has formulated a display system meeting these requirements.

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MODIFIED DISPLAY UNIT-AEROJET

The mechanism of the display is identical to that described in ZP-8-053. The acquisition display previously considered to be a panel mounted instrument, retains the direction to fly concept but is displayed around the circumference of the modified display unit. An arrow is presented to give a direction to fly by means of a synchronized rotating disk and xenon lamp system.

The tracking display previously injected into the optical sight, presents a cross, the center of which indicates the position of the target, inside the acquisition circle.

Tactical application and the display during each phase is shown in Figure 1.

The combined display unit, shown in Figure 2, contains the functions for both the acquisition and tracking units. Light from each of the xenon lamps passes through a coding scanner, through the beam-splitting glass, through the optical attenuator, and through the field projecting lens, and generates an image of the cross and the arrow on the viewing glass, which has a light-diffusing surface. This glass contains the missile field indicator. When the cross appears within the indicator circle, the missile is in angular position to acquire the target.

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The viewing glass and the supporting tube will be provided within a mechanism which will allow them to be folded, so that the pilot will have an unrestricted field of view during landing operations. Brightness of the display is varied by an optical attenuator which is continuously variable, with a mechanical control available to the pilot.

Equipment Characteristics

The display unit performs the following functions:

1. Receive signals from acquisition amplifier.
2. Trigger a xenon lamp in phase with target indications signal.
3. Code light from xenon lamp by means of moving arrow to represent equivalent target direction.
4. Project arrow onto viewing glass.
5. Receive signals from tracker amplifier.
6. Trigger a xenon lamp in phase with target indicator signal.
7. Code light from xenon lamp by means of morning slits to represent equivalent target position.
8. Project cross onto viewing glass for viewing by pilot.

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Factors of interest are:

1. Weight - 7#
2. Controls -
    a. A mechanical control will be provided to adjust display intensity.
    b. A mechanical control will be provided to fold the viewing glass and support tube out of the way when required.

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SYMBOLS AND ABREVIATIONS

FCS	Fire Control System
GAR-2	Infrared Falcon
P_R	Reliability
P_H	Probability of hitting target
P_{HK}	Probability a hit is a kill
P_K	Kill probability
P_{DC}	Probability of detection and conversion
Ro	Initial detection range
σ	Vectoring error
θ	Half angle of target cone of radiation
σ_R	Ground control launch range error

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[redacted]

RESERVATION OF PROPRIETARY RIGHTS

The companies that have submitted proposals (Aerojet-General, Avion and Hughes) on which this report is based, have considered their information to be proprietary and that such information, data and prints as disclosed were to be used for United States Government evaluation, and that this information shall not be divulged to third parties.

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[redacted]

0.00 SUMMARY

Three infrared systems for augmentation of the F-102A fire control system are presented and discussed.

1. Aerojet Infrared Target-Position Indicator.
2. Avion Passive Detection System.
3. H.A.C. Infrared Augmentation System.

Addition of an infrared detection system to the fire control system results in an increase of kill capability to 0.60 from zero at night above 20,000 feet with the fire control system jammed or inoperative and to 0.60 from less than 0.30 in daylight above 20,000 feet with the fire control system jammed or inoperative. At low altitude in clear weather, the infrared system can provide greater detection ranges than the optical sight or a radar set with ground clutter. Due to the narrowness of the GAR-2 launch range, however, range information must be derived from the radar when the clutter cross-over range is obtained or from the optical sight. Ground information is not sufficient.

Due to rigid mount and unobstructed field-of-view requirements, the equipments have been shown mounted as follows:

1. Aerojet Tracker on left wing at inboard fence. Search on right wing at outboard fence.

2. Avion-one on each wing at inboard fence.

3. Hughes-one unit on tail below omni range antenna.

Methods of presentation of target information to the pilot are as follows:

1. Aerojet Tracker-Illuminated cross on combining glass attached to optical sight. Search-Arrow direction to fly on panel instrument.

2. Avion - A dot representing the target is presented on the optical sight combining glass. The outer portions of the search field are compressed indicating a direction to fly while the center [illegible] target azimuth and elevation relative to [illegible]

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[redacted]

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2. Avion - A dot representing the target is presented on the optical sight combining glass. The outer portions of the search field are compressed indicating a direction to fly while the center portion presents target azimuth and elevation relative to the interceptor.
3. Hughes - Alternate display on radar scope.

[redacted]

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Schedules for the Aerojet and Avion systems show initial production units will be available 9 months form "go-ahead". Hughes units because of additional integration, will be available 18 months from "go-ahead".

No significant operational differences exist among the systems. The same type detector and about the same size optics are used in each system.

The Hughes system, although the most completely integrated, is less desirable because of the extended schedule.

No single factor determines the selection of either the Aerojet or Avion system, and no series of factors selects either system.

The Aerojet system is recommended because of:

a) less development risk due to previous experience of installation.

b) display method has been well investigated and defined.

c) Aerojet system has been installed and demonstrated in flight operation.

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0.10 INTRODUCTION

Three infrared systems (Aerojet Infrared Target - Position Indicators, Avion Passive Detection System, and H.A.C. Infrared Augmentation System) for augmentation of the F-102A fire control system are presented and discussed.

Factors that are included in the discussion are:

(1) Objectives of infrared system addition indicate areas where capabilities of the aircraft can be increased.

(2) System description, including system operation, physical data and manufacturer's preliminary data.

(3) Installation of the equipment and the problems associated with each equipment.

(4) Requirements and objectives of a flight test program to evaluate the objectives of the system addition.

(5) Manufacturer's delivery schedule.

(6) System recommended for installation on the F-102A.

(7) Detailed system description.

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1. OBJECTIVES OF INFRARED SYSTEM ADDITION

1.00 Introduction. The F-102 FCS (Fire Control System) has limited effectiveness under the following situations:

1. No kill capability at night or in inclement weather when the FCS is inoperative, jammed or cluttered.

2. A single pass kill capability in daylight during good weather at medium altitudes when the FCS is inoperative or jammed.

3. A limited two pass kill capability in daylight during good weather at low altitudes when the FCS is cluttered.

The most serious deficiency exists at night with an inoperative or jammed FCS since there is no kill capability. An attack requires that the pilot receive target azimuth and elevation angles, and range. During daylight with good visibility, the optical sight provides this information with sufficient precision to use the GAR-2 Infrared Falcon or the 2.75" rockets as the attack weapons.

At night with clear weather conditions, an infrared detection system can present the required angular information to launch the GAR-2 Falcon. Range information must be obtained from another source such as ground control (see para. 1.10). The GAR-2 seeker lock-on range cannot be used to determine launch range as the GAR-2 seeker lock-on range exceeds the missile aerodynamic launch range (Figure 1.). Further, since the GAR-2 is carried internally, no assurance is given to the pilot that lock-on (in direction) will occur.

The 2.75 rockets cannot be fired at night by an infrared sight because of the stringent launch and angle information required to launch the rockets.

1.10 Proposed Alternate Mode. The alternate mode, for night attack using the infrared sight, uses the GAR-2 Falcon as the attack weapon. Appropriate parameter adjustments will permit the use of the Sidewinder or similar missiles.

In the alternate mode, the pilot will be vectored by ground control on a tail approach to the target. The pilot will search for the target with the infrared search and track system. On detection, the system starts tracking the target and the pilot informs the ground that he is

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tracking the target. The pilot then flies a pursuit course from the angular information presented by the infrared display unit.

Assurance to the pilot that the missile seeker will lock-on (in range only) may be derived from the infrared system tracking signal. If the signal level required for tracking is greater than or equal to the signal level required by the GAR-2 to track, then this is sufficient lock-on indication. If the signal level is less, then the infrared detection system must indicate when the signal level is adequate for GAR-2 lock-on.

During the pursuit course closure, the ground controller notes the separation and starts to sense the closure rate. The launching signal will be given to the pilot by the controller through one of the following:

1. If the ground controller sees on a sweep of his radar scope that the range is within allowable launch range, he will inform the pilot to launch, or

2. The controller on estimating the range and closure rate may inform the pilot to "launch in X seconds".

The alternate is for the pilot to withhold launch until system indication denotes that lock-on can occur. However, this does not insure launch within the aerodynamic range of the missile.

1.20 Evaluation of Alternate Mode of Attack. The following constants for the GAR-2 will be used in the evaluation.

(P_R) Reliability = 0.85/missile

(P_H) Probability of hitting target = 0.70/missile

(P_{HK}) Probability a hit is a kill = 0.90/missile

(P_K) Kill probability = $P_R P_H P_{HK}$ = 0.53/missile

P_{K_3} (Three GAR-2 Falcons) = 0.90

1.21 P_{DC}. The probability of detection and conversion is assumed to depend only upon the accuracy of ground vectoring and the half angle of the cone of radiation from the target.

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With expected closing rates approaching 300 fps and a 4 second infrared frame time, detection will occur within 10 seconds from the time the interceptor is correctly positioned. Conversion to a pursuit attack should follow immediately. Figure 2 is a plot P_{DC} versus R_o (R_o = initial detection range) for two vectoring errors ($\sigma = 1, 3$ n. mi.) and for two half angles (Θ) of the target's effective cone of radiation ($\Theta = 10°, 30°$). Using the maximum or minimum P_{DC} to represent the system would not be realistic. Θ and σ are expected to decrease with time. At the time of system introduction, it is assumed that $\sigma = 3$ n. mi. and $\Theta = 30°$, and that their variation with time will maintain a constant value of $P_{DC} = 80\%$.

1.22 Vectoring Errors. The probability of determining the correct launch point depends on the controller's accuracy in estimating relative range and range rate. An accuracy in range measurement of ± 1/2 n. mi. is currently obtained. Launch range (separation) can then be determined within ± 1 n. mi. (σ_R). The alternate mode launch range will be considered to be 1 n. mi. less than the maximum launch ranges of Figure 1. Only those interceptors which lie outside the maximum launch range will have a P_L of zero. For $\sigma_R = 1$ n. mi., 84% of the attacking interceptors will achieve correct launches on ground supplied range information. ($P_L = 0.84$).

1.30 Final Evaluation of the Alternate Mode of Attack. P_K for the alternate mode is composed of these three factors:

$$P_{DC} = 0.80,\ P_L = 0.84,\ P_{K_3} = 0.90$$

$$P_K = P_{DC}\, P_L\, P_{K_3} = 0.60$$

1.40 Daylight Attack Capabilities. Daylight-modified infrared equipment can give greater detection ranges (tail aspect) than visual detection. Visual attention concentrated on the line of sight yields perception up to about 7.5 n. mi. Visual search of a 13-1/2° cone results in a 90% probability of detection at 3.9 n. mi. Infrared equipment discussed herein gives a minimum 9 n. mi. detection range on tail aspect.

1.50 Conclusions.

1.51 Increased P_K. The kill capability of the F-102A may be increased to 0.60 in the following areas (above 20,000 feet) by the addition of an infrared detection system and be exploiting the capabilities of the GAR-2 Falcon and the ground environment:

1. At night with FCS jammed or inoperative (no present P_K).
2. Daylight with FCS jammed or inoperative (poor Present $P_K < .30$).

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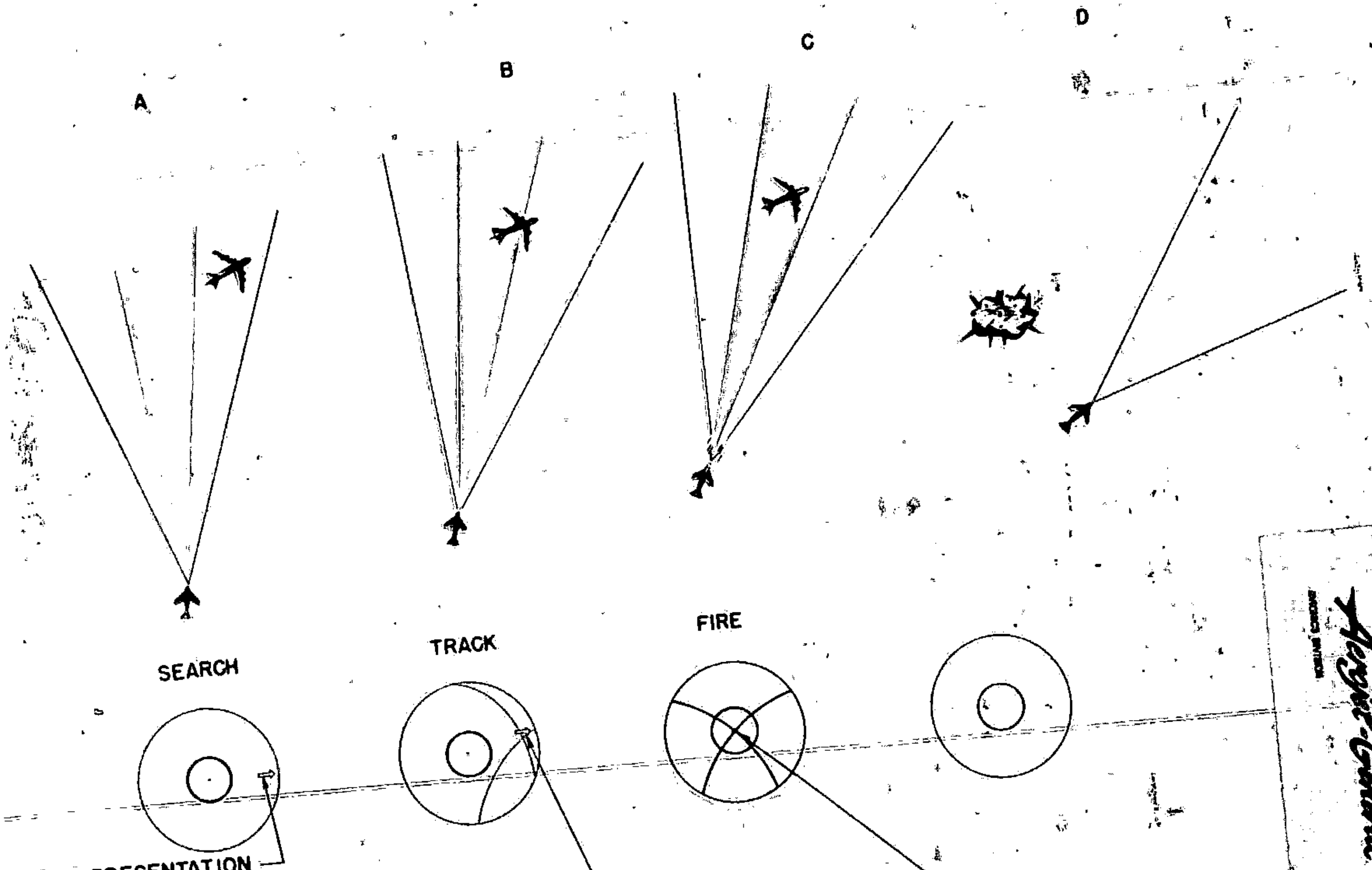
Aerojet-General CORPORATION
AVIONICS DIVISION
AZUSA, CALIFORNIA

FIG.1



ARROW PRESENTATION SHOWS DIRECTION OF TARGET.

CROSS PRESENTATION SHOWS LOCATION OF TARGET.

CROSS COINSIDES WITH CENTER OF FIELD.

TACTICAL APPLICATION OF AN/AAA-2 (MODIFIED FOR F-102)

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1.52 Additional Considerations.

1. Low Altitudes

a) At low altitudes where radar operation may be limited by ground clutter, kill probability can be enhanced by utilizing the IR equipment for detection and conversion followed by obtaining range information with radar when the clutter cross-over range is achieved.

b) If radar is jammed or inoperative some kill probability can be expected by utilization of IR equipment for detection and conversion in conjunction with the optical mode for range information at launch time.

c) The narrowness of the GAR-2 Falcon launch range tolerance from sea level to 20,000 feet requires that target range be more accurately known than the range that can be provided from ground information. Hence, when either radar or optical ranging is not possible, the IR detection system will not provide additional capability.

2. Countermeasures.

When normal radar tracking is denied by chaff, confusion repeaters track breaking repeaters or occasionally by broad hand noise jammers, the IR system can be utilized for detection conversion and angle tracking. Launch range information can be obtained from the radar system. Under conditions where broad band noise jammers succeed in denying radar range, then launch range information from CGI can be employed for targets above about 20,000 feet.

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2. SYSTEM CHARACTERISTICS

2.00 Systems Considered. The following systems have been considered for adding infrared search and track capabilities to the F-102A:

1. Aerojet Infrared Target-Position Indicators.
2. Avion Passive Detection System.
3. H.A.C. Infrared Augmentation System.

2.01 Introduction. The detector units of the equipment available for the F-102A consists of short wave length type, lead sulfide cells. Therefore, the capability of these equipments lies primarily in tail aspect approaches to the target where the hot portions of the engines are unshielded.

Range to target is not determined by these detectors. The target azimuth and elevation is presented relative to the interceptor to permit a pursuit course to be flown.

2.02 General Operation. The lead sulfide detector is chemically deposited on a substrate of glass. Electrodes, attached to the lead sulfide film, are coupled to a preamplifier so that when detector resistance changes due to infrared radiation, the resultant voltage change across the film is detected and amplified. This amplified signal is then processed into useable pilot information. The display signal can be presented through the optical sight or radar scope.

2.03 System Characteristics. General system characteristics are presented in the following tables. Specific system descriptions are presented in Section 7.

	TABLE I
Aerojet	Two distinct units are used. 1) The tracker scans a circle and performs the track function. 2) The acquisition unit, scans an annulus to cover an extended search area. (The present tracker will be modified to provide daylight discrimination, the acquisition unit has daylight discrimination).
Avion	Two identical units are used to provide suitable coverage. (Units have daylight discrimination.)
Hughes	A single unit is sufficient. (Unit has daylight discrimination.)

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TABLE II

Vendor	Aerojet		Avion	Hughes
	Tracker	Acquisition		
Operation	Track while Scan (TWS)	TWS	Search and Track	TWS
Field of View	13-1/2° cone	12° to 42° hollow cone	22° Elevation 40° Azimuth	45° Elevation 90° Azimuth
Instantaneous Field of View	—	—	4.5°	0.3° x 4°
Tracking - Field of View	—	—	4.5°	10° x 20°
Frametime Search Track	Rotational Scan —	Rotational Scan —	4 sec. —	4 sec. .5 sec.
Tracking Accuracy	± 2 mil (.002 radian)	—	—	± 1°
Presentation	Optical Sight Combining Glass	Panel Instrument	Optical Sight	Radar Scope
Controls	Added Panel	← Added Panel	Added Panel	Added to Radar Controls
Detection Range	Figure 3	Figure 3	Figure 4	Figure 4
System Weight	30# + Installation	← Included	31# + Installation	31# + Installation
Volume Required (Electronic)	600 cu. in.	← Included	600 cu. in.	200 cu. in.
Power Required	290 watts (115 v)	← Included	75 watts (115 volts) 30 watts (28 volts)	—

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3. INSTALLATION

3.000 General. Possible installation locations for an Avion[illegible] package infrared detector are shown in Figure 5 as described in preliminary design report "Study - Installation of an Infrared Automatic Search and Point System in the F-102A" (ZP-8-048). The equipment considered in this proposal does not include such a package, however, the potential locations for the detector head are the same.

Each of the proposed equipments favor specific installation locations, due to the system specifications and size and shape of the physical units Locations considered for installations of the units discussed in this proposal are shown in Figure 6. Location of associated electronics equipment is also shown.

3.100 Aerojet. To utilize the effectiveness of the tracker detector a relatively rigid mounting is required. The detector head is capable of producing a tracking accuracy of ± 2 mils (2 thousandths of a radian) but the limitations of the optics permits only an accuracy of ± 4 mils in presentation. Therefore, a location is required where mount rotation and deflection is small. At the inboard fence (BL 84), the wing twist for a 1 g load is less than 2 mils depending on the flight loading condition. The cumulative error that can be introduced by the wing, detector head and optics is within the GAR-2 firing tolerance (± 2° instantaneous field of view).

The acquisition detector, since it performs the function of search only, permits installation at more flexible locations. The output information of the acquisition detector is used to locate, by aircraft maneuvering, the target in the tracker field. Hence, any errors produced by twist and deflections of the acquisition detector will be cancelled out by pilot maneuvers in converting to a pursuit attack on the target.

Contrary to the tracker requirements, the acquisition section sweeps out a greater volume of space. Installation then requires a location where interference from the carrying airplane is absent or at a minimum.

Installation of this unit at the outboard wing fence, (BL 150) satisfactorily meets these conditions. Wing twist at this location can exceed 2° (-3g condition) but is not detrimental to the system function. Interference from the fuselage nose section is shown as the shaded area of Figure 7a. An area consisting of a triangle 3-1/2° high by 9° is eliminated from the total field of view. This is not considered to be restrictive.

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Wing installation of the tracker detector unit is shown in Figure 8, acquisition unit in Figure 9. The visual indicator installation of the tracker unit is shown in Figure 10.

A problem area exists for installation of the acquisition unit display indicator, (Figure 11). No space is available on the instrument panel for an additional display unit. Space is available for this display unit by mounting it adjacent to the radar scope or on the cowl line extending from the radar scope. There is, however, a slight reduction of visibility in these areas with such an installation.

3.200 Avion. The detector units perform both the search and track functions. Therefore, a relatively rigid mount location is required. The inboard fence location is acceptable. In addition, for this location, no fuselage interference is present because of the overlay of individual search fields of the two units. The most severe installation restriction is due to the requirement of non-metalic materials within approximately five inches of the detectors head. This requirement can be met by extending the head forward to provide clearance, or by using a non-metalic insert in place of the wing leading edge in the vicinity of the detector unit. Due to the leading edge sweep the first method requires a projection of about 15" which is excessive. Therefore, the non-metalic insert will be considered.

Detector head installation on the wing is shown in Figure 12.

Minor modifications are required to convert the tracking information from the detector unit into a useable display on the combining glass.

3.300 Hughes. The detector unit, which performs both the search and track function has a large search field. Including a unit on each wing would provide a field coverage of 180° which can not be effectively utilized. In addition the detector unit weight of 25 pounds introduces considerable additional weight for very little gain in system effectiveness when a second detector head is added. The greater weight also adds to the installation problem due to the fuel tight requirements of the wing structural box.

Therefore, a tail mounted location is considered for this equipment. Two restrictions are present with this location. Any maneuver or trim load of 1000# on the vertical fin can cause a twist of about 2°. An azimuth reference can be supplied by installing an infrared emitter on the top of the aircraft fuselage to eliminate this problem in search. In track this reference will not be available. However,

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flight maneuvers approaching and at the weapon release point should be at minimum so that no twist is present. The second restriction is that the fuselage blankets out a sizeable portion in the forward down-elevation search field. See Figure 7b. One 'g' angle of attack lines for 5 and 10 degrees are shown in Figure 13. Under most flight conditions some forward down-elevation search field is available. Installation of the detector unit is shown in Figure 14. System controls are added by modification and addition to the existing radar controls. Display is presented through the radar scope as an alternate presentation.

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4. PROPOSED FLIGHT TEST

4.00 The purpose of the proposed flight test program will be to:

1. Evaluate the performance of the infrared sight; and

2. Establish applicable techniques for its employment.

Item 1 above considers performance items such as infrared sight detection and lock-on range and infrared sight aiming error. Item 2 includes such items as GCI ability to establish interceptor to target range and fire point, and applicable safety cut off points for avoiding collision.

4.01 Test Techniques and Conditions. The test program consists of two (2) phases. The first phase of the program will consider Item 1 above. This portion of the program can be accomplished at Convair facilities.

The second portion of the program will consider Item 2 above, and will require testing in the Air Defense Command GCI environment.

4.10 PHASE I (Convair Facilities). The tests to be accomplished in this phase will consist of simulated attacks employing the infrared sight. Each attack will consist of runs on the target from some predetermined point behind the target. Detection and lock-on range will be obtained during these runs. Aiming errors will be examined during the final phase of the attack by miss-evaluation techniques.

The conditions at which these simulated attacks are required are given in the table below:

Target Altitude	Target Mach No.	Attack Description	Interceptor Altitude	Interceptor Mach No.	Initial Interceptor Location With Respect to Target	
					Range Behind in N. Mi.	Off Range in N.Mi.
20,000	.8	Co-alt.	20,000	Mil Power	15	0
					15	4
30,000	.85	Co-alt.	30,000	V_{max}	15	0
					15	4
40,000	.9	Co-alt.	40,000	V_{max}	15	0
					15	4
50,000	.85	Co-alt.	50,000	V_{max}	15	0
					15	4

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PHASE I TABLE (continued)

Target Altitude	Target Mach No.	Attack Description	Interceptor Altitude	Interceptor Mach No.	Initial Interceptor Location With Respect to Target	
					Range Behind in N. Mi.	Off Range in N. Mi.
50,000	.85	Snap-Up	40,000	V_{max}	10	0
60,000	.85	Snap-Up	40,000	V_{max}	10	0

4.20 PHASE II (GCI Environment).

The tests to be accomplished in this phase consist of simulated attacks against targets where tactical vectoring procedures are employed. The scope of this phase will depend to a large extent upon the results of the Phase I tests.

Conditions under which simulated attacks will be required in this phase are given in the table below:

Target Altitude	Target Mach No.	Attack Description	Interceptor Altitude	Interceptor Mach No.
20,000	.8	Co-alt.	20,000	Mil Power
30,000	.85	Co-alt.	30,000	V_{max}
40,000	.9	Co-alt.	40,000	V_{max}
50,000	.85	Co-alt.	50,000	V_{max}
50,000	.85	Snap-Up	40,000	V_{max}
60,000	.85	Snap-Up	40,000	V_{max}

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5. SCHEDULE

5.00 Comparative equipment schedules based on 1 October, 1957, as go-ahead date are shown in Figure 15.

6. RECOMMENDATIONS

The relative merits of the three systems are as follows:

Factors	Remarks
Probability of Detection and Conversion	No significant difference exists among the systems.
Tracking Accuracy	The Aerojet S-11 since it is specifically designed for tracking, has the best characteristics.
Detector Installation	The Aerojet "clamp-on" fairing is preferred over the Avion "insert". The Hughes unit, because of the electronics package is excessive in size and weight, and is therefore not desirable. The Avion system is preferred for minimum aerodynamic interference.
Display Installation	The Aerojet display is obtained by addition to the optical sight and by addition of an acquisition display unit. (Physical requirements only). The Avion display will require internal modification of the optical sight. (Possible design changes).
Schedule	No significant difference between Aerojet and Avion. Hughes schedule too extended.
System Demonstration	The Aerojet system has been installed and demonstrated as a flight article. The other systems are not as far advanced.

The Hughes system is less desirable because of the extended schedule although it is the most completely integrated system.

No single factor determines the selection of either the Aerojet or Avion system and no series of factors selects either system. The Aerojet system is recommended because of:

a) Less development risk due to previous experience or installation.
b) Display method has been well investigated and defined.
c) Aerojet system has been installed and demonstrated in flight operation.

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7. SYSTEM DESCRIPTIONS

7.100 Aerojet. The Aerojet infrared system consists of two detector units. The system components represent qualification tested items, the performance of which has been evaluated in the laboratory and in flight tests, with the exception of the daylight discrimination modification of the tracker. The S-11 (tracker) and W-11 (acquisition unit) Target-Position Indicators are similar in basic design principles, but they differ widely in function. Acquisition equipment must be able to detect targets at a long range and within a large region of space, but does not have to indicate target position with a high degree of accuracy. Indication of the direction of a target within a small region of space is generally adequate for aircraft interception. Tracking equipment differs from acquisition equipment in that it must detect targets at ranges sufficient for maneuvering into position for firing of weapons. The accuracy requirements for the tracking system impose design limits on sensitivity and field of view which do not apply to the acquisition systems.

7.110 S-11 Tracker. Design principle is as follows: The slits of a scanner disc are rotated in a flat scan of the image plane of an objective-lens, and the objective-lens aperture is imaged on an infrared detector by a field-lens. The scanner "chops" the target energy so that a train of energy pulses is incident upon the detector; the train of pulses defines the target position, and the frequency of pulses optimizes the signal-to-noise ratio as processed by the electronic system. The scanner slits are alternately slanted in opposite directions so that consecutive slits when processed and displayed approximate a cross.

Display is provided by a xenon lamp located in the optical indicator behind a synchronized (with detector scanner) rotating flat scanner which is opaque except for slits. An equivalent target image position is projected as a cross on the combining glass and appears at "infinity" to the pilot.

Daylight discrimination, to be added to the production equipment, is obtained by a chopper in the focal plane of the objective lens. A reticle pattern subdivides the instantaneous field of view into small alternate zones of transmission. A point source (target) will appear in only one zone, generating a high chopping frequency signal. An extended source (cloud) covers more than one zone, resulting in a lower chopping frequency. Electronic circuitry discriminates the signals and removes false target information.

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7.111 Equipment Characteristics. The S-11 tracker performs the following functions:

1. Receives target radiation and provides coding of day and night target position into time-modulated direction impulses.

2. Passes to amplifier assembly signals carrying target position data.

3. Passes to power supply signals indicating synchronism, or lack thereof, of tracker scanner with indicator scanner.

4. Provides protection against high-intensity radiation when not in operation.

5. Provides for boresighting in aircraft installations.

Factors or interest on the S-11 tracker are as follows:

1. Weight - 7#.

2. Characteristics

(a)	Field of view.	13.5°
(b)	Equivalent focal length.	8.5"
(c)	Focal ratio.	f:6
(d)	Objective aperature.	14"
(e)	Lead sulfide detector "Jones S."	3 X 10^{-8} watt.
(f)	Synchronous transmitter signal 1 V/° error.	

7.120 W-11 Acquisition Unit. The acquisition system optically views an annulus in space surrounding the tracker field by deviating and rotating the field of view imaged by an infrared objective lens. This is achieved by rotating an assembly consisting of a prism, objective lens and a reticle. The prism is placed in front of the objective lens so that the energy collected by the lens originates from points off the optical axis.

The infrared energy from a target is imaged as a point in the focal plane of the lens at which is situated a reticle with a pattern of clear and opaque areas. The reticle rotates with the prism and thereby chops the imaged target. A target within the annulus is

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detected by the system only when its image is traversing the width of the reticle. The width of the reticle represents a narrow sector of the annulus, and the length of the reticle extends radially to determine the inner and outer diameter of the annulus, 12° and 42° respectively. Targets within the annulus are resolved by the system within 5° radial separation. The reticle pattern is designed to give a fixed-frequency pulse train when a target is scanned. The chopped energy is refracted onto an infrared detector, and the signal from the detector is amplified sufficiently to operate a triggering circuit which fires a xenon lamp. The detector, amplifiers, and xenon lamp are similar to the equivalent components in the infrared tracker.

The xenon lamp is located behind a rotating mask which has a cut-out pattern of an arrow. The assembly rotates in synchronism with the acquisition optics. Energy on the infrared detector causes the arrow to be illuminated by the triggered xenon lamp to give indication of the target direction. These elements are assembled in a form suitable for instrument mounting in the aircraft. Results of field tests have shown that this arrow presentation provides a satisfactory means of indicating to the pilot the direction he should turn his aircraft to bring the target within the field of the tracker and thus intercept the target. (Acquisition unit weight is 5.0#).

7.130 Equipment.

7.131 Amplifiers. The amplifier performs the following functions:

1. Accepts from detectors the signal carrying target-position data.
2. Amplifier filters and shapes this signal for indicator presentation.
3. Passes this processed signal to indicator.

Factors of interest are:

1. Unit size 3.60 wide, 5.0 in. high and 8.2 in. long.
2. Weight 3.6#.
3. Characteristics, preamplifier.
    (a) 50 μv input signal gain 500
    (b) Noise referred to input (shorted) 20 μv

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4. Driver amplifier, provides 125-V negative pulses to thyration circuit over range of input signals.

7.132 Power [redacted] Supply. The power supply performs the following functions:

1. Input 115 v , 400 cps power
2. Provides 2-phase 115 v , 400 cps power to unit drive motor.
3. Provides the following to electronic equipment.
    (a) +400 v , dc, unregulated.
    (b) +150 v , dc, series-regulated.
    (c) +108 v , dc, V-R tube-regulated.
    (d) +6 v , dc, filament supply.
    (e) 6.3 v , ac, filament supply.
4. Receives error signal from synchronous transformer in tracker.
5. Processes synchronizing-error signal and causes tracker drive. motor to reposition to synchroninism with indicator drive motor.
6. Contains operating electronic circuiting for sun-protection-shutter operation.

Factors of interest are:

1. Unit size, 9.0 in. wide, 5.6 in. high and 8.0 in. long.
2. Weight 6.0#.

7.133 Control Box. The control box performs the following functions:

1. Accepts power from external power source.
2. Provides fused power to power supply.
3. Provides control of threshold sensitivity.
4. Accepts target indication signal from power supply and transmits to indicator.

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Factors of interest are:

1. Unit size, 4.1 in. wide, 3.0 in. high and 3.25 in. long.

2. Weight 1.0#.

7.134 Visual Indicator, Tracker. The following functions are performed:

1. Receives target indication signals from the amplifier via power supply and control box.

2. Triggers a xenon lamp in phase with target indication signal.

3. Codes light from xenon lamp by means of moving slits to represent equivalent target position.

4. Projects collimated light to pilot.

Factors of interest are:

1. Unit size, 5.0 in. wide, 8.0 in. high and 7.5 in. long.

2. Weight 6.0#.

3. Characteristics:

   (a) Visual flicker frequency, 12 flashes/sec/cross leg.

   (b) Aperature, collimating lens. 2.25"

   (c) Focal length, collimating lens. 4.0"

7.135 Visual Indicator, Acquisition. Functions of this unit are the same as for the tracker except that it is a panel mounted arrow-type presentation unit.

7.136 Power Requirements. Power requirements for the installation are 290 watts.

7.137 Weight. System weight, not including installation and wiring is approximately 30#.

7.140 System Performance. Measured range performance with existing equipment is shown in Figure 3. The expected ranges for the modified

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production equipment are also included. The infrared sights currently being manufactured detect a tail aspect single-jet aircraft with full military thrust at 9 n. mi. at 30,000 feet altitude. Multiple engine targets with full military thrust have been detected at 18 n. mi. at 30,000 feet altitude.

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7.200 Avion. The proposed IR Automatic Search and Track System will consist of two infrared optical seekers mounted on the wings of the aircraft. Each seeker will have the capability of searching a square field-of-view 22.5° by 22.5°. The search rate will approximately be one frame in three seconds. The instantaneous field-of-view will be 4.5°. Axial alignment of the two seekers will be adjusted to produce a total forward search field of 40° (azimuth) by 22.5°.

The system will be capable of being programmed to search and track modes. The output of the system will be compatible with the input of the optical sight of the F-102A MG-10 fire control system. In order to give the tracking display accuracy required, a non-linear optical display will be used, giving maximum sensitivity in the center of the display. The outer portion of the display will then give the direction to fly to acquire the target while the inner position will display the azimuth and elevation angular position of the target.

The output signal voltages from each seeker are brought together through a single channel selector relay. The channel selector is used to program one of the channels into the optical display equipment. Each channel utilizes identical equipment consisting of: (1) optical seeker unit, (2) signal amplifier, (3) search pattern generator, (4) tracking indicator and lock-on relay, (5) precision amplifier and (6) demodulator.

7.210 Optical Seeker. The optical seeker consists of a space stabilized optical tracking gyro and a coated quartz dome. The detector is a 3 x 3 mm non-precessing lead sulfide cell and a filter with a pass band of 1.8 to 2.5 microns. The optics consist of a cassegrain type telescope with a 4.5° field-of-view imaged on a chopping reticle. The molded housing will be designed to fit either in, or on the wing and will protrude forward of the leading edge by 6 inches to keep the gyro away from metal in the wing.

7.211 Equipment Characteristics.

1. System sensitivity for 1:1 signal to noise ratio will be 3×10^{-11} watts/cm^2.
2. Quartz dome - 3-1/2 inches diameter.
3. Optics
   - (a) 3 x 3 mm PbS detector.
   - (b) Focal Length of 2.25 in.
   - (c) Field-of-view 4.5°
   - (d) f number :1

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4. Motor Drive.

   (a) Gyro speed, 70 + 1 cps
   (b) Chopping frequency, 840 cps

7.220 Preamplifier. The preamflifier consists of a single stage of amplification (gain of 20) and a cathode follower. It is mounted in the molded optical seeker unit and amplifies the signal from the lead sulfide cell. The amplified signal is then transmitted through the cathode follower to the electronics section.

7.230 Electronics. The major units consist of: (1) a signal amplifier, (2) a precession amplifier, (3) a search pattern generator, and (4) a demodulator. These units will be packaged on a single chasis and housed within a standard aircraft cabinet of approximately 600 cu. in. The electronic equipment has the function of amplifying the incoming chopped signal and provides the required precession signals. In addition, this circuitory generates, by means of a stepping relay, the voltages required to precess the gyro into a search mode. A final function of the electronic equipment is to convert the incoming polar target information data to rectangular information.

7.231 Electronic Characteristics.

1. Signal amplifier

   (a) Gain 15,000
   (b) Center frequency, 840 cps
   (c) Band width , 150 cps

2. Power required for system:

   (a) 110 volts, 400 cps, 1 phase, regulated; 75 watts.
   (b) 28 volts dc; 30 watts

7.240 Control Panel. The control panel contains the necessary operating switches and indicator lamps. The mode selector switch will be used to select either a search and track or caged mode of operation. On acquisition of a target, from either seeker, the seeker channel indicator lamp will become illuminated. During track of an acquired target, should the other seeker channel lamp light, the indication would be of another target acquired by the opposite seeker head. Switching the channel selector release switch feeds information from the opposite seeker head into the MG-10 optical sight. A light will indicate when the detector signal level is sufficient to produce lock-on for the GAR-2.

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7.261 Search Mode. Under the search mode, the scan generator sets the seeker in the upper left-hand corner of the scan pattern. The seeker begins its scan cycle, moving from left-to-right, down one "field-of-view", right-to-left, and so on, until five horizontal scans are completed. The scan generator then resets the seeker to the upper left-hand corner of the frame and a new frame scan begins. Complete scan requires three seconds. The search pattern is controlled by the position of a set of reference coils fixed to the axis of the aircraft. Therefore, as the aspect of the aircraft changes, the search pattern changes with it.

7.262 Track Mode. Both seekers start to search at the same time, but do not operate in synchronism. When a 3:1 signal to noise signal is received, the seeker stops searching and tracks the target. On acquisition, an indicator is illuminated and signals are fed to MG-10 fire control system. These signals represent in Cartesion coordinates, the position of the target with respect to the aircraft. A dot, representing the target, would be presented on the optical sight combining glass.

After the seeker starts to track, should the target be lost for a period greater than one-half second, the system returns to the search mode.

On acquisition of a target by the other seeker, the output is not presented to the optical sight when one seeker is tracking. Presentation from one channel to the other is accomplished by manual switching. Automatic cross-over tracking circuits have not been designed into the system.

7.263 Search System. The search system will be a programmed system in which position information will be fed into the precession amplifier through a stepping relay. The position information will be defined from combinations of up-down and right-left reference coils located in the head coil assembly. These positions voltages will be fed in series with the axially wound cage coil in such a manner as to permit gyro movement to a position where the voltage induced in the cage coil will null the position voltage. At this point, a null amplifier will activate the stepping relay which selects and injects a new-position voltage. It is possible to vary the search pattern by programming the position voltages, as desired, at the contracts of the stepping relay. The search pattern is repeated until a signal exists in the signal amplifier. At this point the lock-on relay disconnects the search output and connects the signal information to the precession

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amplifier. The lock-on relay simultaneously energizes a tracking indicator lamp on the pilots control panel. A channel selector relay connects the tracking channel to the presentation device. The electronics for each searching channel are identical up to the channel selector relay. The relay will be provided with a manual release switch permitting the pilot to switch from one search channel to the other if desired. Each channel will utilize two ring demodulators to provide the Cartesian coordinate output information to the presentation circuitry.

7.270 System Performance. The system requires a signal-to-noise ratio of 3:1 in order to produce lock-on. Range figures for target lock-on have been calculated for the proposed system based on the following assumptions:

1. B-47 type aircraft having a tail pipe temperature of 630° C.
2. Atmospheric attenuation is neglected.
3. Tail-on aspect.
4. Clear sky background conditions.

Under these conditions a range of 14.7 n. miles (tail aspect) can be expected with the optical filter installed. Approximately ranges from other aspects are shown in Figure 4.

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7.300 Hughes. The system, using a modified GAR-2A head, would be integrated with the MG-10. The modified head is basically a scanner with a rapid scan mode for track-while-scan operation. The IR search and tracking information is presented on a C-Display which can be operator selected in place of the radar search display. Controls are integrated with the radar mode switch and the hand control. A panel light will indicate when signal level is adequate for missile lock-on.

7.310 Equipment Requirements.

1. Search: 90° azimuth, 45° elevation, frame time 2-4 seconds, instantaneous feild 0.3° X 4°.
2. Track: Collapsed field 20, X 10°, frame time 0.25 to 0.5 seconds, tracking accuracy = 1 degree.
3. Roll stabilization of IR head.

7.320 Proposed Equipment. The optical system, servo drives, servo amplifier, signal amplifier, roll control syncro and power supply are contained in a cylindrical unit approximately 5-1/2 inches in diameter and 14-1/2 inches in length. Estimated weight of the head is 25#.

Detection range should be comparable to that of the Avion system (Section 7.270).

7.330 Integration. Information from the infrared detector would be displayed on a C-Display which would replace the radar search display. Target marker circle, artificial horizon, and time circle would also be displayed. The infrared system would be activated by switching to 'IR' on a new position which must be provided on the Radar Mode Switch. The infrared system would be put in the tracking mode (collapsed scan) by depressing and releasing the Action Switch. Return to search can be initiated by pressing the Auto-Search Button on the hand control.

To integrate the infrared system into the MG-10 system one new box 5" X 5" X 8" and weighing about 6 pounds must be provided. This box must be located in the nose section of the airplane because of the requirement for low capacitance on the video and deflection leads.

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FIGURE 1

LOCK ON AND LAUNCH RANGES OF THE GAR-2 FALCON VS. ALTITUDE

Pursuit Course
B-47 Type Target M = .85
Interceptor Mach = .90
30° Tail Aspect

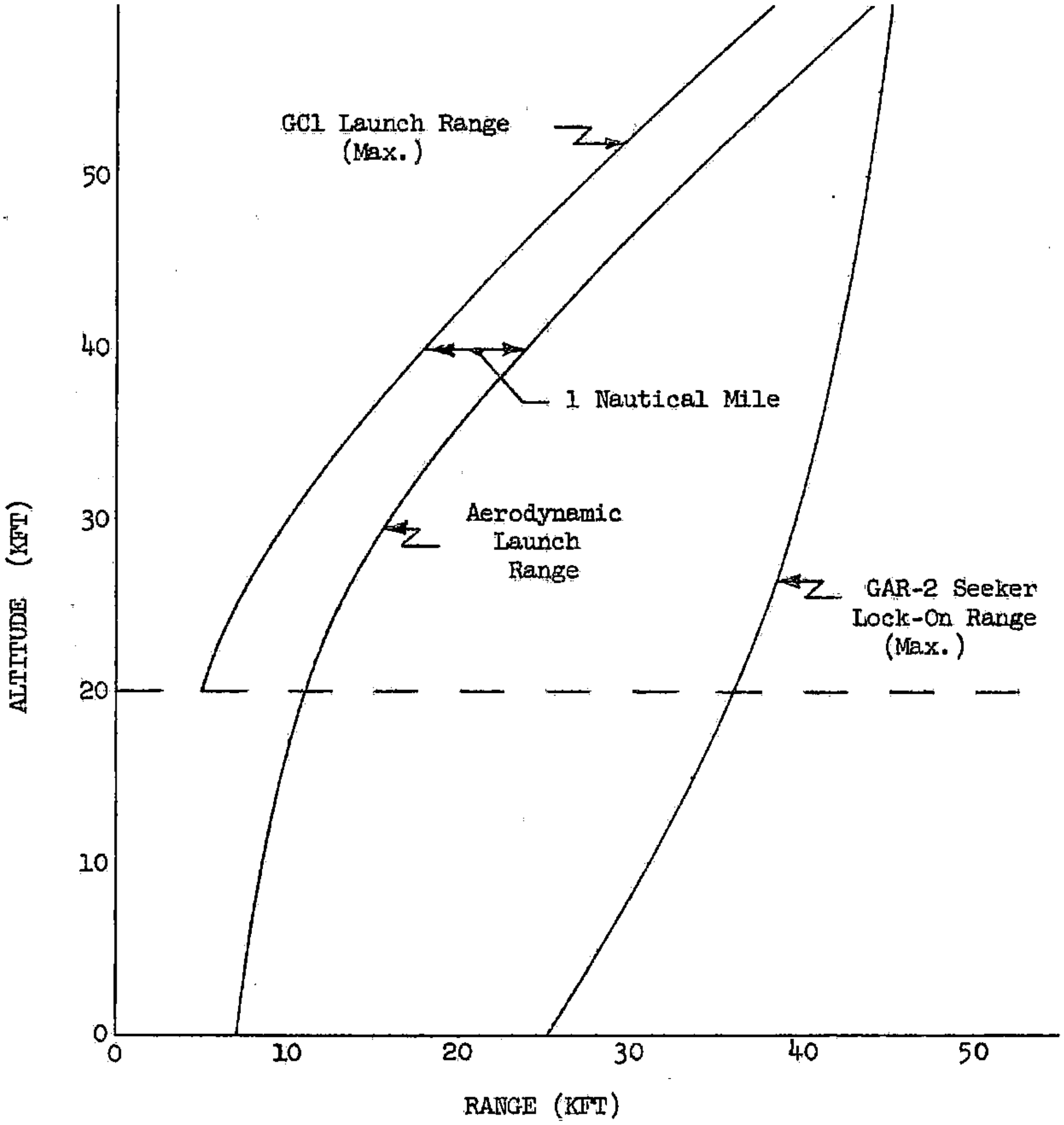

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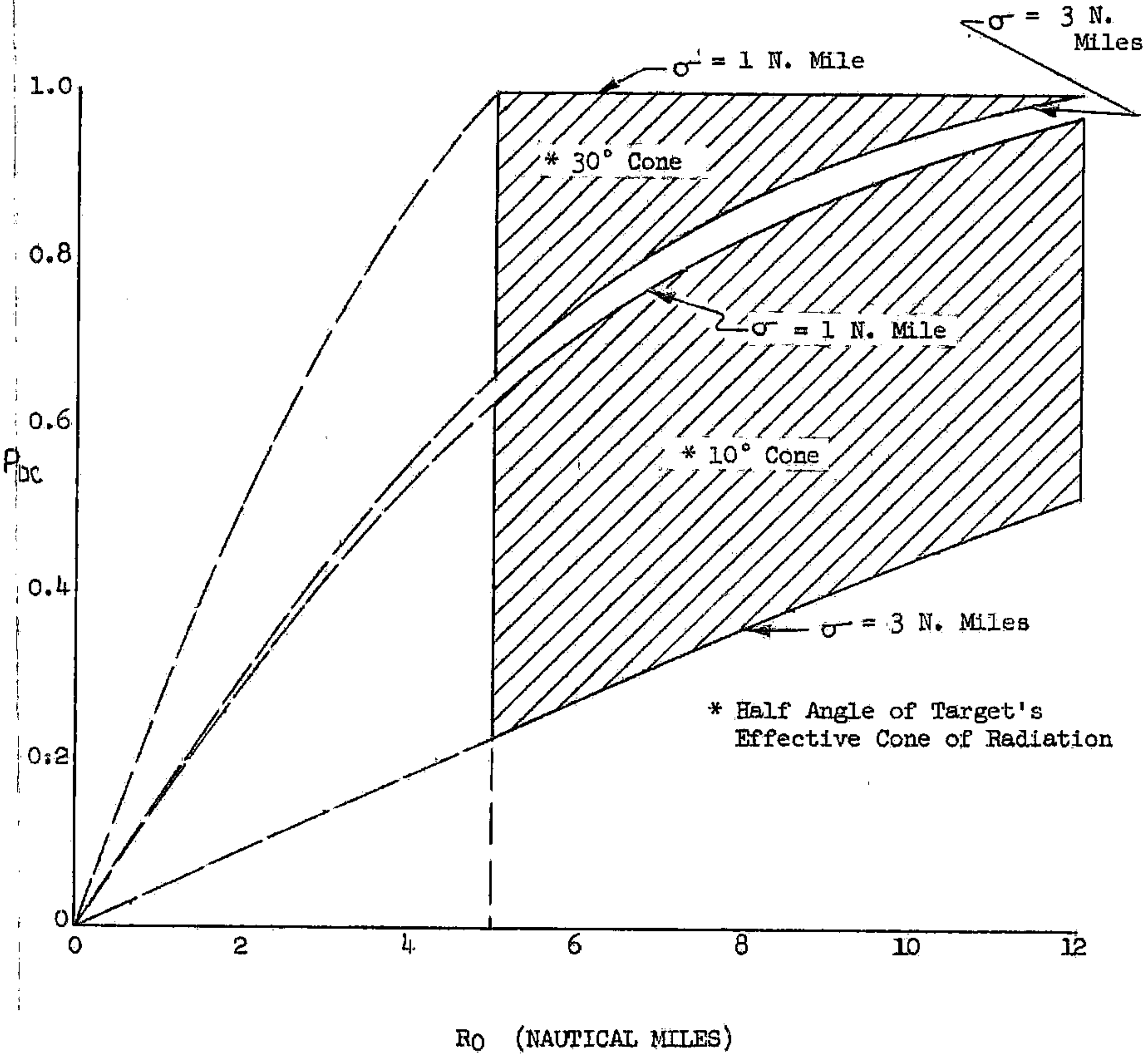
FIGURE 2

INITIAL RANGE TO TARGET VS. PROBABILITY OF DETECTION AND CONVERSION



[redacted]
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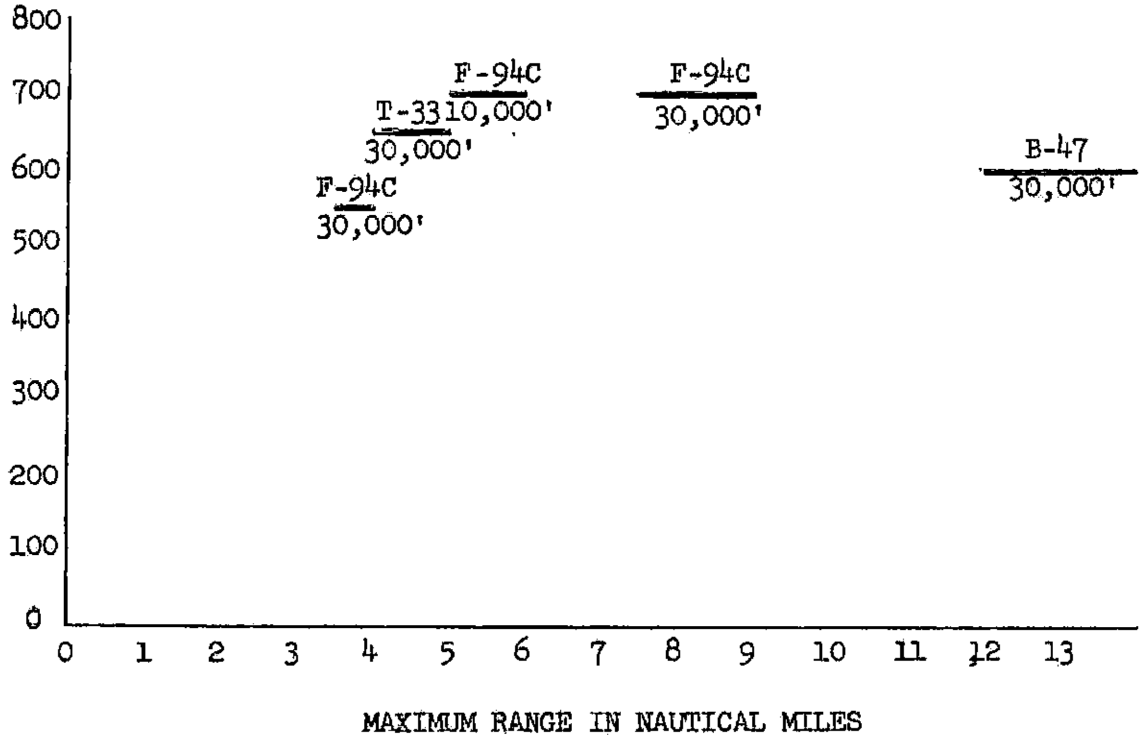
[redacted]

FIGURE 3

NIGHT RANGES OBTAINED WITH MA-10 IR SIGHT (AN/ASG-14) UNDER LARGE VARIATIONS OF METEOROLOGICAL CONDITIONS (TAIL ASPECT)



NOTE: ANTICIPATED RANGES OF
MODIFIED DAYLIGHT AN/AAA-1
AGAINST F-94C AIRCRAFT -
9 MI. DAY, 13 MI. NIGHT

[redacted]

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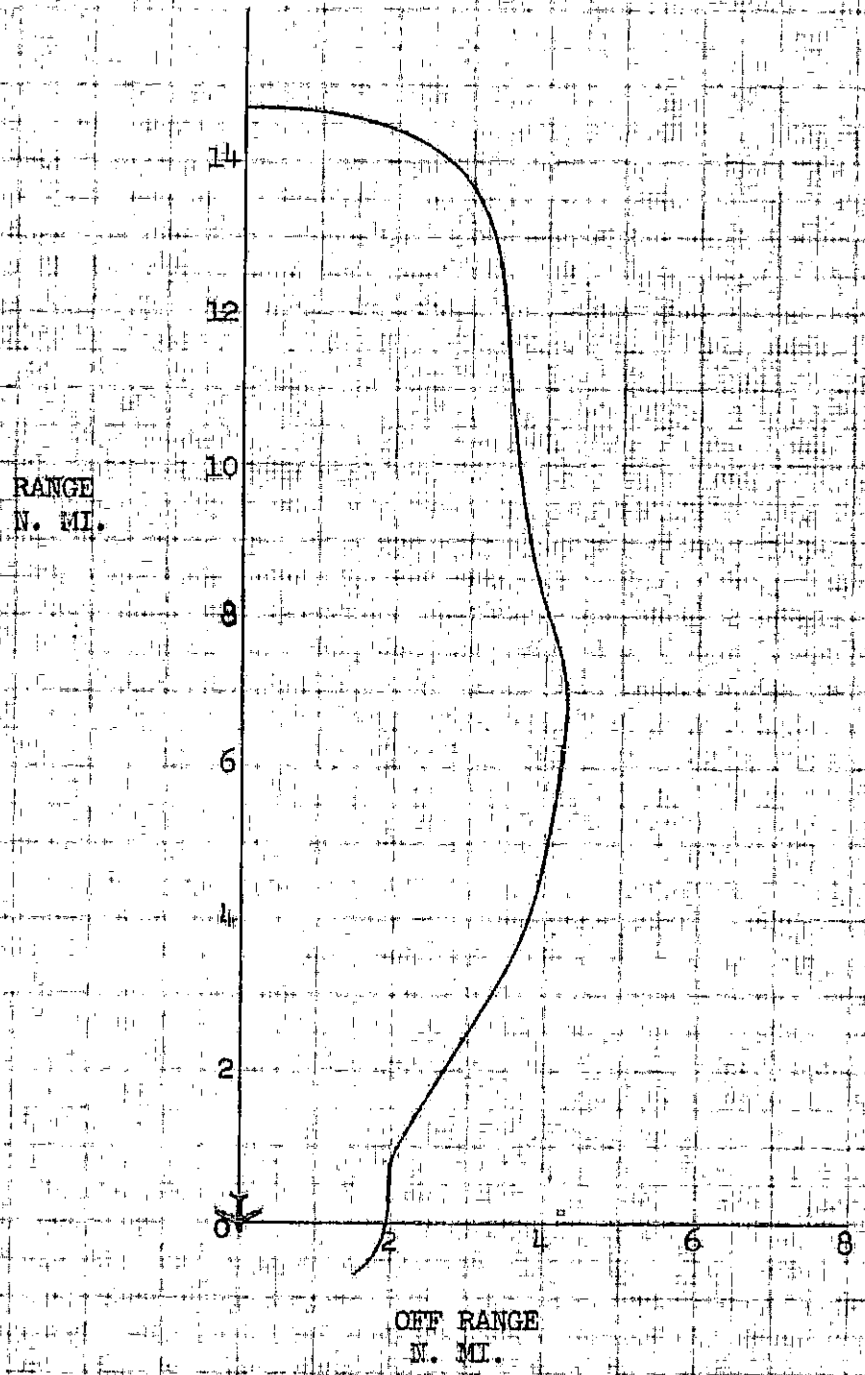
[redacted]

FIGURE 4

INFRARED DETECTION RANGE
B-47 TYPE TARGET
630°C TAILPIPE TEMPERATURE
(PREDICTED)



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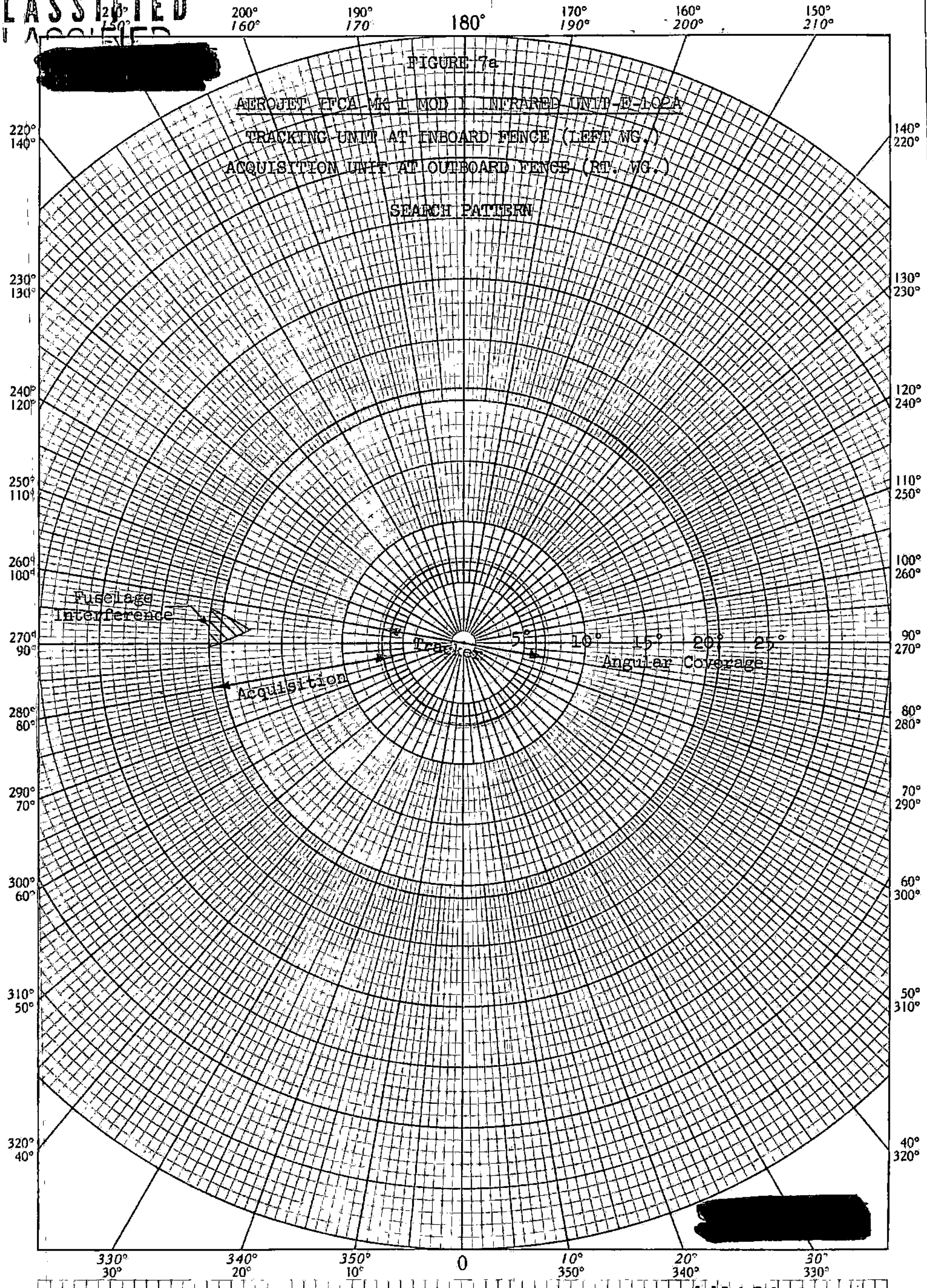

FIGURE 7a

AEROJET FFCA MK 1 MOD 1 INFRARED UNIT E-102A

TRACKING UNIT AT INBOARD FENCE (LEFT WG.)

ACQUISITION UNIT AT OUTBOARD FENCE (RT. WG.)

SEARCH PATTERN

[redacted]

359-21 KEUFFEL & ESSER CO.
Polar Co-ordinate
MADE IN U. S. A.

7a

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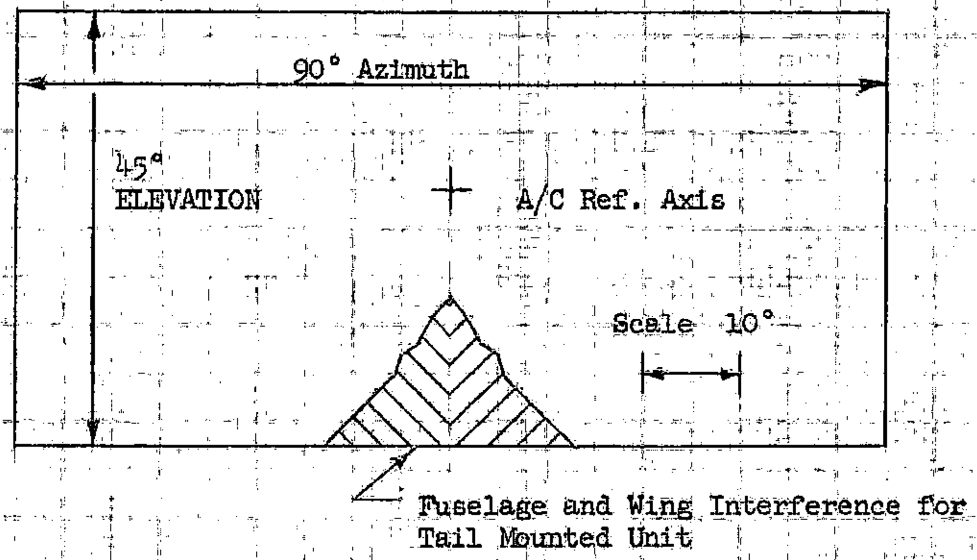
[redacted]

FIGURE 7b

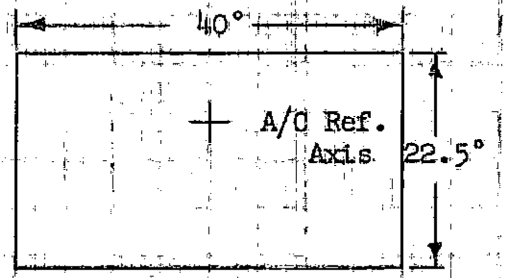
H.A.C. INFRARED AUGMENTATION SYSTEM
4° DOWN LINE OF SIGHT SEARCH PATTERN



Tracking Field is 20° x 10°

FIGURE 7c

AVION PASSIVE DETECTION SYSTEM
4° DOWN LINE OF SIGHT
2 UNITS, 1 @ WING
SEARCH PATTERN



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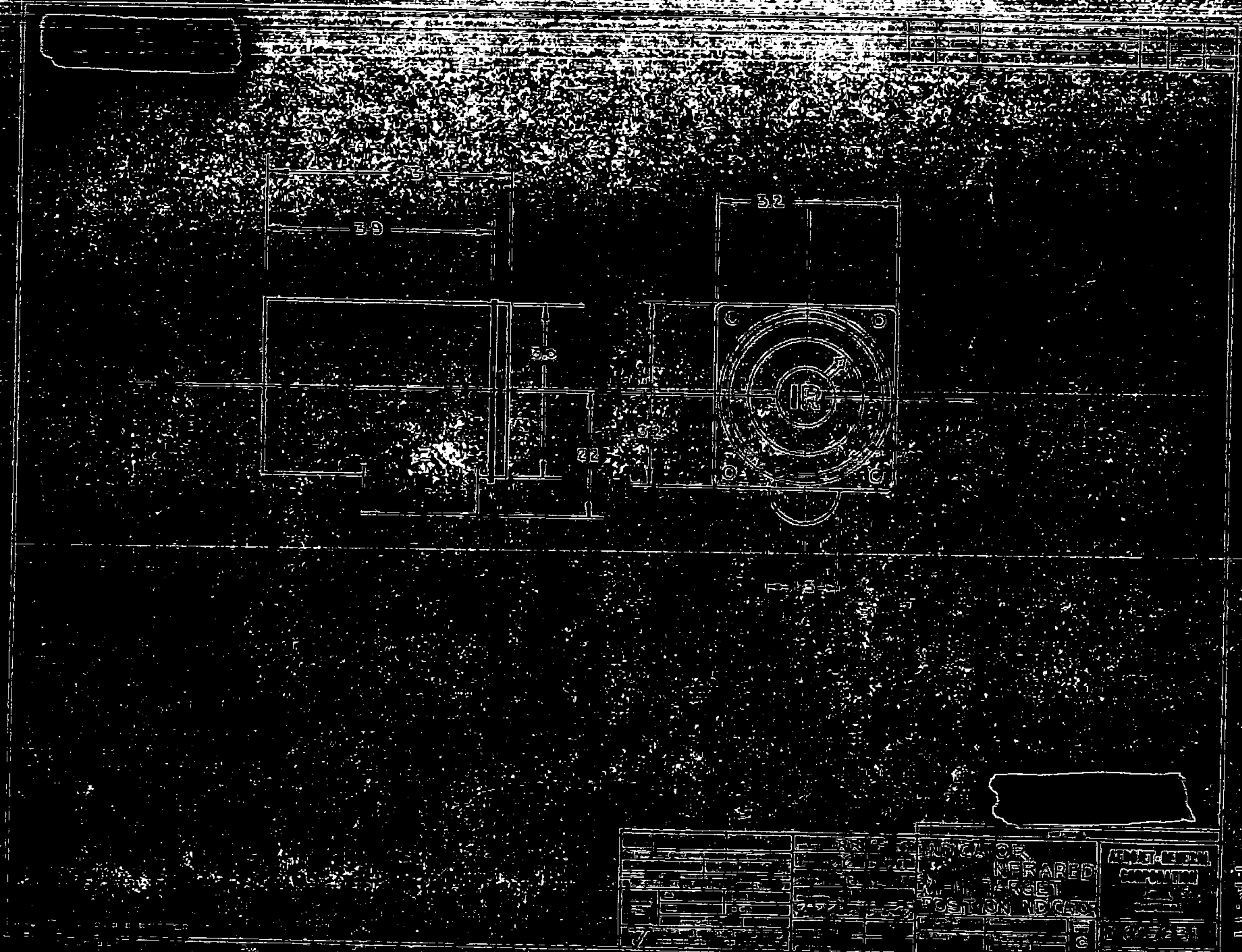
7b,7c

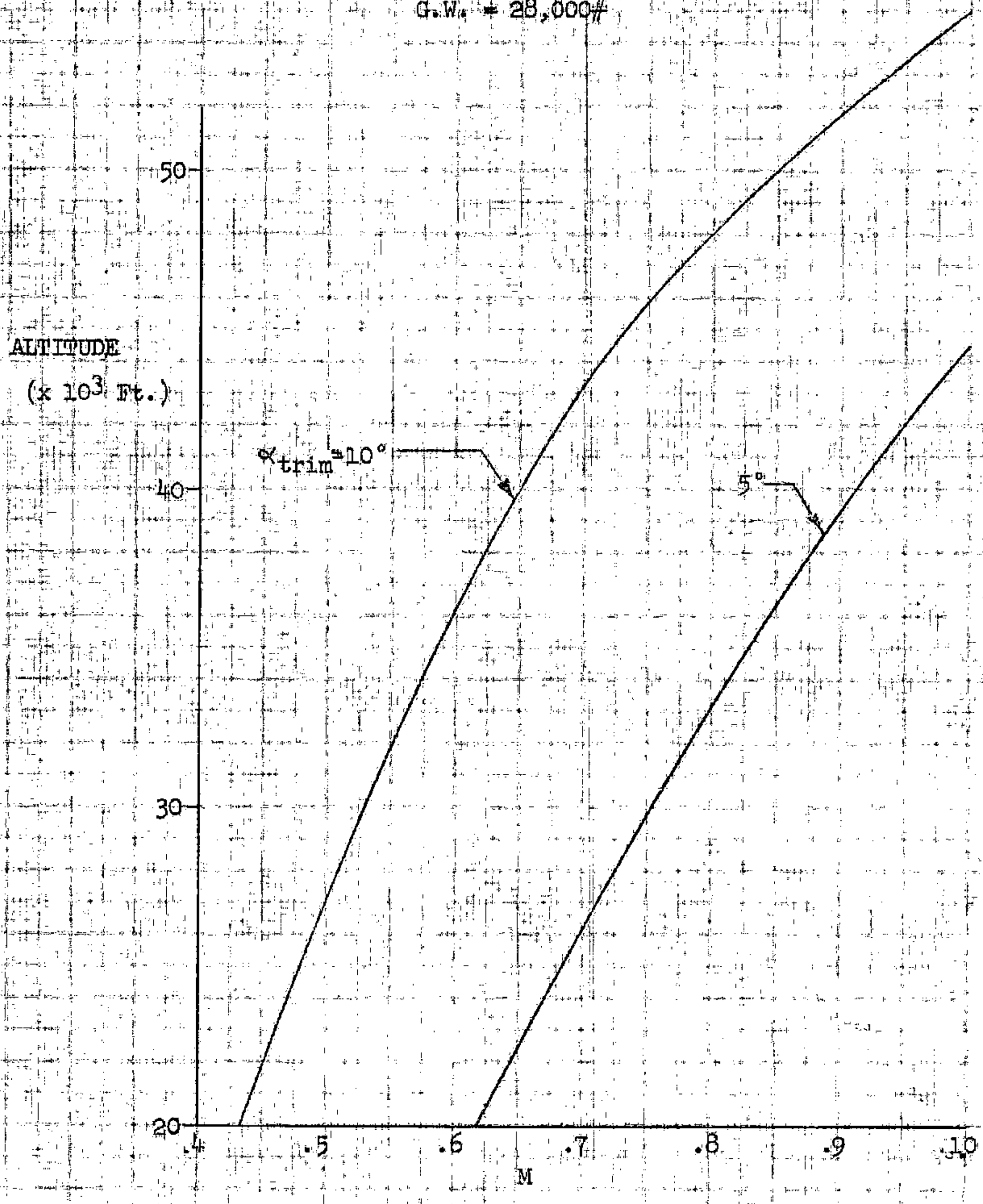

3.9
3.2
3.0
2.2
3.2
IR
INDICATOR, INFRARED WITH TARGET POSITION INDICATOR
C

[redacted]

FIGURE 13

F-102A, 1 CONSTANT ANGLE OF ATTACK

G.W. = 28,000#

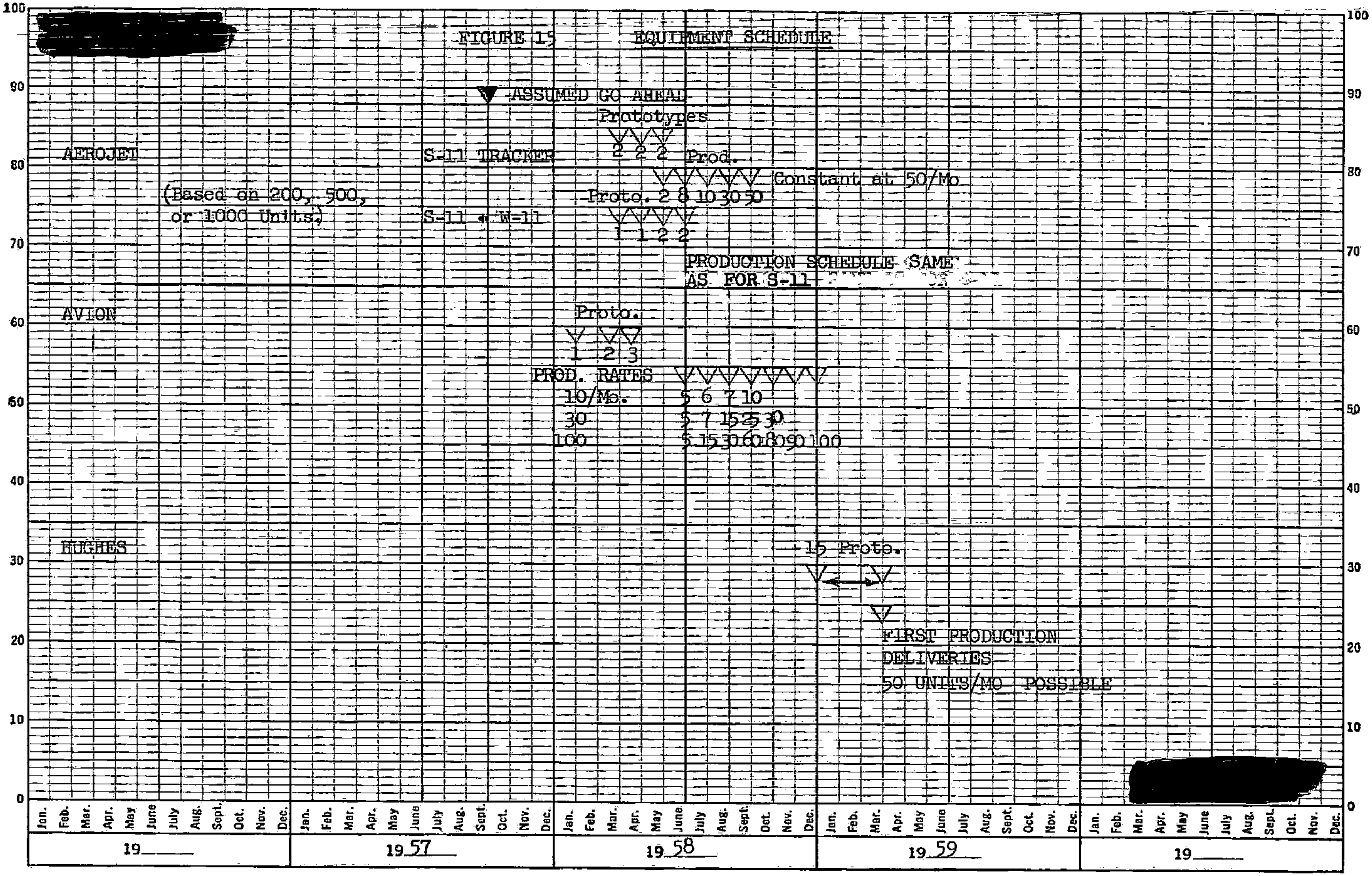


[redacted]

K+E 5 YEARS BY MONTHS X 100 DIVISIONS 359-192
KEUFFEL & ESSER CO. MADE IN U.S.A.

[redacted]

FIGURE 15 EQUIPMENT SCHEDULE



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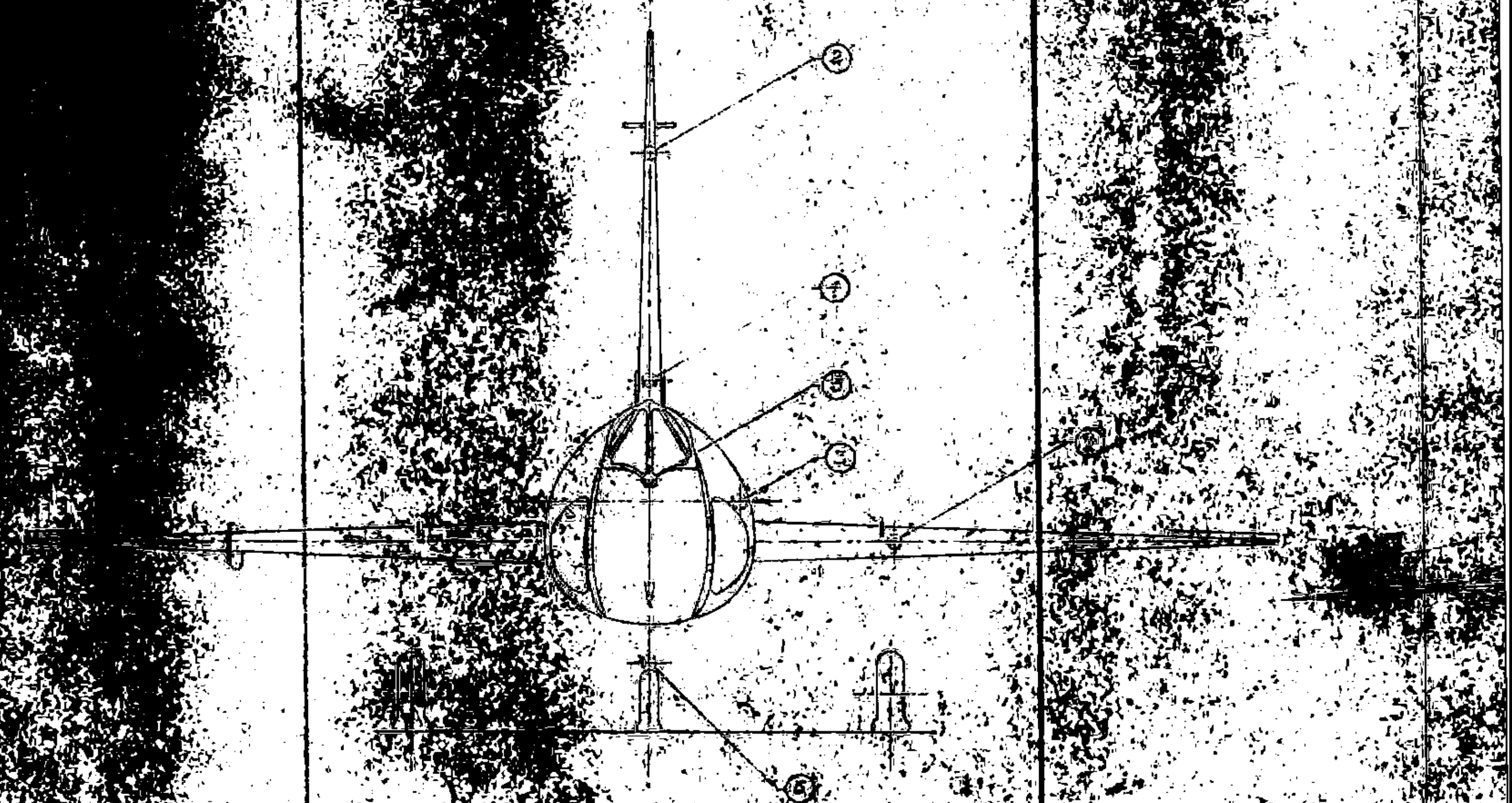


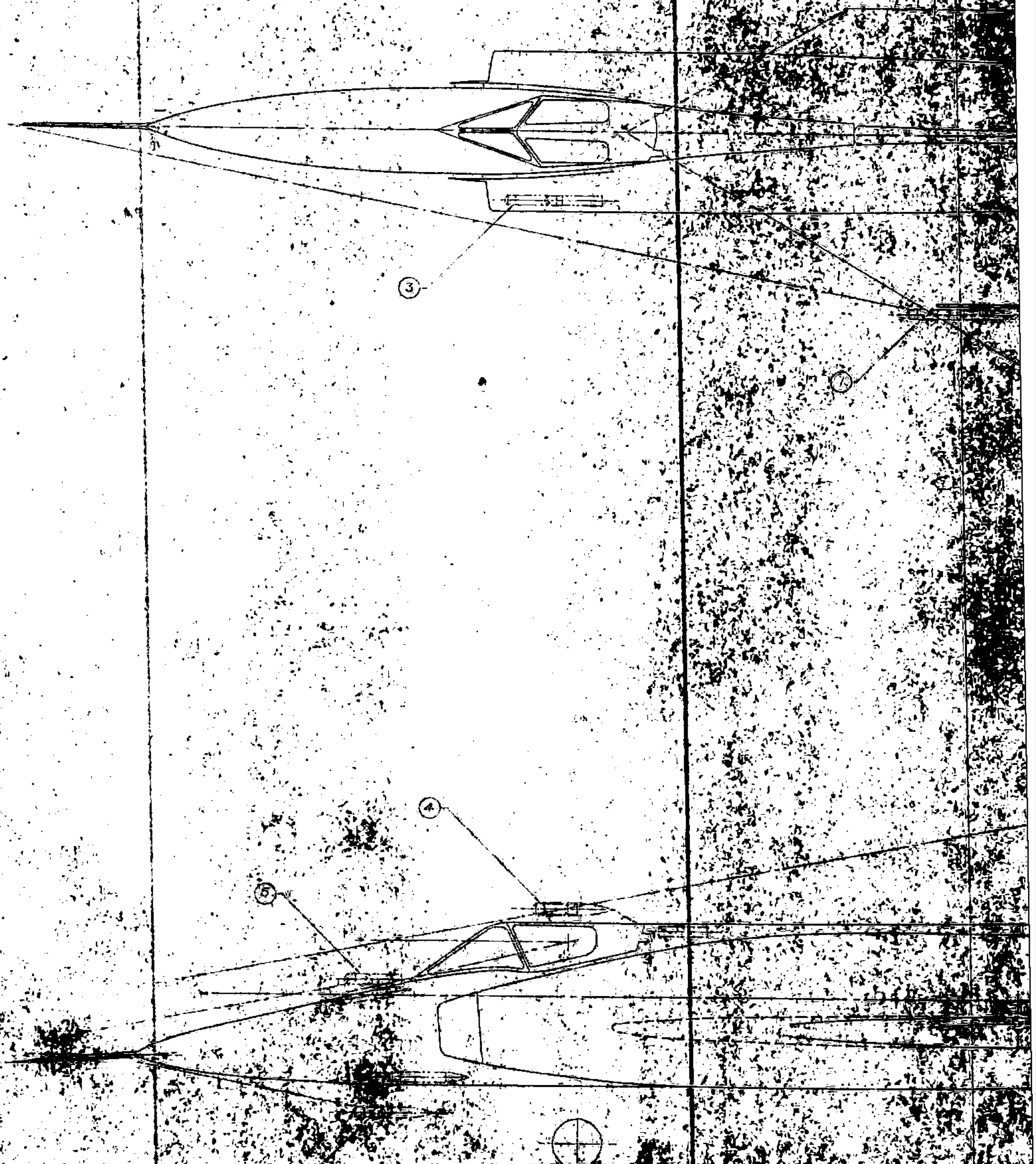
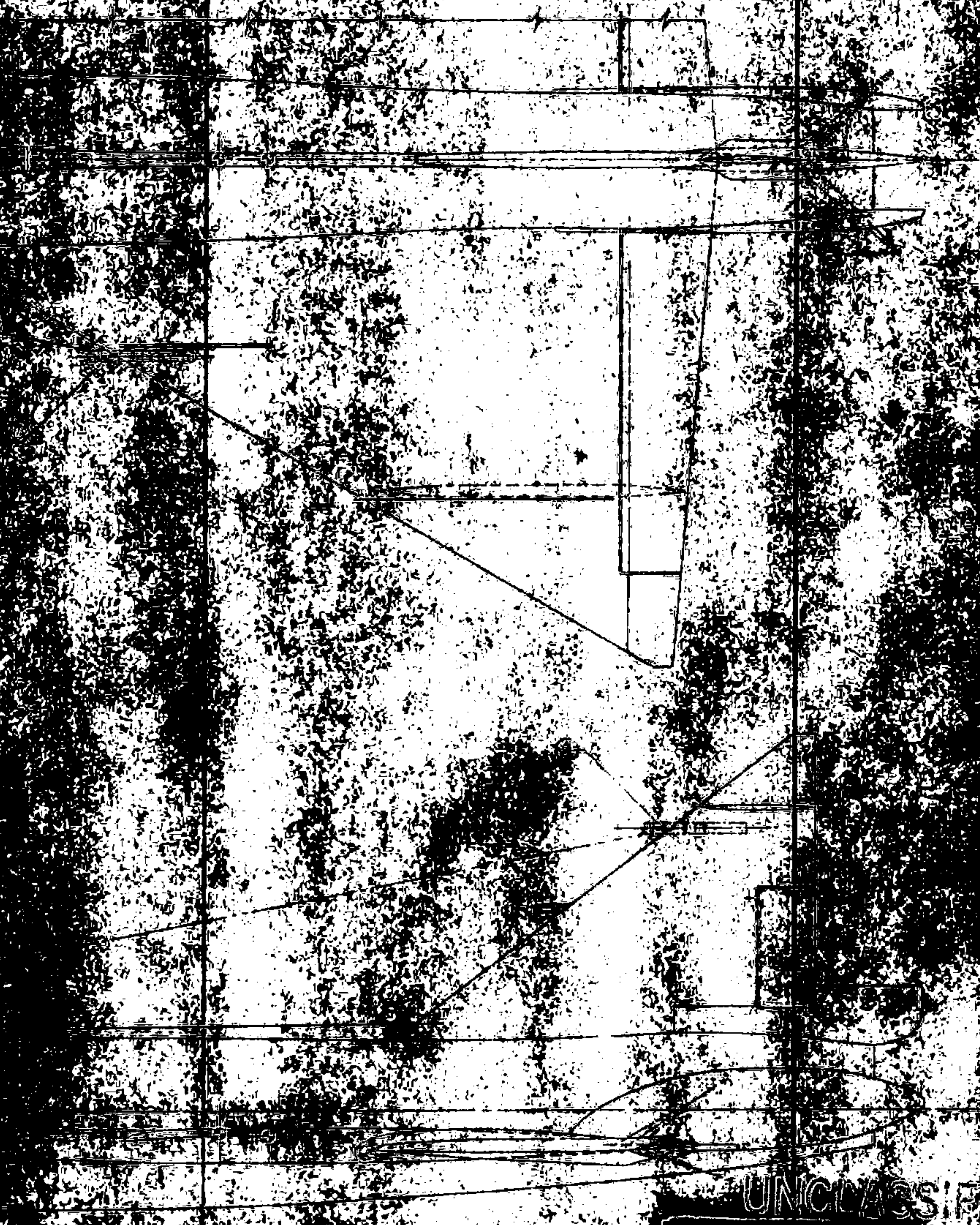

③
⑦
④
⑤
⑥

FIG. 5



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CONVAIR

PRELIMINARY DESIGN DRAWING

INSTALLATION AVIONICS MODEL 333 ASP

VARIOUS LOCATIONS F-102A

CONVAIR SAN DIEGO CALIFORNIA 5G -

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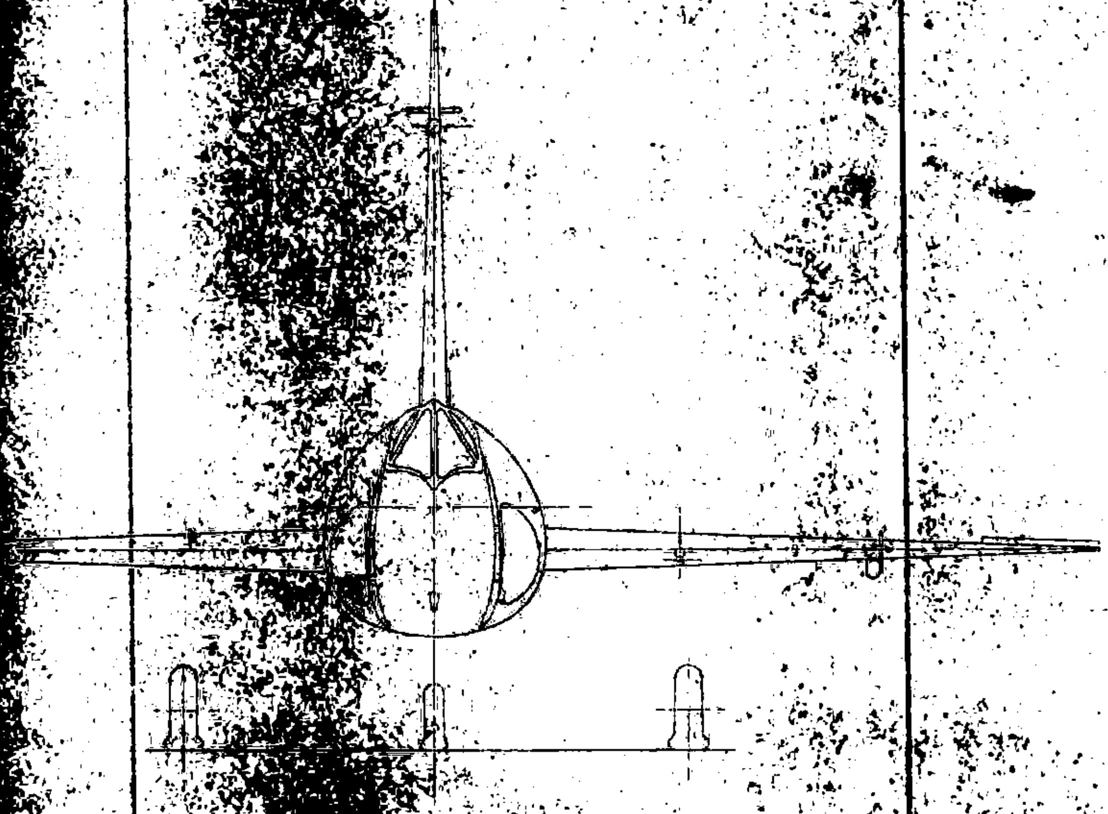


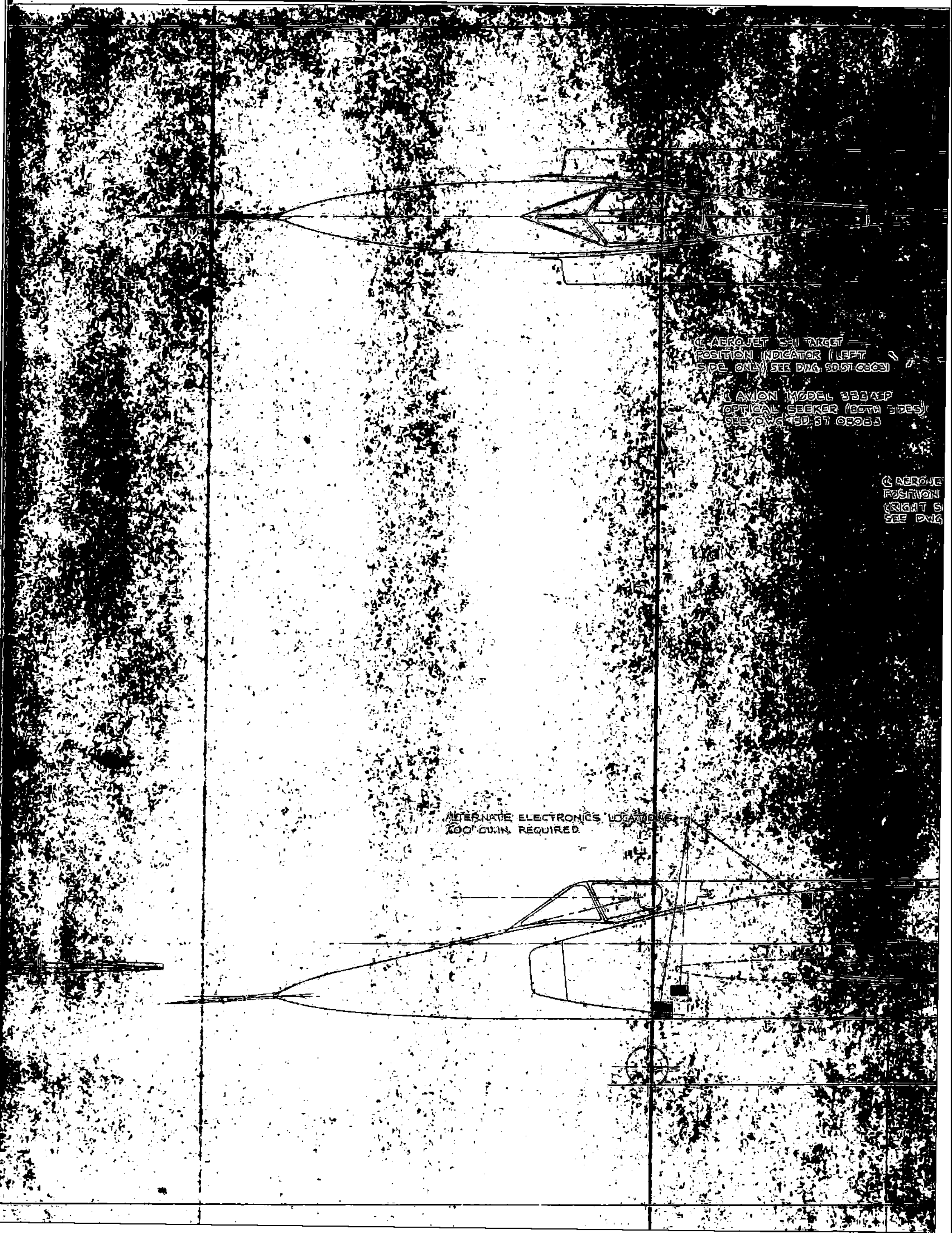
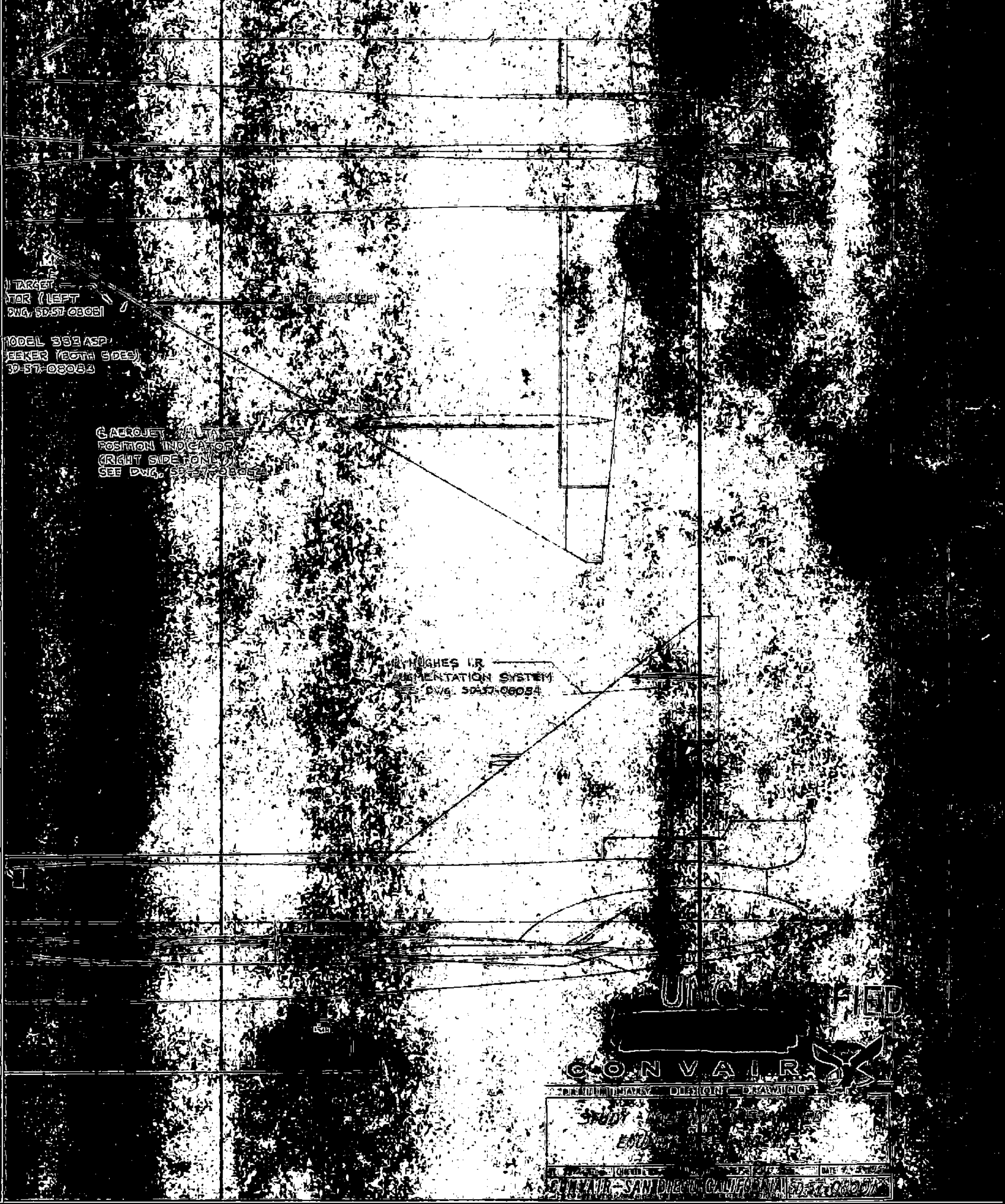
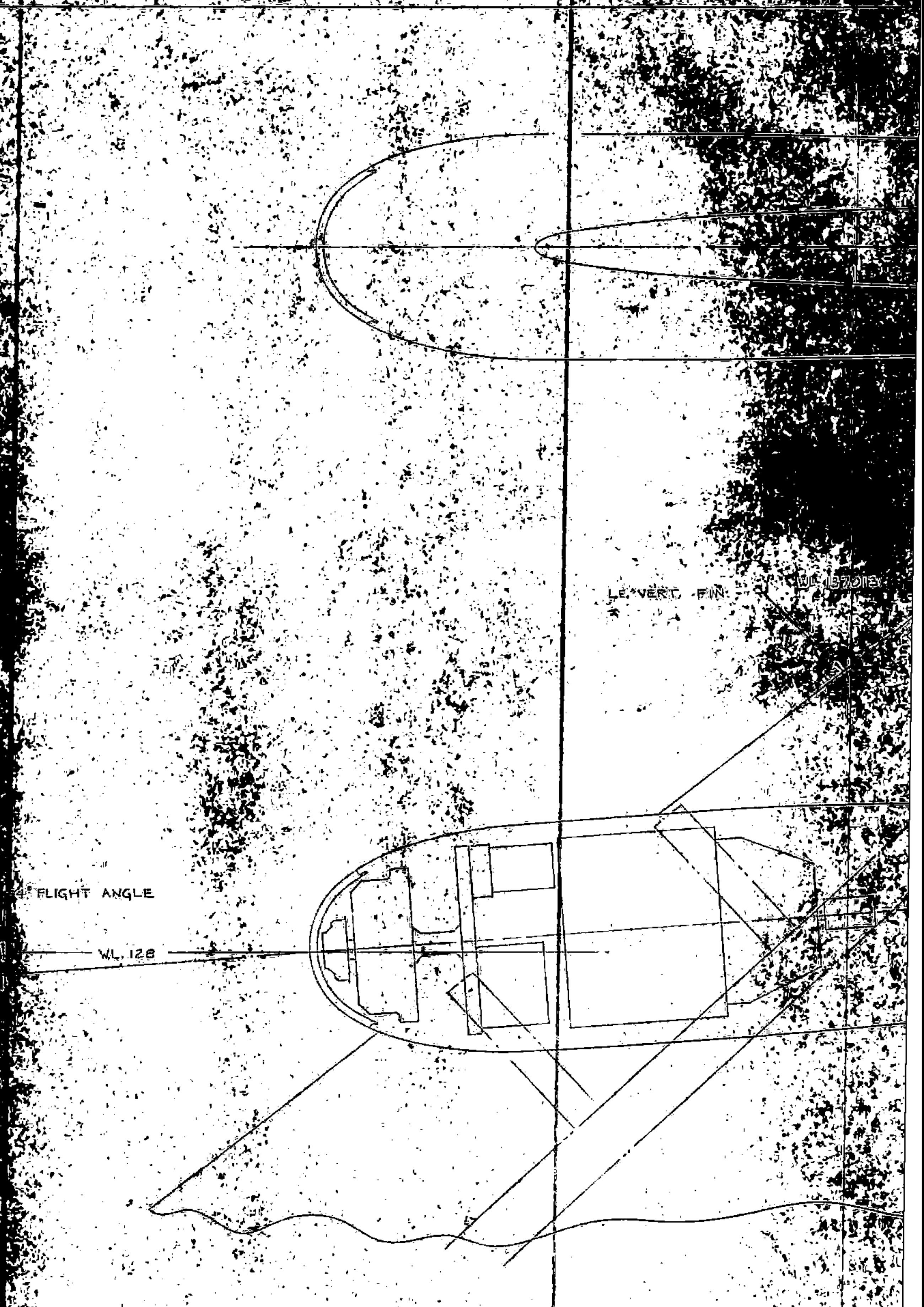

℄ AEROJET S-II TARGET POSITION INDICATOR (LEFT SIDE ONLY) SEE DWG. SD57-08081
℄ AVION MODEL 333 ASP OPTICAL SEEKER (BOTH SIDES) SEE DWG. SD 57 08083
℄ AEROJE
POSITION
(RIGHT S
SEE DWG
ALTERNATE ELECTRONICS LOCATIONS
600 CU.IN. REQUIRED

Fig. 6



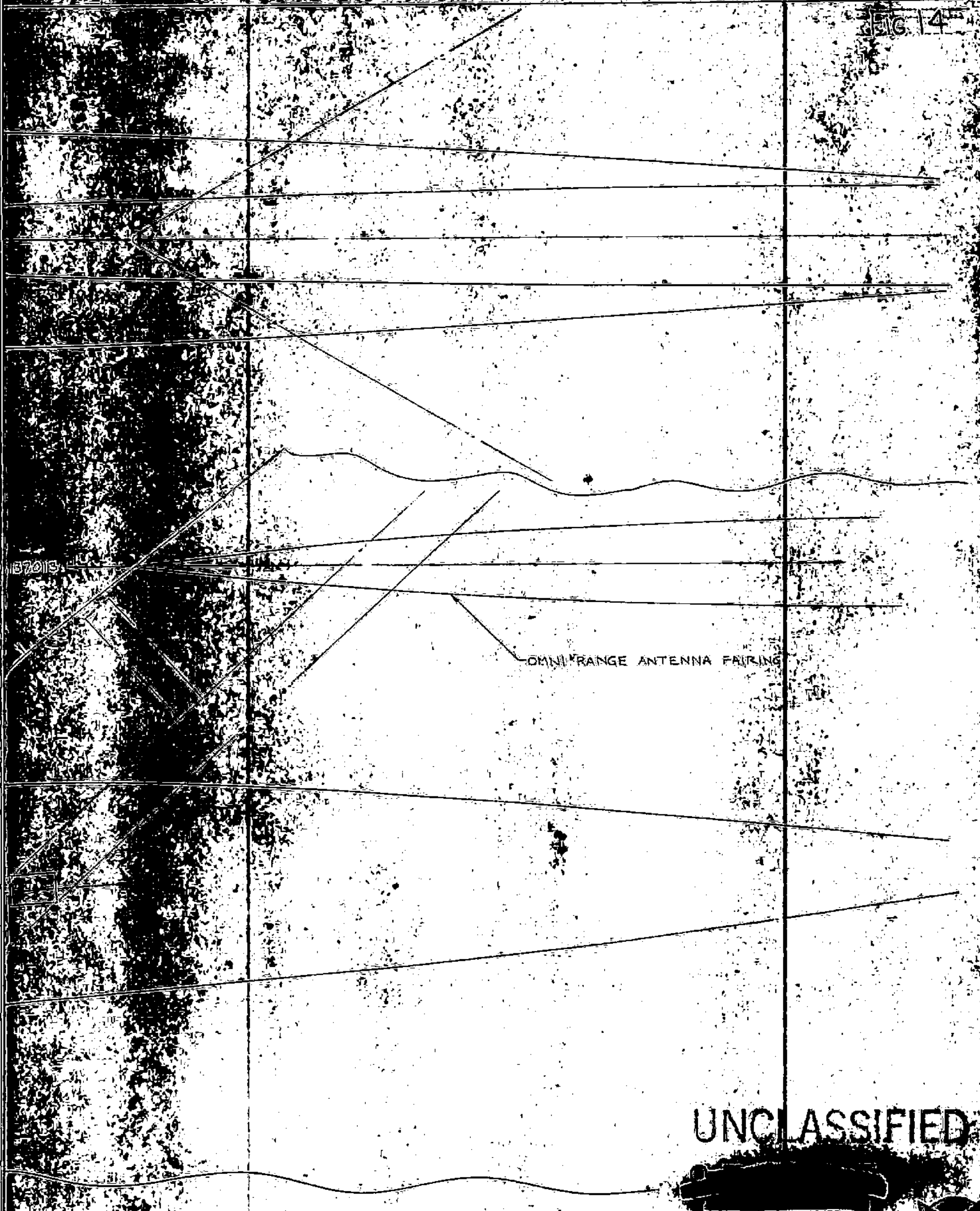
TARGET
TOR (LEFT
DWG. SD-57-08081
ODEL 332 ASP
EEKER (BOTH SIDES)
SD-57-08083
℄ AEROJET [illegible] TARGET
POSITION INDICATOR
(RIGHT SIDE ONLY)
SEE DWG. SD-57-08082
HUGHES I.R.
AUGMENTATION SYSTEM
SEE DWG. SD-57-08084
UNCLASSIFIED
CONVAIR
PRELIMINARY DESIGN DRAWING
STUDY
CONVAIR - SAN DIEGO CALIFORNIA
SD-57-0800

ASSIFIED



ZP [illegible]-053

FIG 14



UNCLASSIFIED

[redacted]

CONVAIR

PRELIMINARY DESIGN DRAWING

INSTALLATION – HUGHES I.R.
AUGMENTATION SYSTEM
LEADING EDGE VERT. TAIL F-102A

BY D. [illegible]	CHECKED	APPROVED	SCALE FULL	DATE 5-26-57

CONVAIR–SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION
SD-57-08084

[redacted]

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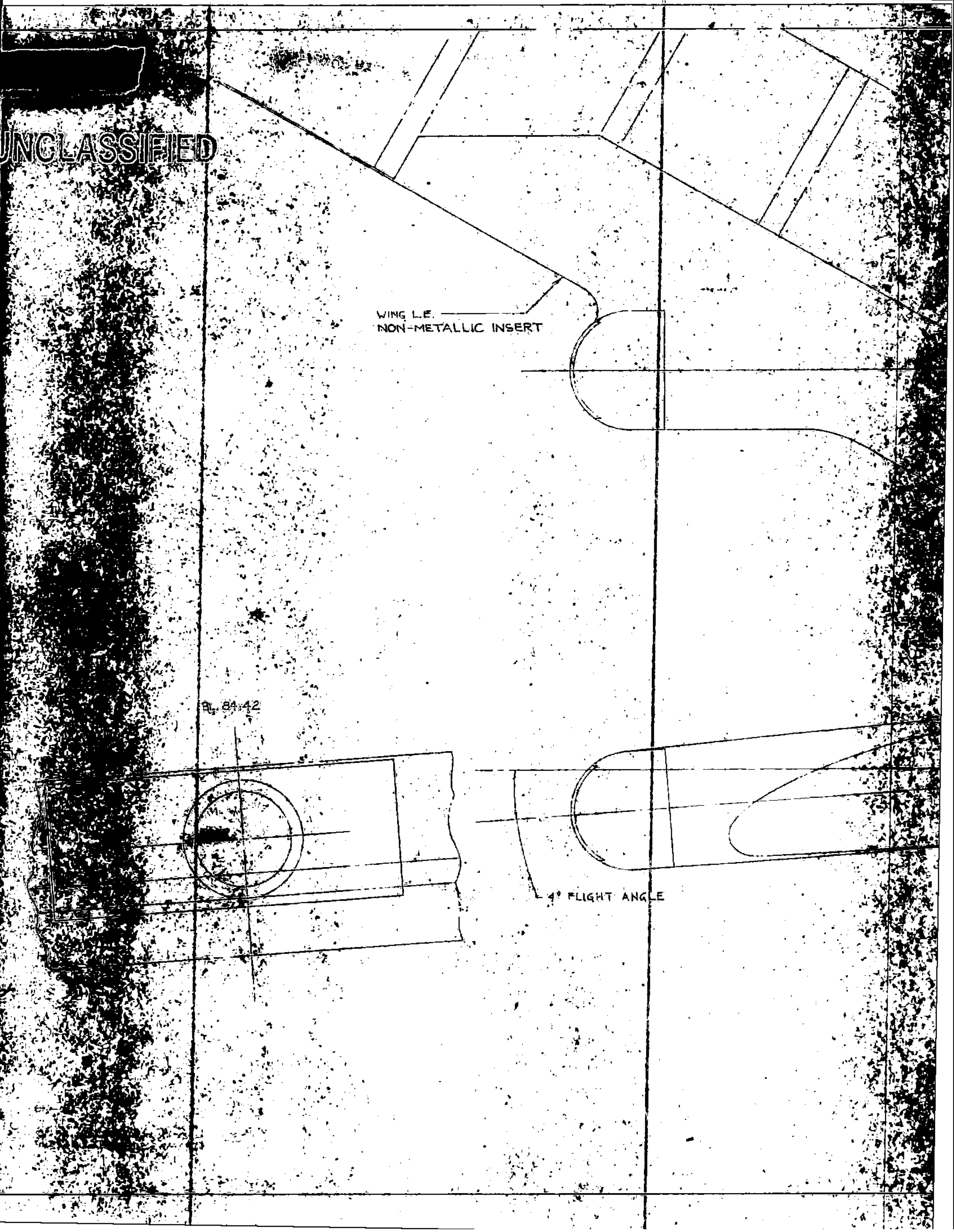
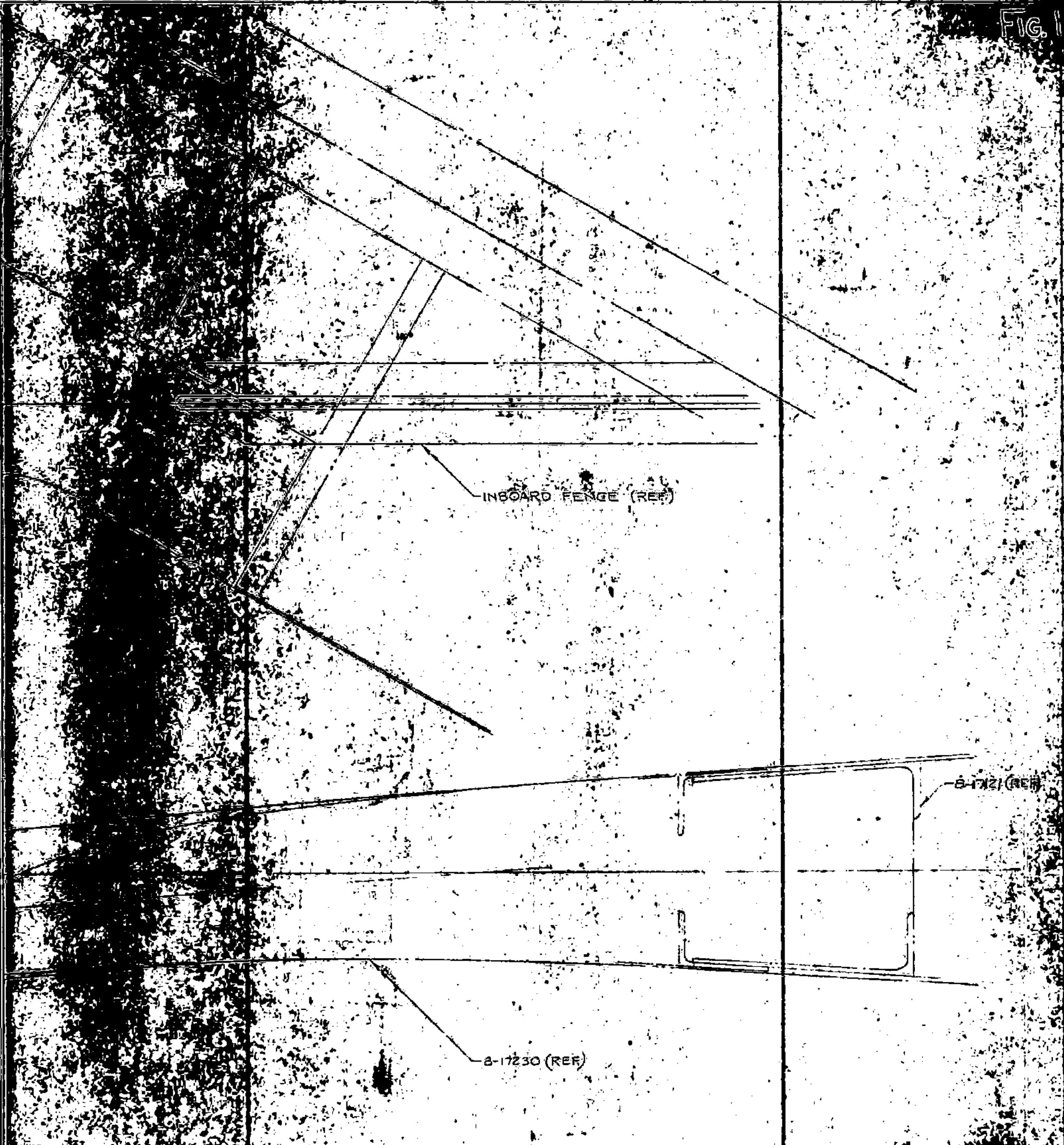


ZP-B-10

FIG. 12



CONVAIR

PRELIMINARY DESIGN DRAWING

INSTALLATION - AVION MODEL
ASP OPTICAL SEEKER
LEADING EDGE WING F-102A

CHECKED APPROVED SCALE

CONVAIR - SAN DIEGO, CALIFORNIA
DIVISION OF GENERAL DYNAMICS CORPORATION

[redacted]

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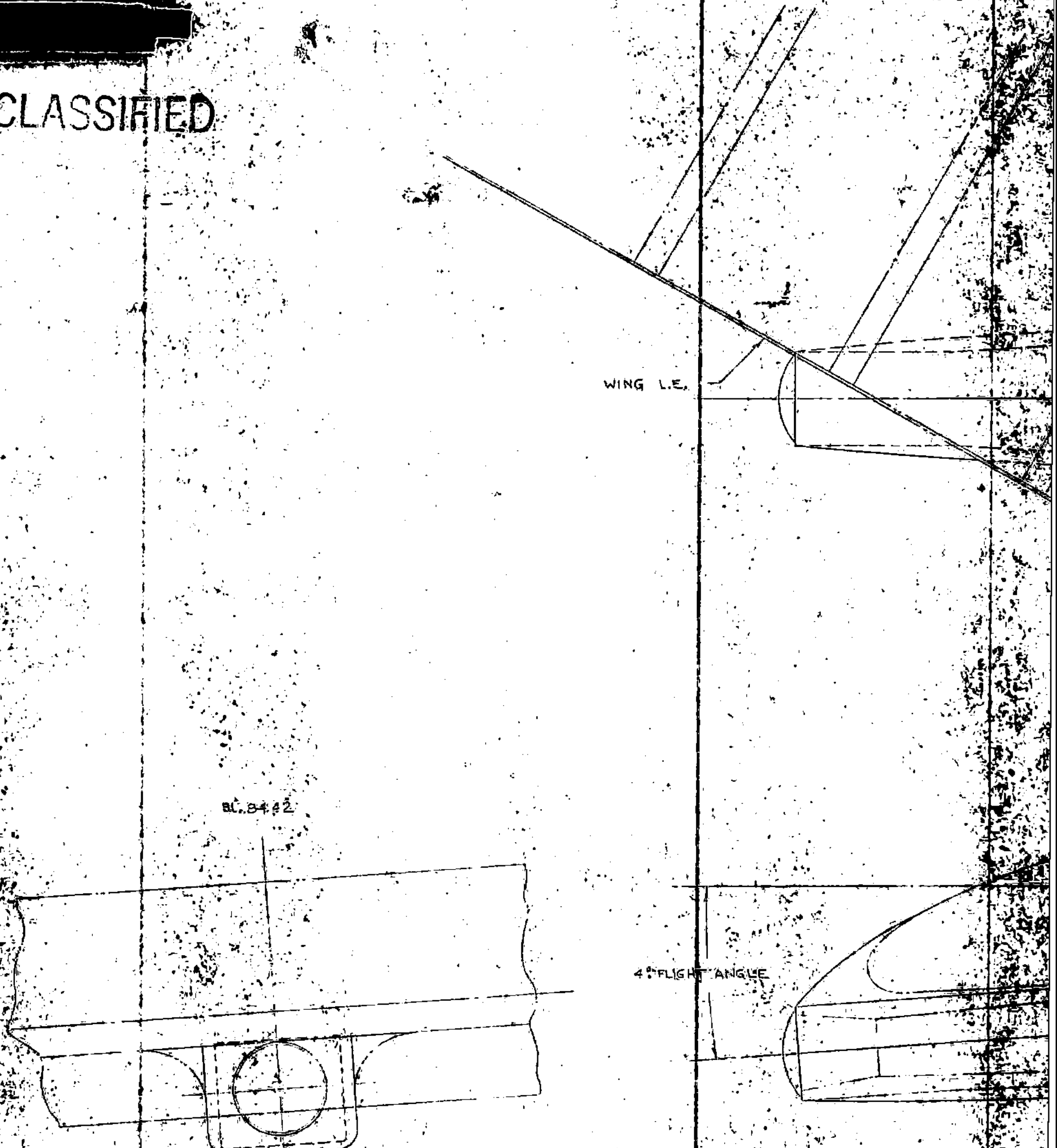
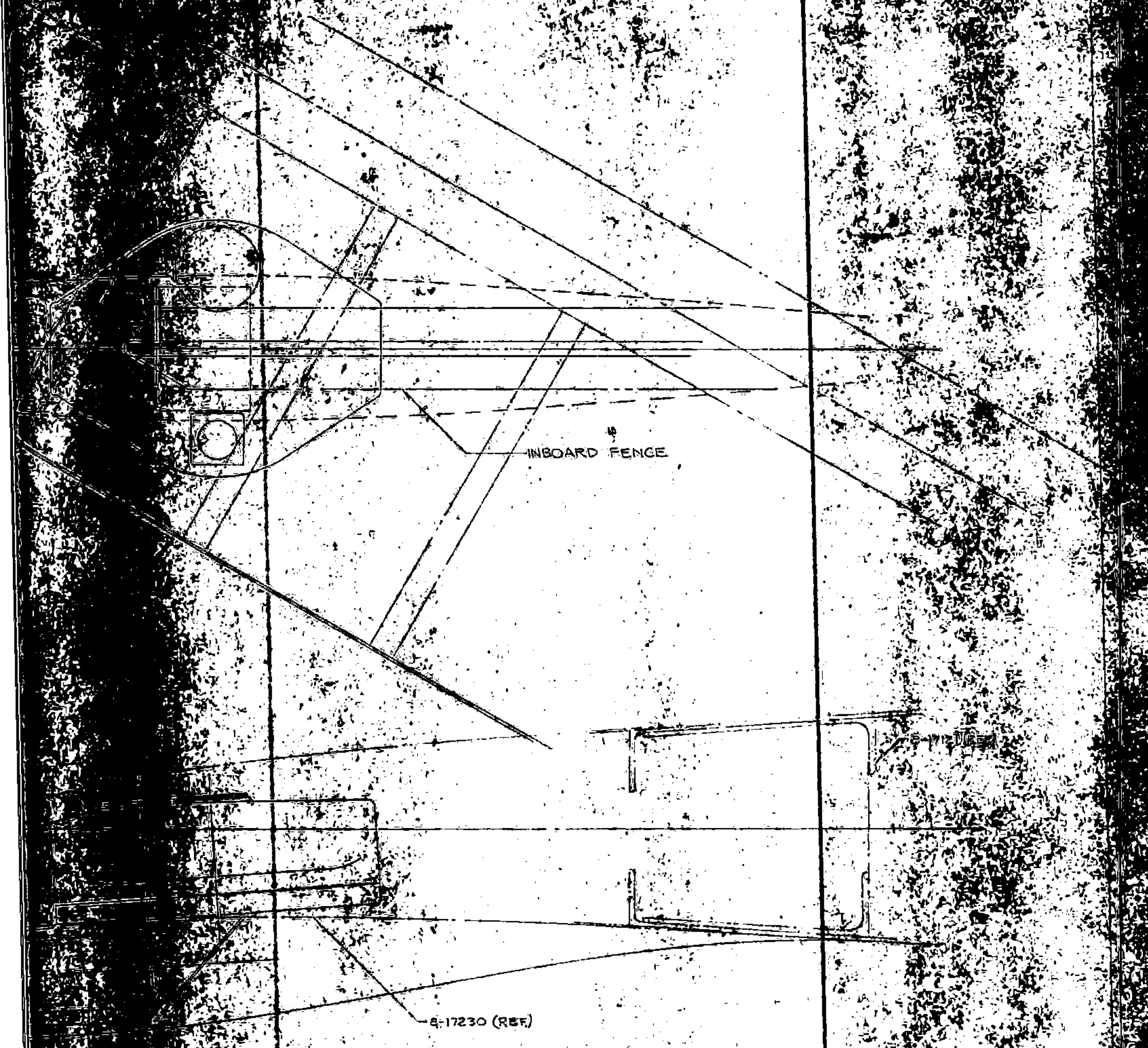


Fig. 8



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CONVAIR

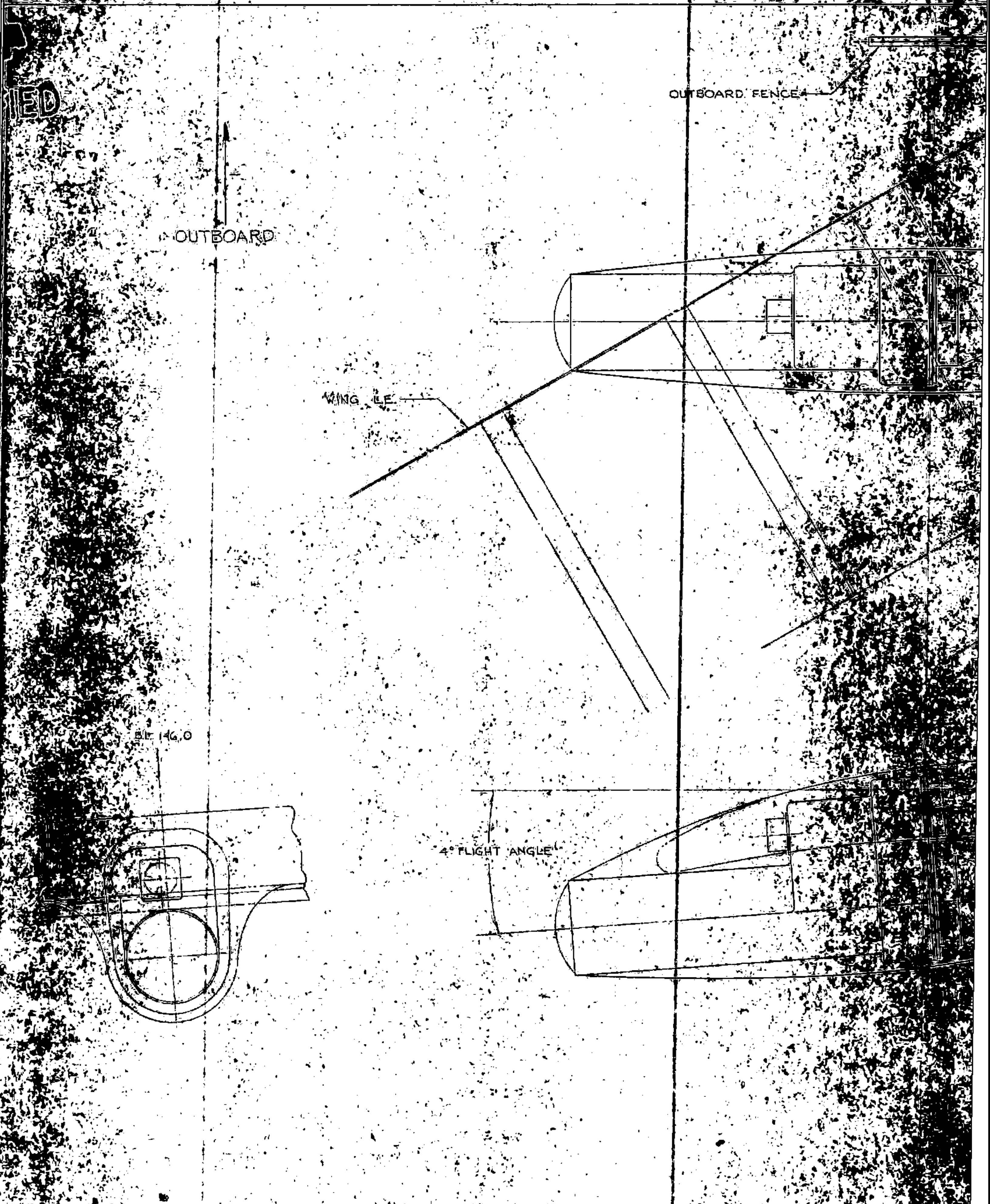
PRELIMINARY DESIGN DRAWING

INSTALLATION-AEROJET S-11
TARGET POSITION INDICATOR
LEADING EDGE WING F-102A

CHECKED APPROVED SCALE

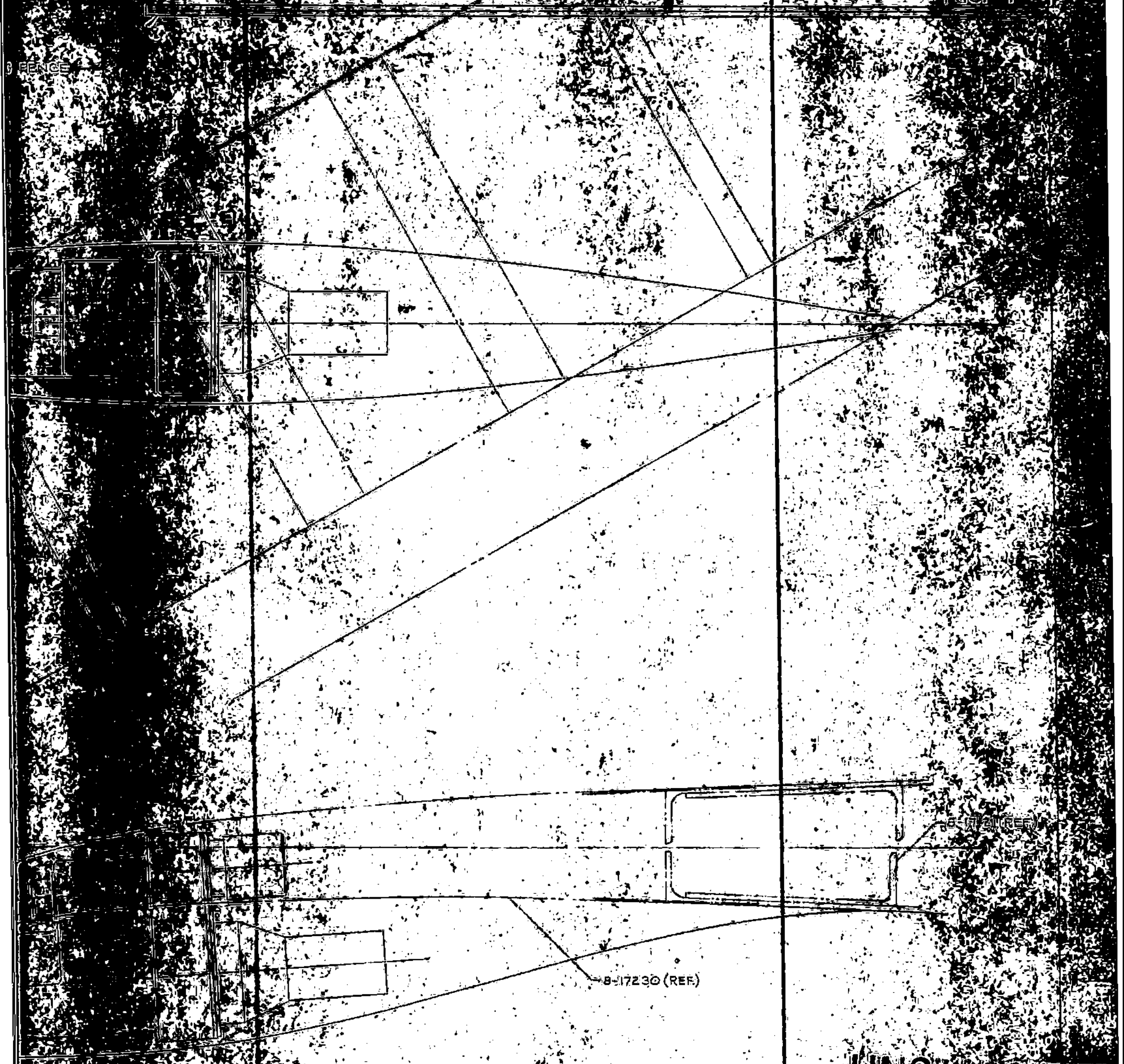
CONVAIR-SAN DIEGO, CALIFORNIA SD-57-0808
A DIVISION OF GENERAL DYNAMICS CORPORATION

IED



ZP-9

FIG. 9



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[redacted]

CONVAIR

PRELIMINARY DESIGN DRAWING

INSTALLATION, AEROJET
TARGET POSITION INDICATOR
LEADING EDGE WING F-102A

BY | CHECKED | APPROVED | SCALE FULL | DATE

CONVAIR — SAN DIEGO, CALIFORNIA
DIVISION OF GENERAL DYNAMICS CORPORATION

SD-57-080

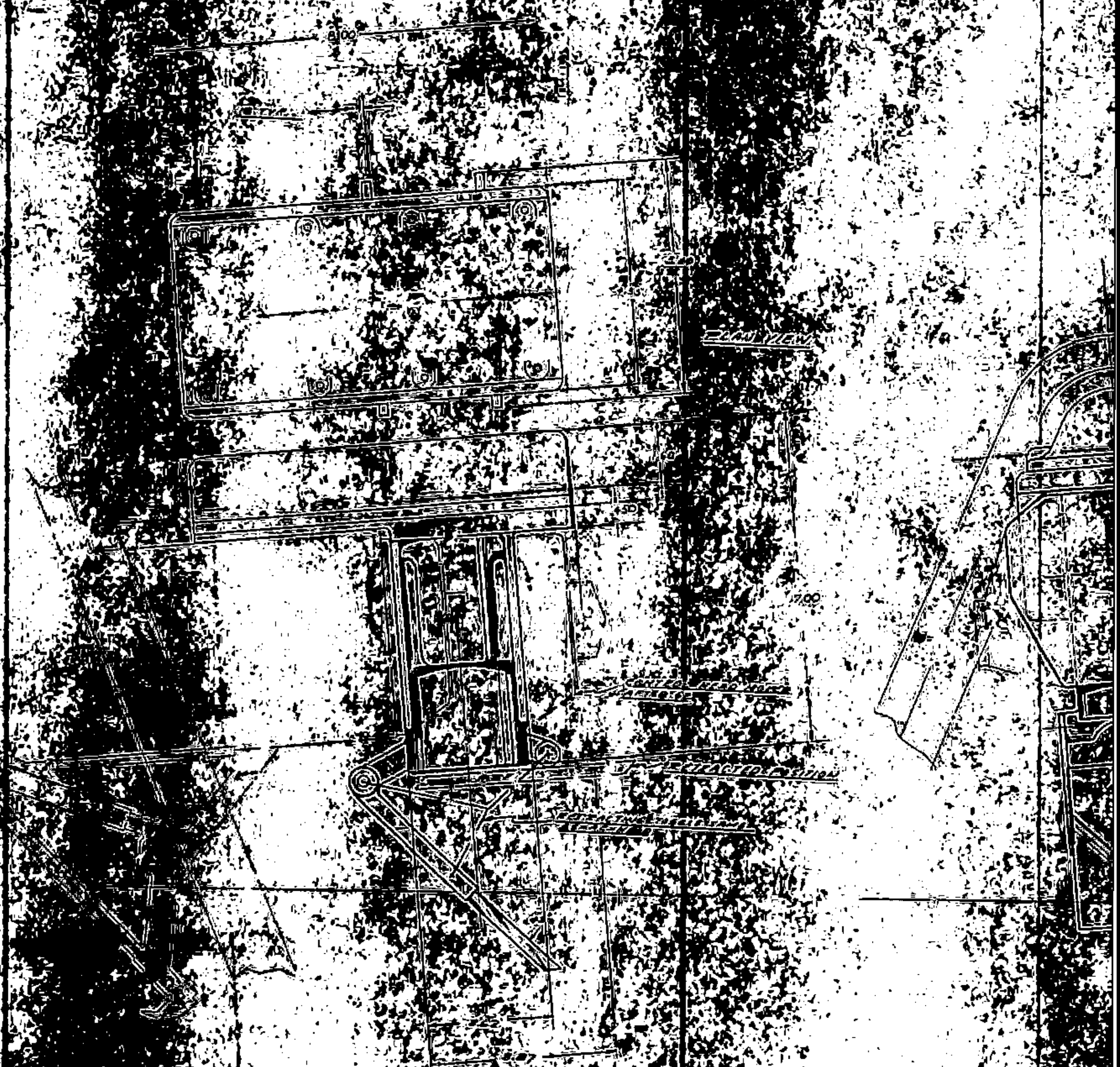
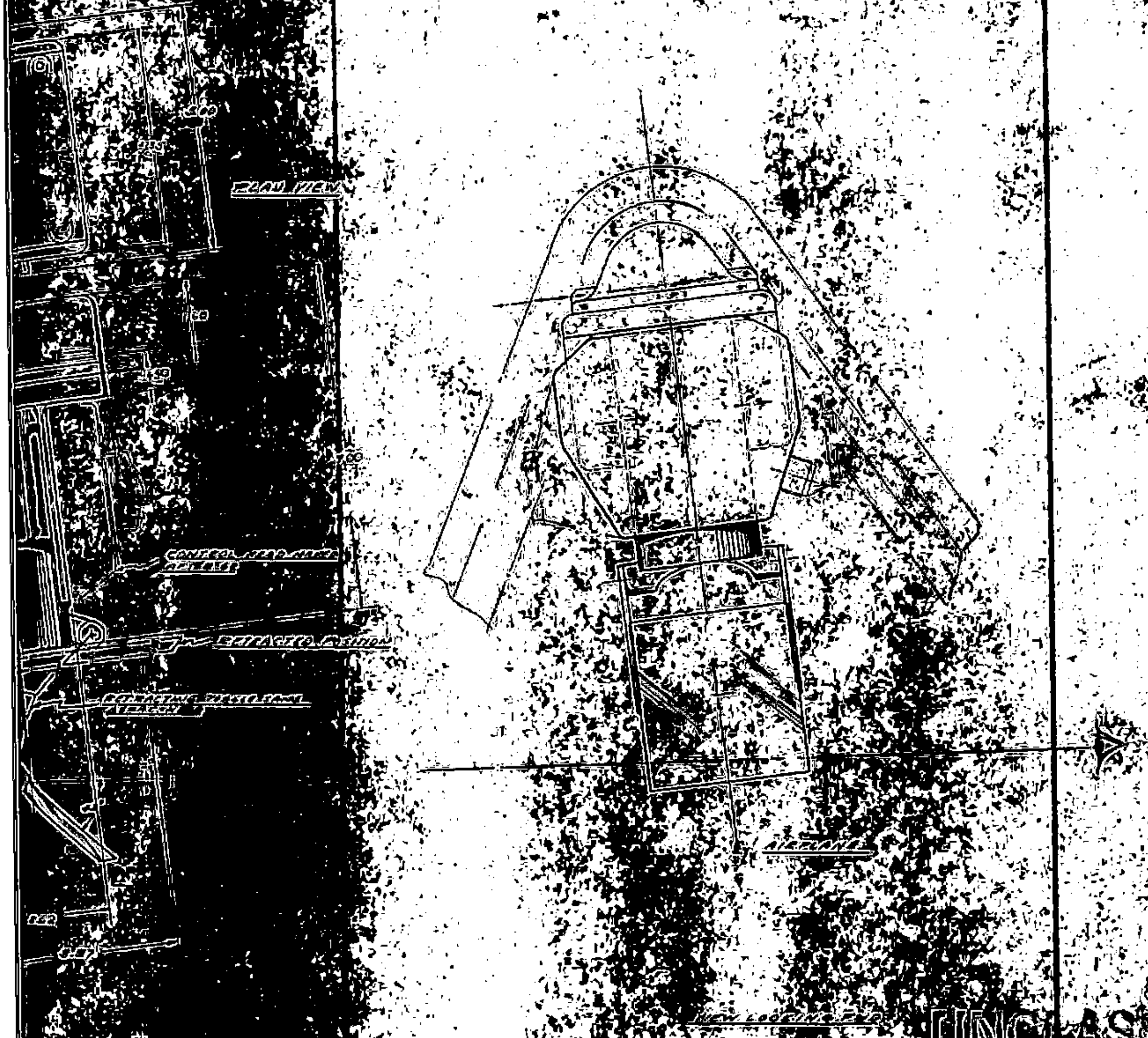

8.00
PLAN VIEW
17.00
RETRACTED POSITION

Fig 10



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[redacted]

INDICATOR
FOR F-102 AIRCRAFT
INSTALLATION

0045629

ZP-8-042
ZP-8-042

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP- 8 -042 (Rev.)
DATE 15 August 1956
MODEL

[redacted]

UNCLASSIFIED

TITLE
STUDY
INSTALLATION OF THE GAR-X MISSILE IN THE F-102A AIRPLANE

ZP-8-042
Revised 15 August 1 9 5 6

PREPARED BY R. C. Lingley
R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY E R Tuttle for
A. R. Lambert

Classification Canceled
CD Form 254 10 June 60
Authorized by: Date
130 6-19-68
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FORM 1612 A-4

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INTRODUCTION

This preliminary study of the installation of the GAR-X missile in the F-102A airplane has been prepared at the request of the Air Force. Three GAR-X missile configurations have been proposed by Hughes Aircraft Co.

1. 133.50" long, 625 lbs. (Ref. HAC Drawing SP-30865)
2. 113.90" long, 607 lbs. (Ref. HAC Drawing SP-30876)
3. 113.90" long, 563 lbs. (Ref. HAC Drawing SP-30875)

It is assumed that ultimately only one of the above configurations will be produced, hence no provisions for interchangeability have been considered.

Due to the anticipated operational date of 1959 for the GAR-X, it is assumed that the entire modification would be accomplished in a recycling program since the production schedule is not compatible with the weapon availability schedule.

Studies show that the most practical arrangement would be one GAR-X missile in the center bay in place of the existing two Falcon missiles. This would confine the modification to the center bay only which, on a retrofit basis, would minimize the task. The existing four Falcons in the outboard bays would remain unchanged. The (24) 2.75" rockets in the doors would be removed.

This change would provide a versatile armament load of one GAR-X missile, two radar Falcons and two infra-red Falcons insuring flexibility of armament selection.

Convair has been conducting tests at Holloman Air Development Center on an ejection system for displacing the MB-1 rocket from the airplane. The success of these tests to date indicate that it is a practical method of carrying the weapon. For this study, it is assumed that a similar method will be used for the GAR-X installation. If it develops that certain characteristics peculiar to the GAR-X preclude the use of an ejection system, then further studies will be necessary to devise a satisfactory displacement gear.

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SUMMARY

Conversion of the F-102 airplanes to carry one GAR-X missile, while retaining four Falcon missiles, can be accomplished with minor structural changes and a slight modification to the armament and related systems. (Reference Drawing SD-56-08052)

The GAR-X missile will be installed in the center missile bay in place of two Falcon missiles. The existing four Falcon missiles in the outboard bays will be retained.

The long (133.50") GAR-X will clear the existing missile bay beams by indenting the revised center doors; however, due to the larger folded geometry of the short (113.90") weapons it will be necessary to cut local clearance holes in the missile bay beams and to modify stiffeners accordingly, as well as providing indentations inside the center doors. In addition a 1" fairing will be required on the outside of the doors to provide for the fixed wing portion of the short GAR-X.

Rockets and their provisions will be removed from the missile bay doors. The center missile bay doors will be made shorter and revised to accommodate the GAR-X missile. The forward end of the center missile bay will be faired over, creating a compartment for the marker beacon antenna (now located on the center bay door) and future equipment growth.

The missile bay doors will be sequenced to open individually to accommodate the particular armament selected. In addition the outboard flipper doors will be sequenced to open simultaneously with the center bay doors in order to relieve pressure in the outboard bays when the GAR-X is fired. Minor changes will be made to the pneumatic and electrical systems to accomplish this feature.

With the proposed armament load, the tactical airplane gross weight will be reduced, and there will be a slight aft C.G. shift which will remain within acceptable limits. It will be necessary to add ballast for the extended range condition.

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SUMMARY (Continued)

The change in gross weight and the C. G. shift will have a negligible effect on the airplane performance.

Installation of the GAR-X weapon will cause no performance degradation of the F-102A airplane due to the basic Convair concept of carrying all armament internally.

Recommendation:

If the fixed wing portion of the short GAR-X can be folded (or eliminated), the extent of the modifications to the airplane structure will be reduced.

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DISCUSSION

The following brief description of the changes required to provide for the GAR-X missile in the F-102A airplane are based upon an ejection system for displacing the weapon. The four Falcon missiles in the outboard bays will be retained and the rockets and their provisions in the missile bay doors will be removed.

A. General

(Reference Drawing SD-56-08052)

1. Replace (2) center Falcons with (1) GAR-X missile. (Any one of the three proposed sizes.)
2. Remove provisions for (24) 2.75" rockets.
3. Add shorter center bay doors and increase travel to clear GAR-X during ejection.
4. Fair over forward end of center bay and provide access into this compartment.
5. Revise structure to provide for ejection loads.
6. Revise door sequencing to accommodate the particular armament being launched.
7. Revise pneumatic, electronic, electrical, and control systems as required by above changes.
8. Modify missile bay beams to clear the wings and fins of the short GAR-X. (Not required for long GAR-X.)

B. Armament Group

1. Remove (2) Falcon displacing gear from central missile bay.
2. Remove provisions for rockets. (The existing outboard doors may be retained by removing rocket tubes and blast pans.)

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3. Add ejection mechanism and provisions for (1) GAR-X missile.

4. Add shorter center bay doors from station 245 to station 405.75.

5. Add indentations inside the center doors to provide clearance for the GAR-X wings and fins. Add a 1" fairing on outside of doors to provide for the fixed wing portion of the short GAR-X.

6. Modify missile bay beams to provide for relocation of center door actuators and removal of existing trunnion supports for two center Falcons. Modify hinge for increased travel of center doors.

7. Add local cutouts in missile bay beams and revise stiffeners to provide clearance for the larger folded geometry of the short GAR-X. (Not required for long GAR-X.)

C. Fuselage Group

1. Revise structure to provide for ejection loads of the long GAR-X missile. The following frames are affected: Stations 329.96, 336.53, 349.75, 356.75 and 363.75.

2. For the short GAR-X revise stations 336.53, 343.10, 349.75, 363.75 and 370.75.

3. Add structure to fair over forward end of center missile bay stations 217 to 245. Add partition between missile bay beams at station 245. Provide access door into this compartment.

D. Mechanism Group

1. Revise center bay door actuator mechanism to provide for shorter doors and increased travel of door in open position.

[redacted]

E. Pneumatic Group

1. Revise pneumatic system to provide for revision to center missile bay doors.
2. Revise sequencing of doors to provide for the particular armament being launched. (Note: Flipper doors must be opened simultaneously with center doors to relieve pressure in the outboard bays when the GAR-X is fired.)
3. Relocate (4) air flasks to center bay forward compartment.

F. Electronics Group

1. Remove provisions for (2) center Falcons and (24) 2.75" rockets.
2. Add provisions for (1) GAR-X missile in center bay.
3. Add missile umbilical plug for GAR-X.
4. Relocate marker beacon antenna from center bay door to forward end of bay.
5. Revise missile auxiliaries to accommodate the GAR-X. (It is estimated that two units will be replaced by four.)

G. Electrical Group

1. Delete harnesses and provisions for two center Falcon missiles.
2. Delete rocket wiring and provisions.
3. Provide harnesses and provisions for one GAR-X missile.
4. Revise armament control panel in cockpit.
5. Revise miscellaneous harnesses, junctions boxes, relay boxes, switches, and circuit breakers to provide for above changes.

[redacted]

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H. Controls Group

1. Revise elevon control cable and relocate turnbuckles in center missile bay.

[redacted]

See the following pages for weight summary, C.G. envelope, and a drawing of the proposed installation.

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F-102A AIRPLANE WEIGHT SUMMARY

	Status 29 Tactical 6 Falcon 24 Rockets	1 GAR-X & 4 Falcons
Wing Group	3051	3051
Tail Group	559	559
Body Group	3371	3472
Alighting Gear Group	1045	1045
Surface Controls Group	435	435
Engine Section	39	39
Propulsion Group	(6400)	(6400)
Engine Installation	5015	5015
Air Induction System	697	697
Exhaust System	132	132
Cooling System	14	14
Lubricating System	27	27
Fuel System	396	396
Engine Controls	25	25
Starting System	94	94
Fixed Equipment Group	(4377)	(4201)
Instruments & Navigational Equipment Group	145	145
Hydraulic and Pneumatic Group	317	317
Electrical Group	573	573
Electronics Group	2248	2259
Armament Group	592	406
Furnishings and Equipment Group	213	212
Air Conditioning and Anti-Icing Group	237	237
Auxiliary Gear Group	52	52
WEIGHT EMPTY	19277	19202
USEFUL LOAD		
Pilot, Survival Kit and Parachute	248	248
Fuel, Unusable (JP-4) 23.7 Gallons	154	154
Fuel, Usable (JP-4) 1,050 Gallons	6825	6825
Oil, Trapped and engine	46	46
Falcon Missile GAR 1-A and/or 1-C	762	508
Missile Launchers	222	148
2.75" Rockets	432	0
GAR-X		625 *
NORMAL TAKE-OFF GROSS WEIGHT	27966	27756

* See next page for 607# and 563# GAR-X

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F-102A AIRPLANE WEIGHT SUMMARY (Cont'd.)

Normal Take-Off Gross Weight with 625# GAR-X	27756
Less	
GAR-X	- 625
Plus	
GAR-X	+ 607
Beef-up to Structure	+ 37
Beef-up to Doors	+ 56
Normal Take-Off Gross Weight with 607# GAR-X	27831

Normal Take-Off Gross Weight with 625# GAR-X	27756
Less	
GAR-X	- 625
Plus	
GAR-X	+ 563
Beef-up to Structure	+ 21
Beef-up to Doors	+ 19
Normal Take-Off Gross Weight with 563# GAR-X	27734

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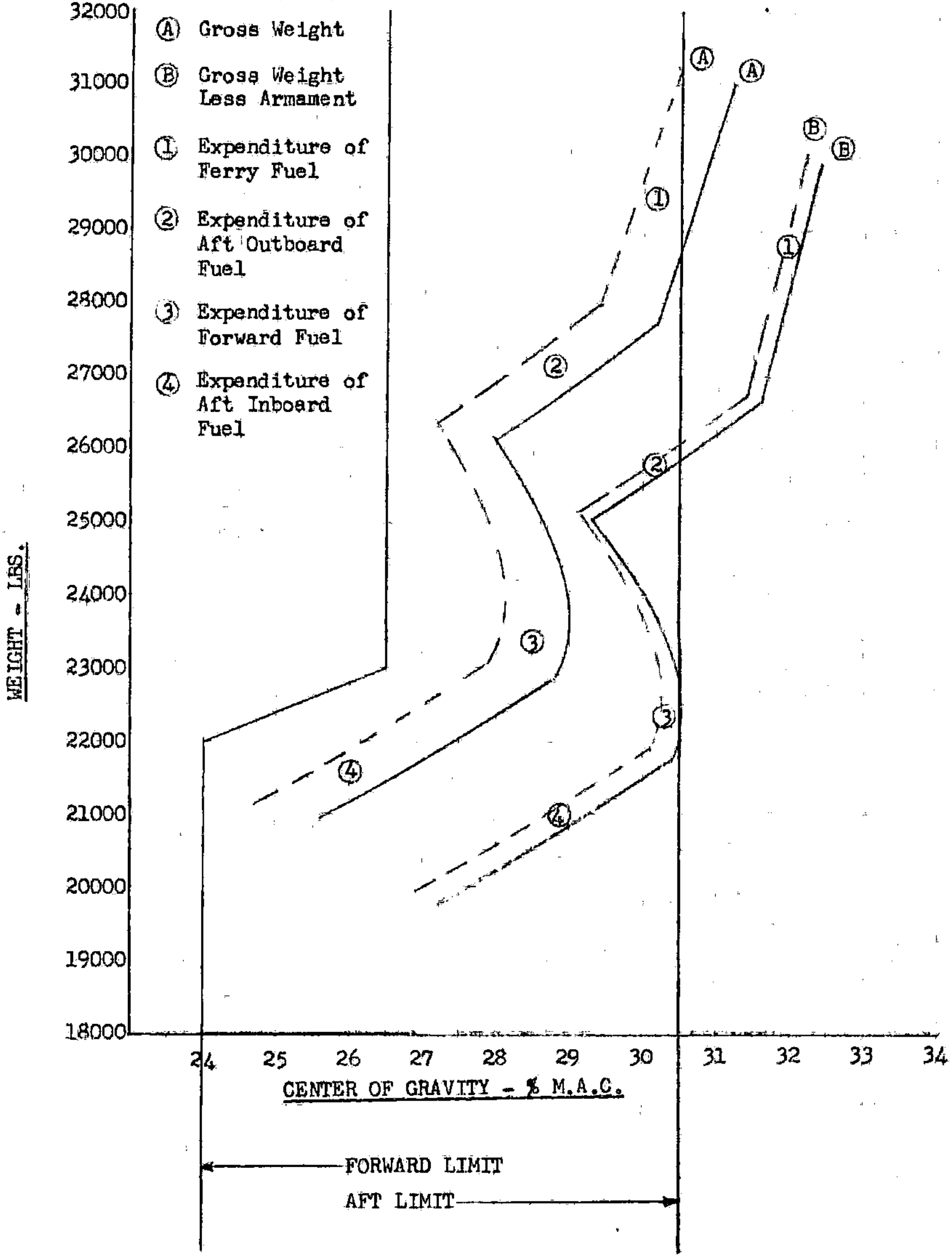
F-102A
AIRPLANE CENTER OF GRAVITY CHART

Dotted Line Represents F-102A Tactical Airplane with 6 Falcons, and 24 2.75 Inch Rockets

Solid Line represents F-102A Tactical Airplane with 4 Falcons, and 1 GAR-X @ 625 Lbs.



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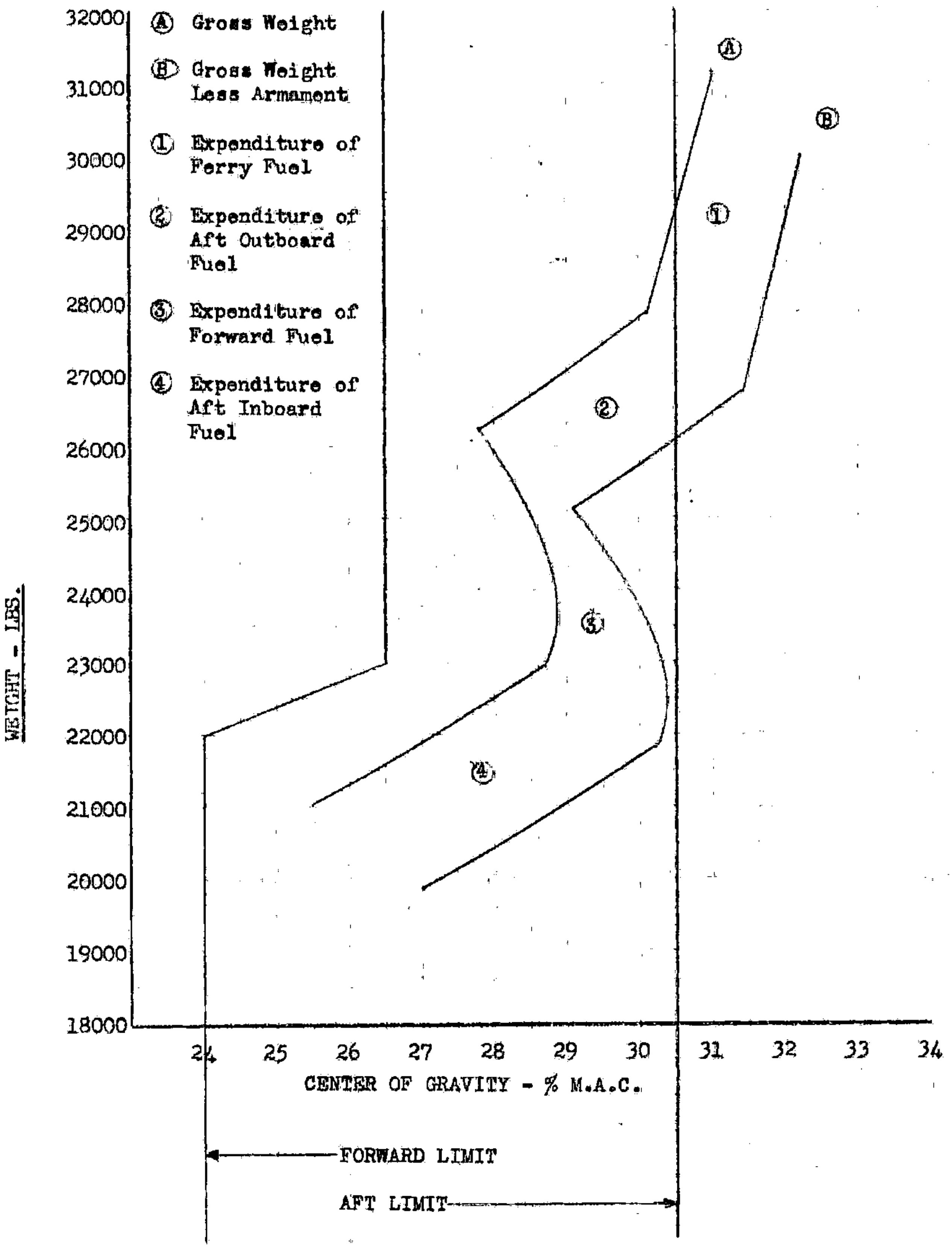
F-102A
AIRPLANE CENTER OF GRAVITY CHART

F-102A Tactical Airplane with 4 Falcons
and 1 GAR-X @ 607 Lbs.



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F-102A
AIRPLANE CENTER OF GRAVITY CHART

F-102A Tactical Airplane with 4 Falcons
and 1 GAR-X @ 563 lbs.

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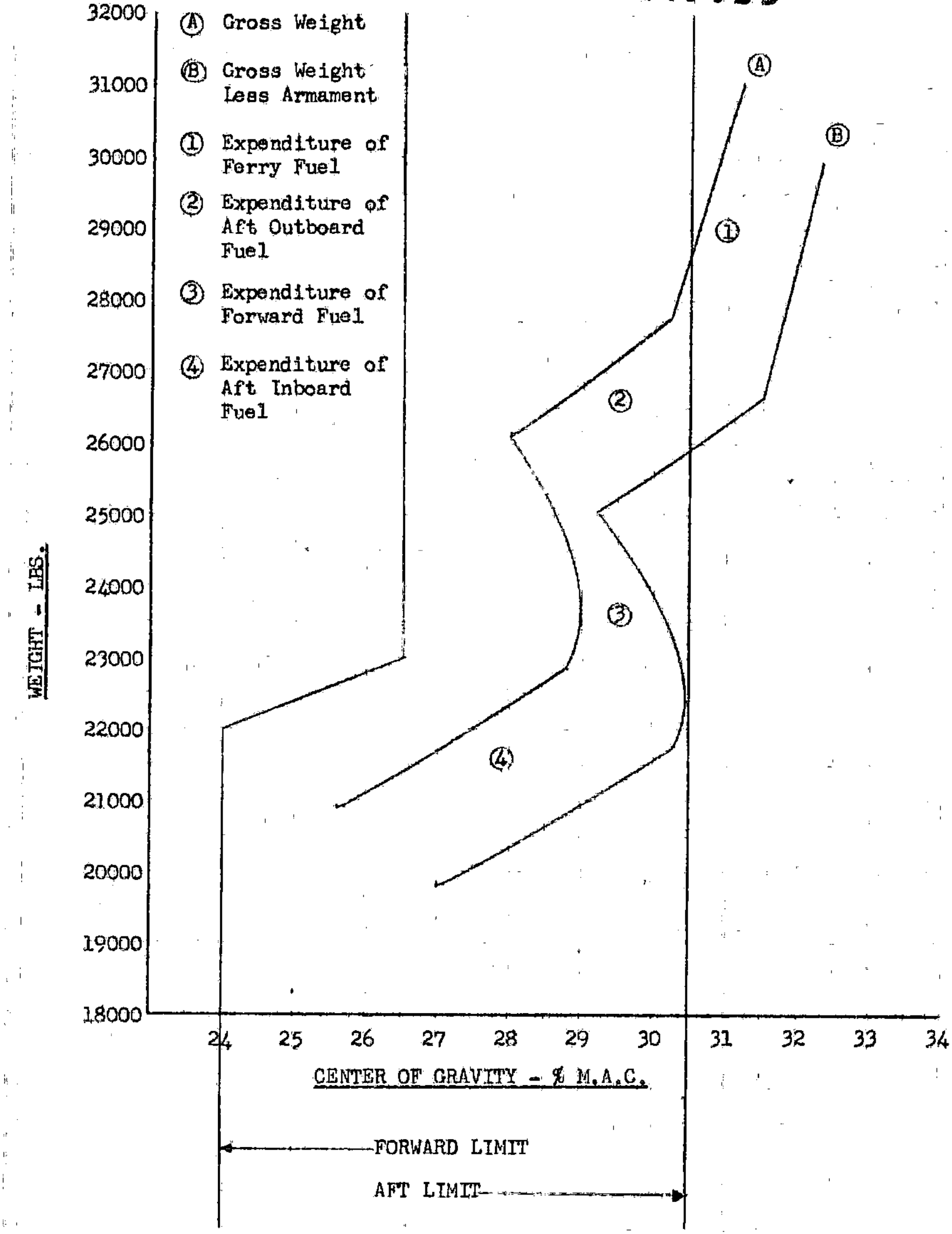

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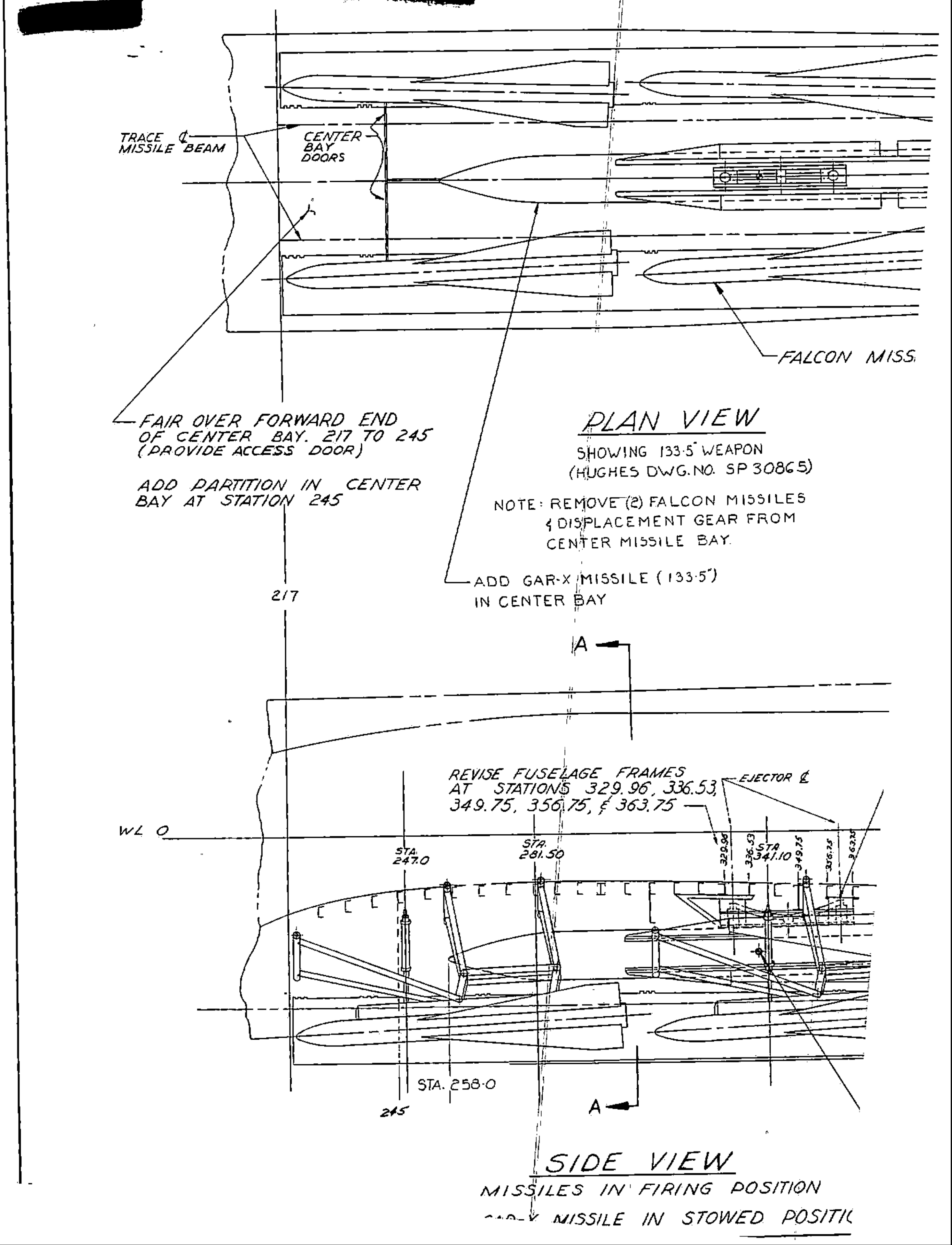


FAIR OVER FORWARD END OF CENTER BAY. 217 TO 245 (PROVIDE ACCESS DOOR)

ADD PARTITION IN CENTER BAY AT STATION 245

PLAN VIEW

SHOWING 133.5" WEAPON
(HUGHES DWG. NO. SP 30865)

NOTE: REMOVE (2) FALCON MISSILES
& DISPLACEMENT GEAR FROM
CENTER MISSILE BAY.

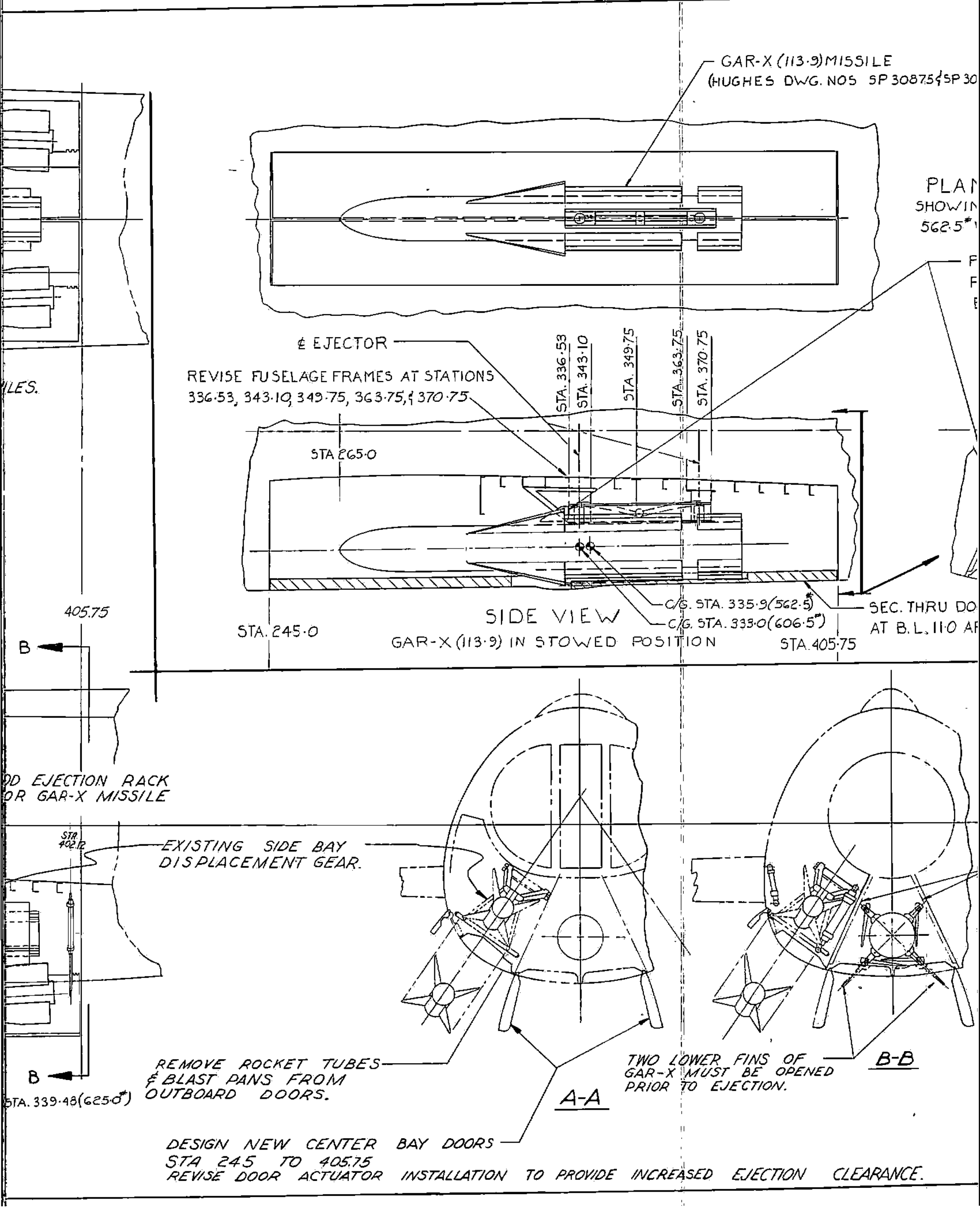
ADD GAR-X MISSILE (133.5")
IN CENTER BAY

REVISE FUSELAGE FRAMES AT STATIONS 329.96, 336.53, 349.75, 356.75, & 363.75

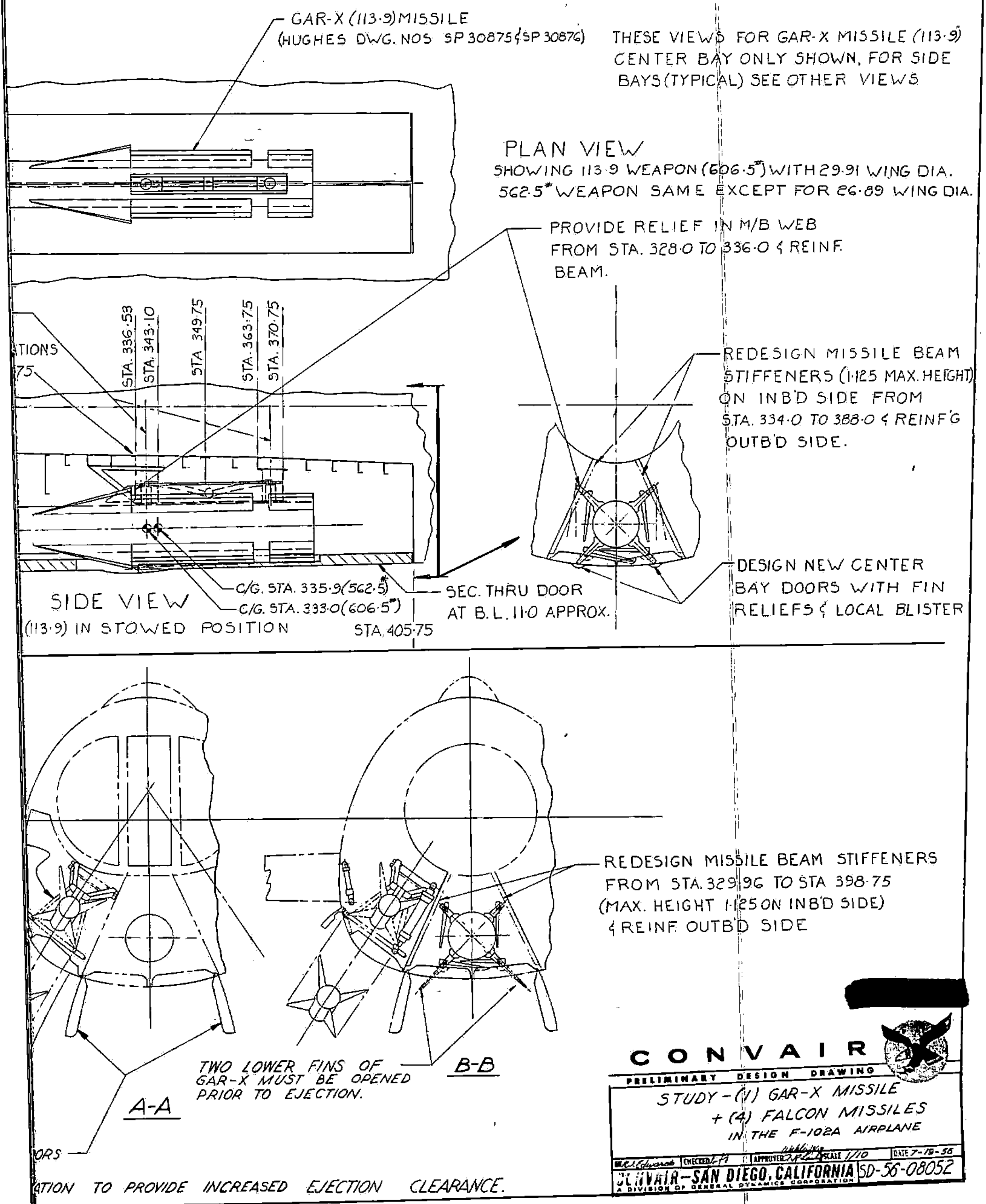
SIDE VIEW

MISSILES IN FIRING POSITION

GAR-X MISSILE IN STOWED POSITI


GAR-X (113·9) MISSILE
(HUGHES DWG. NOS 5P 30875 & SP 30
PLAN
SHOWIN
562.5#
F
F
E
℄ EJECTOR
REVISE FUSELAGE FRAMES AT STATIONS
336·53, 343·10, 349·75, 363·75, & 370·75
STA. 336·53
STA. 343·10
STA. 349·75
STA. 363·75
STA. 370·75
STA 265·0
C/G. STA. 335·9 (562·5#)
C/G. STA. 333·0 (606·5#)
SIDE VIEW
GAR-X (113·9) IN STOWED POSITION
STA. 245·0
STA. 405·75
SEC. THRU DO
AT B.L. 11·0 A
405.75
ILES.
B
OD EJECTION RACK
OR GAR-X MISSILE
STA 402.12
EXISTING SIDE BAY DISPLACEMENT GEAR.
B
STA. 339·48 (625·0#)
REMOVE ROCKET TUBES & BLAST PANS FROM OUTBOARD DOORS.
A-A
TWO LOWER FINS OF GAR-X MUST BE OPENED PRIOR TO EJECTION.
B-B
DESIGN NEW CENTER BAY DOORS
STA 245 TO 405.75
REVISE DOOR ACTUATOR INSTALLATION TO PROVIDE INCREASED EJECTION CLEARANCE.

2P-8-042
Page 10


GAR-X (113·9) MISSILE
(HUGHES DWG. NOS SP 30875 & SP 30876)
THESE VIEWS FOR GAR-X MISSILE (113·9)
CENTER BAY ONLY SHOWN, FOR SIDE
BAYS (TYPICAL) SEE OTHER VIEWS
PLAN VIEW
SHOWING 113·9 WEAPON (606·5#) WITH 29·91 WING DIA.
562·5# WEAPON SAME EXCEPT FOR 26·89 WING DIA.
PROVIDE RELIEF IN M/B WEB
FROM STA. 328·0 TO 336·0 & REINF.
BEAM.
ATIONS
75
STA. 336·53
STA. 343·10
STA. 349·75
STA. 363·75
STA. 370·75
REDESIGN MISSILE BEAM
STIFFENERS (1·125 MAX. HEIGHT)
ON INB'D SIDE FROM
STA. 334·0 TO 388·0 & REINF'G
OUTB'D SIDE.
C/G. STA. 335·9 (562·5#)
C/G. STA. 333·0 (606·5#)
SEC. THRU DOOR
AT B.L. 11·0 APPROX.
DESIGN NEW CENTER
BAY DOORS WITH FIN
RELIEFS & LOCAL BLISTER
SIDE VIEW
(113·9) IN STOWED POSITION
STA. 405·75
REDESIGN MISSILE BEAM STIFFENERS
FROM STA. 329·96 TO STA 398·75
(MAX. HEIGHT 1·125 ON INB'D SIDE)
& REINF. OUTB'D SIDE
TWO LOWER FINS OF
GAR-X MUST BE OPENED
PRIOR TO EJECTION.
B-B
A-A
ORS
ATION TO PROVIDE INCREASED EJECTION CLEARANCE.

[redacted]

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PRELIMINARY DESIGN DRAWING

STUDY - (1) GAR-X MISSILE
+ (4) FALCON MISSILES
IN THE F-102A AIRPLANE

R.J. Edwards	CHECKED	APPROVED	SCALE 1/10	DATE 7-19-56

CONVAIR—SAN DIEGO, CALIFORNIA
A DIVISION OF GENERAL DYNAMICS CORPORATION
SD-56-08052

ZP-8-032

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CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZP-8-032
DATE 24 Jan. 1956
[redacted]DEL 8

[redacted]

TITLE

STUDY

REVISED WINDSHIELD AND CANOPY

EXISTING TF-102A AIRPLANE

ZP-8-032

24 Jan. 1956

PREPARED BY R. C. Lingley
R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY A. R. Lambert
A. R. Lambert

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INTRODUCTION

Buffet difficulties have been encountered during tests of the TF-102A side-by-side trainer. Preliminary tuft tests have indicated that separation is taking place in the region of the canopy.

Presented herein are the results of an investigation which should result in an improvement in the aerodynamic lines of the windshield and canopy area.

The change has been confined to the area above the longerons.

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SUMMARY

The new windshield and canopy fairing will be tested in the CO-OP wind tunnel. Should the results prove favorable, this change could be incorporated on an airplane for early flight test evaluation.

The basic side-by-side seating concept can be retained.

PERFORMANCE SUMMARY

		Existing TF-102A	With New Wind-shield & Canopy
High Speed	Mach No.	.98	.99
Combat Ceiling	Feet	49,300	49,500
Combat Radius	N. Mi.	316	318

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Discussion: (Ref. Dwg. No. SD-56-08001)

In order to show an appreciable reduction in the height of the canopy, it is necessary to lower the pilots' theoretical eye positions two inches. This is accomplished by lowering the seats with the electrical adjustments. The pilots' down vision over the nose is thereby reduced from 15° to 13-1/2°. (Equivalent to the F-102A.)

All other items in the cockpit remain unchanged.

In profile, the windshield slope is reduced which lowers the height at the parting line frame. The present "V" windshield is replaced with a flat center panel 18" wide. The diagonal panels are then at a less severe angle to the center line, which improves the transition from the windshield flat into the canopy contour.

A new profile line has been established from the reduced windshield slope aft to station 304.

In general, it is lowered in the forward portion and raised aft in order to move the peak of the canopy hump further aft into a more favorable pressure area.

Due to the contour change, it is necessary to design a new access door in place of the existing "ATM" door.

The above changes are confined to the area above the upper longerons and between station 70.5 and station 304.

The following detailed breakdown shows the changes required:

A. Fuselage

1. Redesign upper torque box station 70.5 - 122.
2. Redesign center line beam station 85 - 122.
3. Design windshield posts (BL 9 and BL 22).
4. Redesign windshield longeron.

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5. Redesign and relocate windshield parting frame.

6. Redesign windshield glass panels incorporating 18" wide center panel.

7. Redesign canopy structure:

    a. Parting frame.

    b. Longerons (BL 9 at post to BL 15.38 at hinge).

    c. Corner fittings.

    d. Hinge fittings.

    e. Aft overhead structure.

8. Redesign canopy transparent panels.

9. Redesign canopy island structure.

10. Redesign canopy seals.

11. Revise canopy hook mechanism.

12. Design new parting frame attach fitting at Sta. 159.

13. Reinforce fuselage frame at Sta. 156.90 due to relocation of parting frame.

14. Redesign fuselage frames at Sta. 156 and 164 due to relocation of parting frame.

15. Design new ATM door and frame.

    Note: New lines prevent use of straight line hinge on top center line which will require either gooseneck hinges or removable door.

16. Design new deck frames from ATM door to Sta. 310.25.

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B. Hydraulic System

1. Revise pneumatic tubing due to rain clearing duct changes.
2. Revise air speed support bracket.
3. Revise air speed system and tubing above floor.

C. Electrical System

1. The windshield anti-ice system and canopy de-fog system will require new wiring and schematic diagrams, harnesses and connection diagrams.
2. New windshield anti-ice and de-fog transformers will be required causing revision to present installation in refrigeration area.
3. Add one windshield anti-ice transformer and control box.

D. Air Conditioning System

1. Redesign windshield rain clearing nozzles and one duct assembly forward of Sta. 122.
2. Redesign cabin distribution duct between canted bulkhead and canopy longeron.
3. Redesign cabin safety valve and reroute necessary hoses and tubing attaching to the valve.

E. Controls System

1. Revise teleflex routing to accommodate redesigned canopy hook mechanism.

F. Electronics System

1. Revise installation of upper DME/IFF antenna (use

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same antenna).

2. Revise installation of MG-IOT gyro at Sta. 260.

3. Revise gyro harness.

4. Revise miscellaneous equipment installations in area of upper longeron Sta. 238 to 266 due to revisions in structure.

G. Furnishings

1. Cut-off top of seat rails for canopy clearance.

2. Revise glare shield.

3. Revise glare curtain.

4. Revise stud locations and design new blind flying enclosures.

5. Design new magnetic compass installation.

6. Revise installation of optical sight and track.

It is estimated that these changes will have a negligible effect on weight and balance, and airplane performance.

Weight and balance information and the performance summary of the existing TF-102A are included in the following pages.

WEIGHT SUMMARY

EXISTING TF-102A TRAINER

Wing		3,044	
Tail		315	
Fuselage		3,816	
Landing Gear		1,047	
Surface Controls		466	
Nacelle Group		38	
Propulsion Group		6,317	
Engine	4,980		
Air Induction	612		
Exhaust System	129		
Cooling System	18		
Lubricating System	28		
Fuel System	427		
Engine Controls	36		
Starting System	87		
Instruments and Navigation		164	
Hydraulics and Pneumatics		291	
Electrical		563	
Electronics		2,311	
Armament		592	
Furnishings and Equipment		380	
Air Conditioning and Anti-Icing		237	
Auxiliary Gear		61	
Unaccountable Weight		- 45	
Weight Empty			19,597
Pilots (2)		460	
Fuel		6,979	
Unusable	154		
Wing	6,825		
Oil		46	
Trapped	21		
Engine	25		
Armament		1,366	
Missiles (6)	762		
Rockets (36-2.00")	382		
Launchers	222		
Equipment		36	
Useful Load			8,887
Gross Weight			28,484

Note: Weights are from Actual Weight and Balance Report ZW-8-021 for the TF-102A Airplane.

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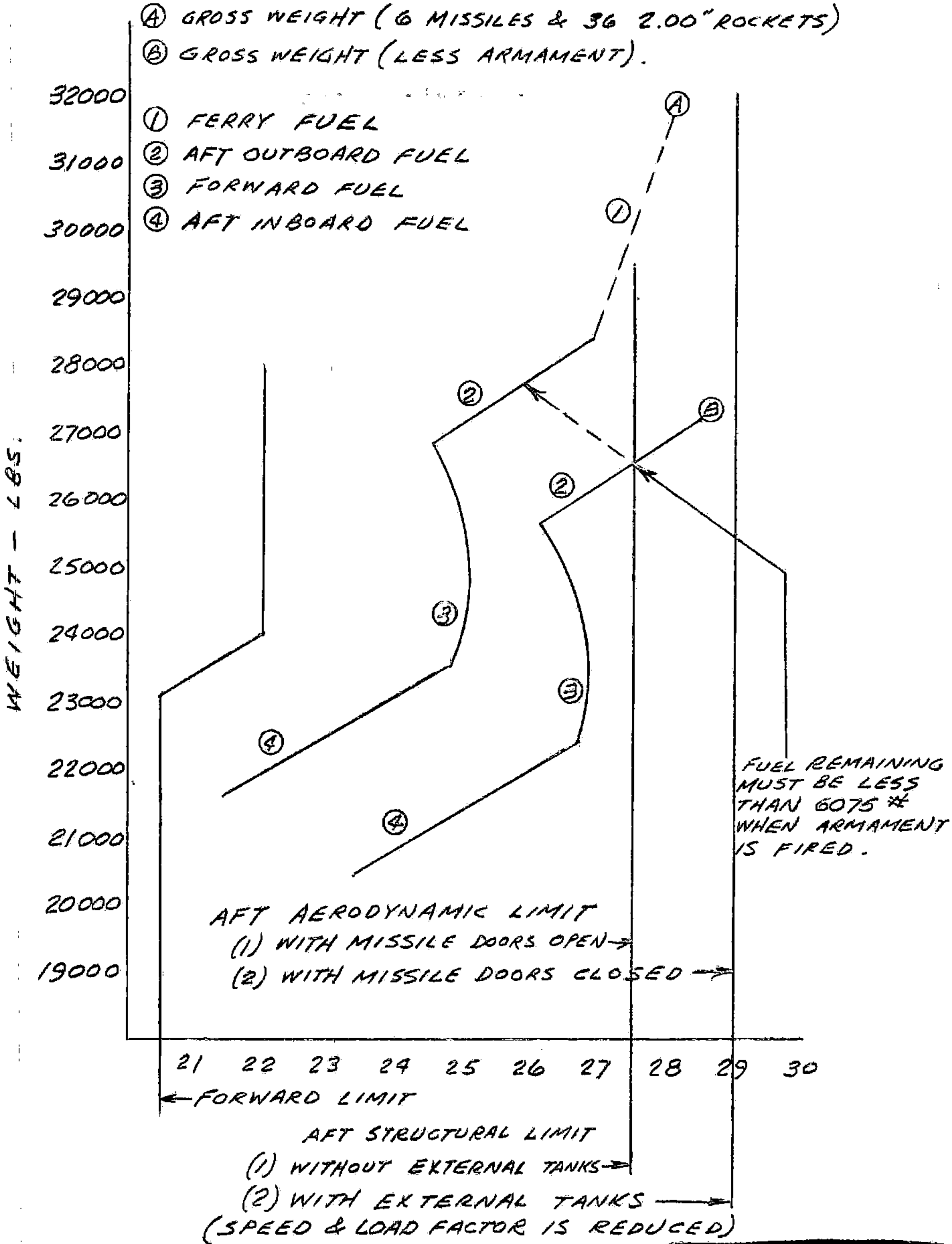
EXISTING TF-102A
CENTER OF GRAVITY – % M.A.C.
(JP-4 FUEL)



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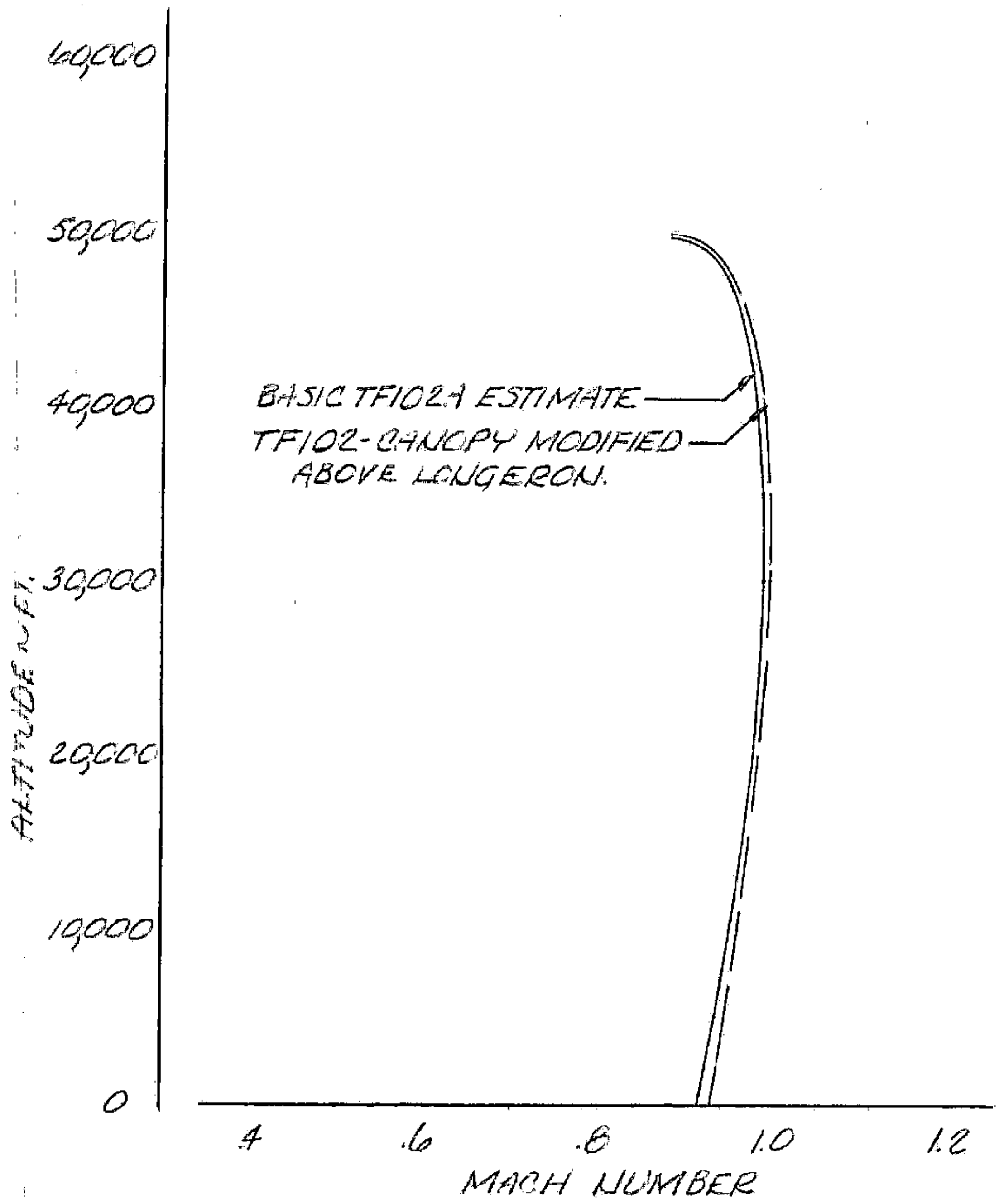
TF-102A REVISED WINDSHIELD AND CANOPY

HIGH SPEED & ALTITUDE

J57-P-23 MILITARY + A.B. POWER



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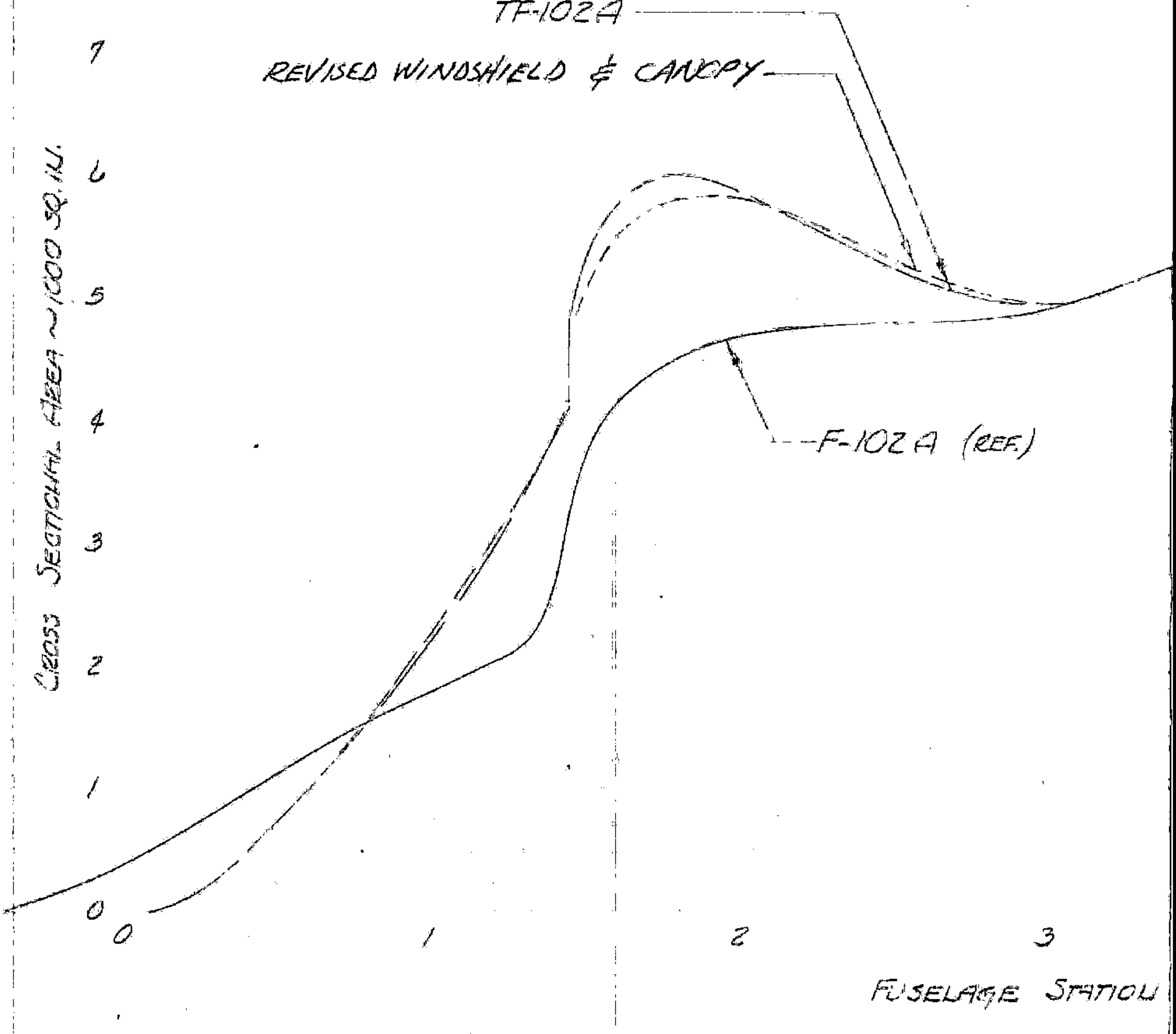
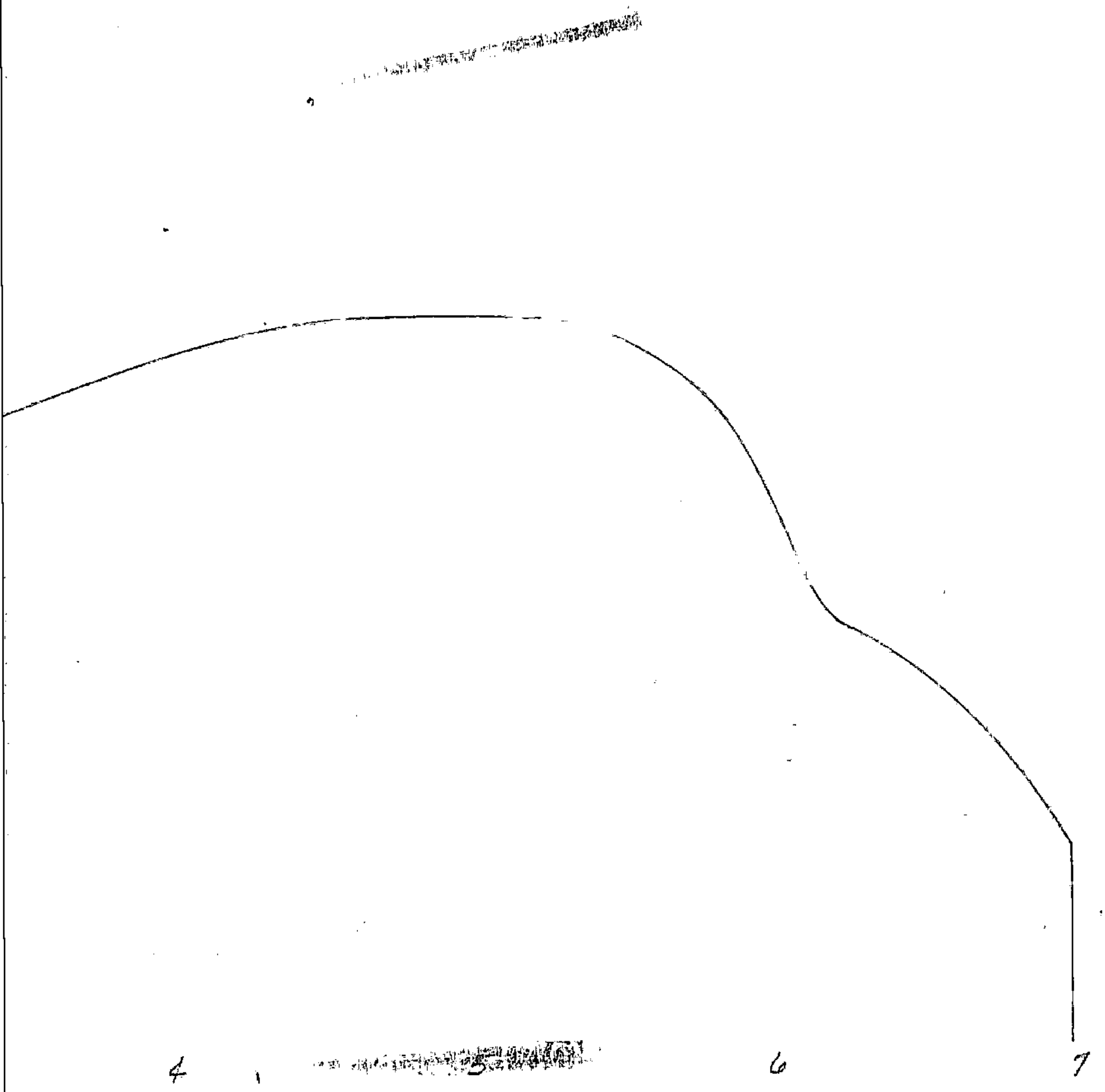
COMPARISON OF CROSS-SEC

TF-102 A AND TF-102A WITH REVISED



TIONAL AREAS

WINDSHIELD AND CANOPY



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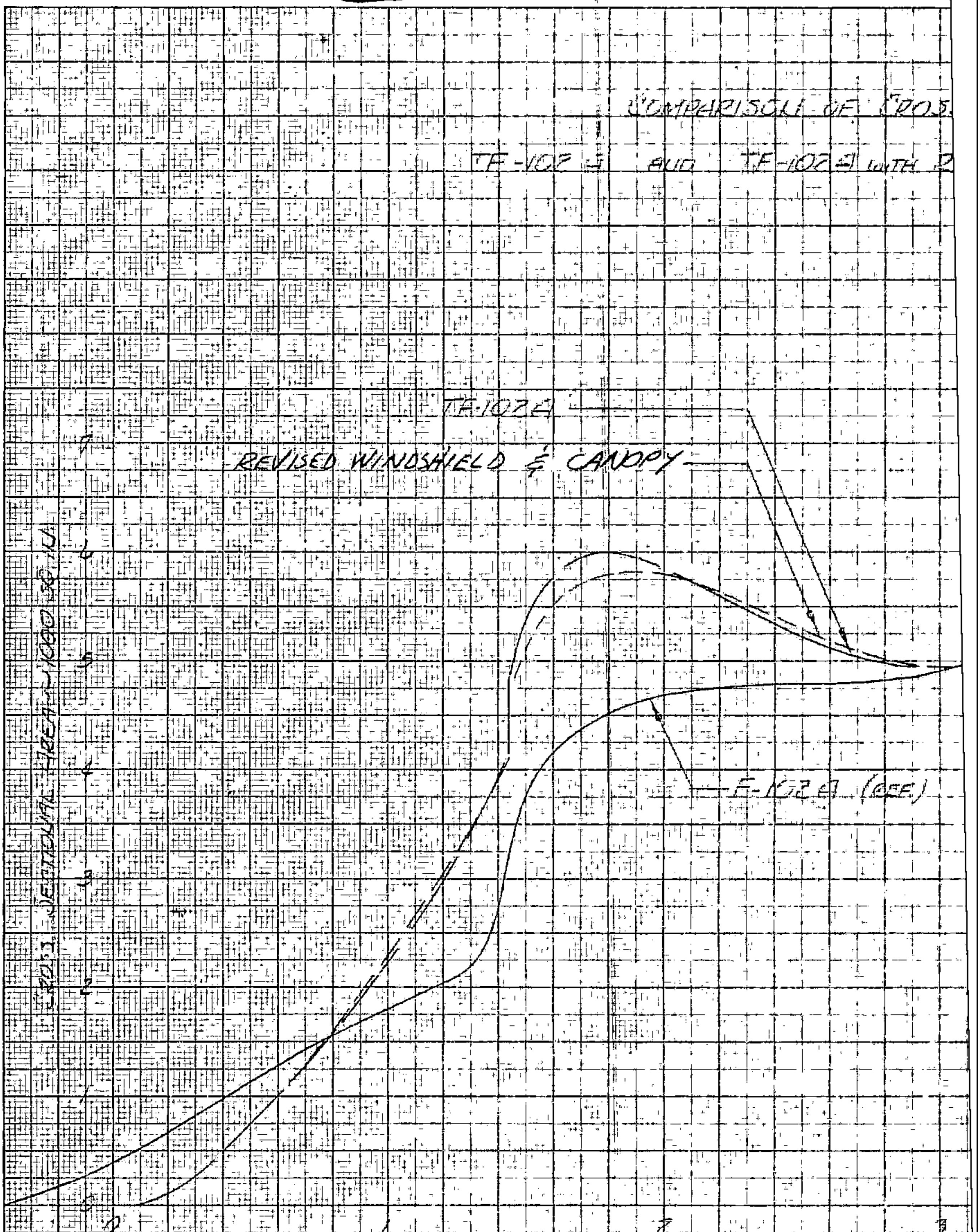

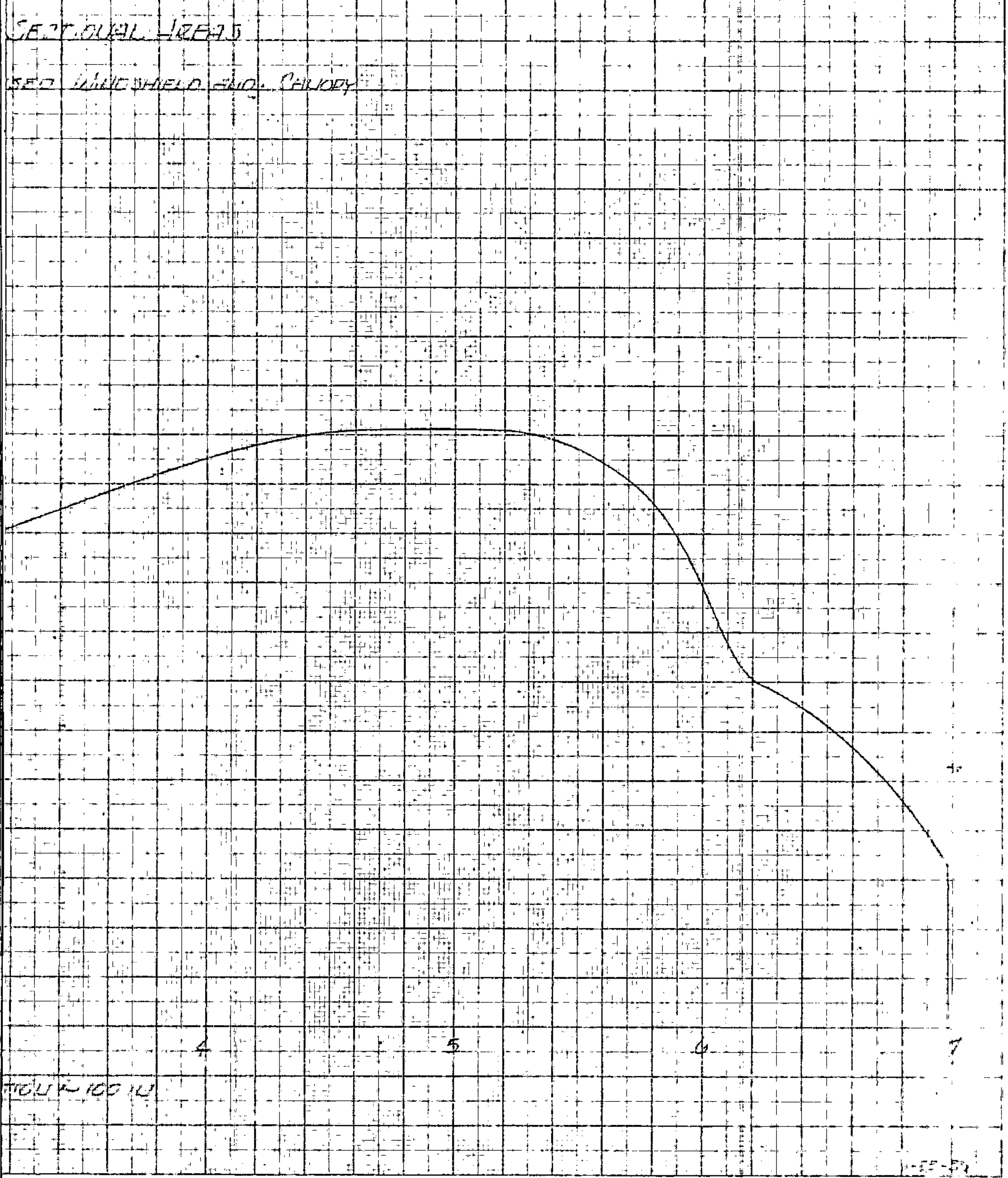

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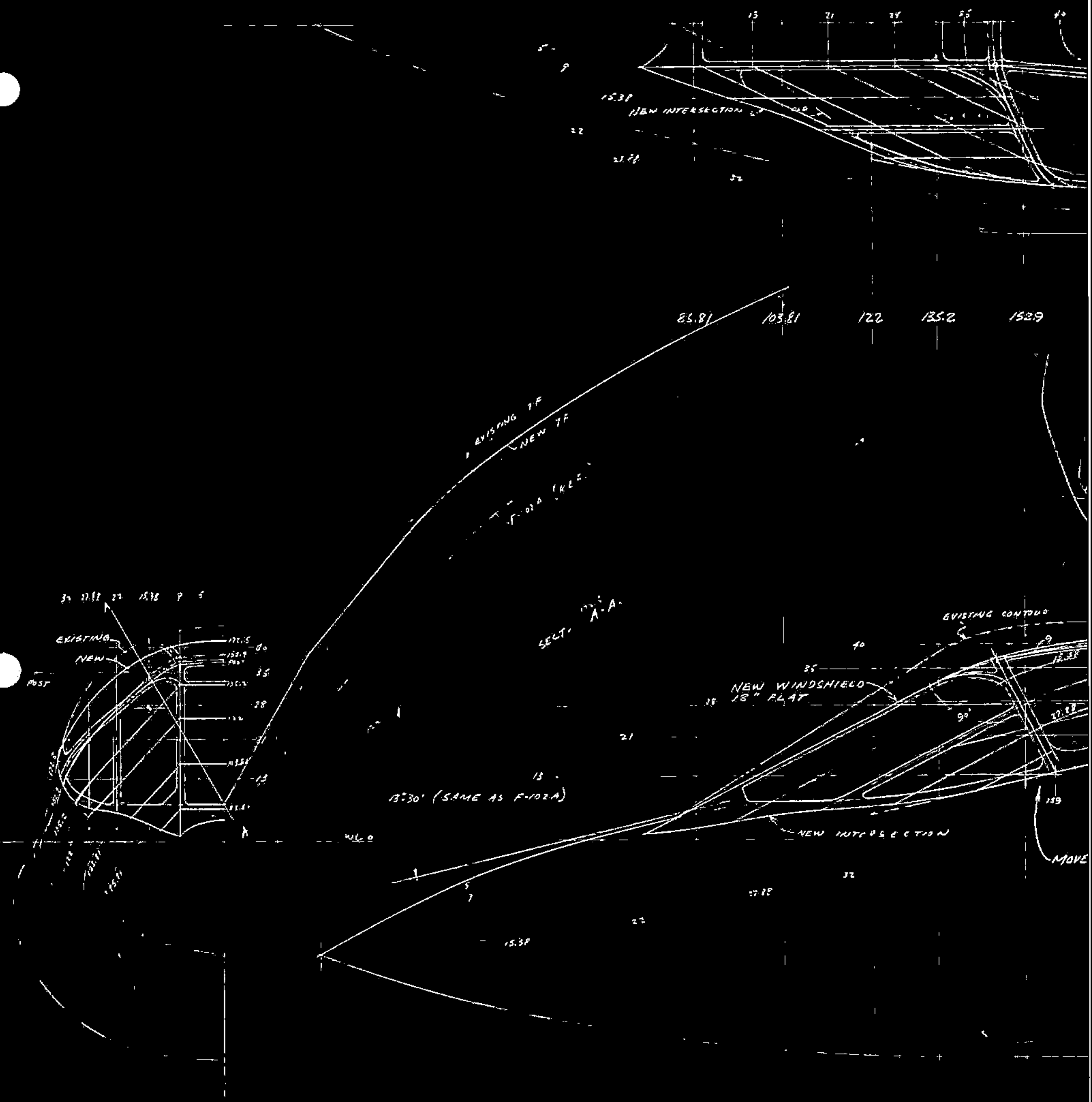
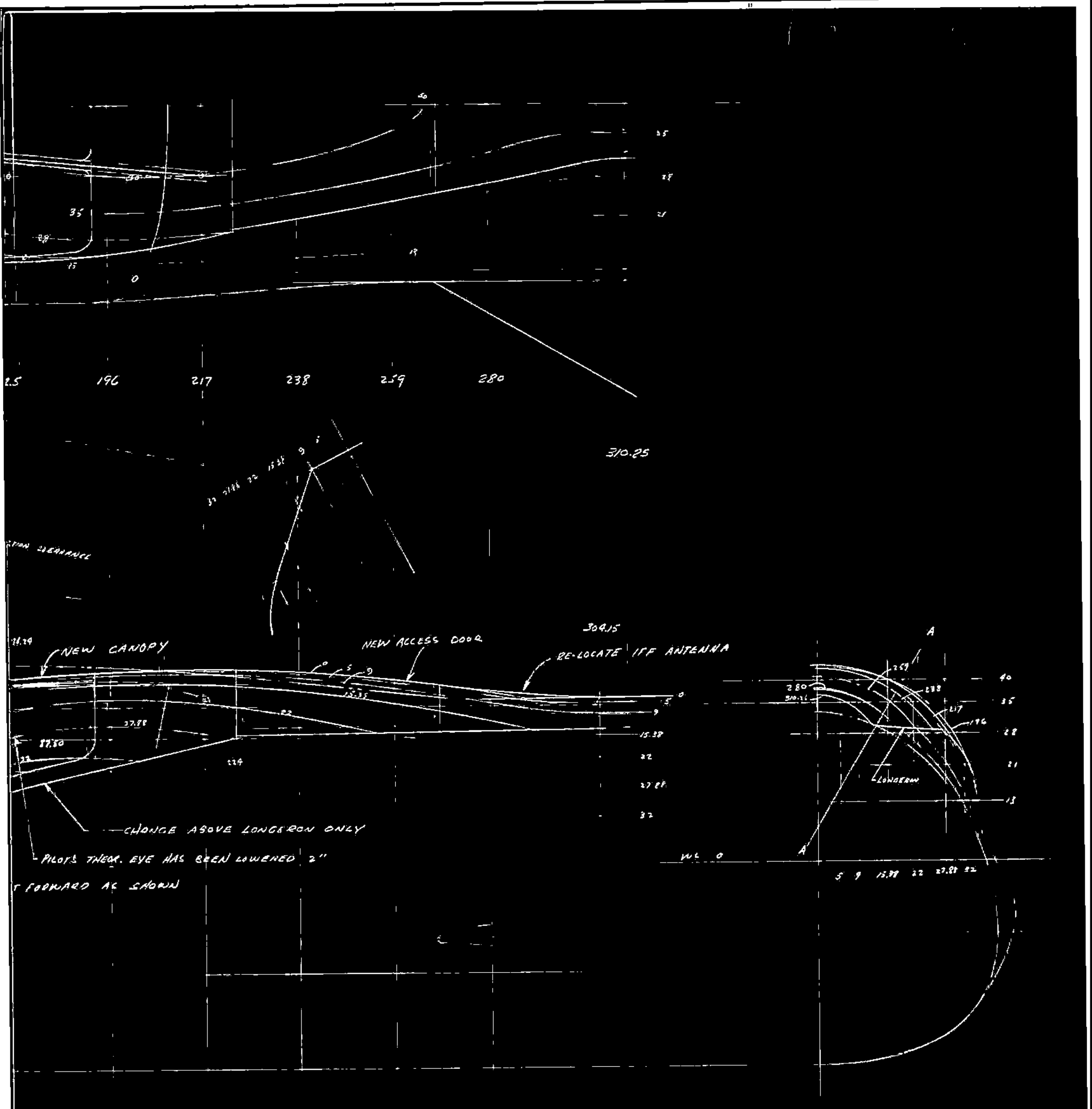

13
21
28
35
40
15.38
NEW INTERSECTION
OLD
22
27.78
32
83.81
103.81
122
135.2
152.9
EXISTING 7F
NEW 7F
F-102A (REF.)
SECT. A-A
32 27.78 22 15.38 9 5
EXISTING
NEW
POST
80
35
28
13
WL 0
13°30' (SAME AS F-102A)
EXISTING CONTOUR
NEW WINDSHIELD 18" FLAT
90°
12.38
27.78
159
NEW INTERSECTION
MOVE



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PRELIMINARY DESIGN DRAWING

- STUDY -

REVISED WINDSHIELD & CA

EXISTING TF-102A AIRPL

LINGLEY | APPROVED | SCALE 1/10

SAN DIEGO, CALIFORNIA
DIVISION OF GENERAL DYNAMICS CORPORATION

SD-

ZP-8-038

CONVAIR - A DIVISION OF GENERAL DYNAMICS CORPORATION
ENGINEERING DEPARTMENT . . SAN DIEGO, CALIFORNIA

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Herewith Copy No. ____ of Convair Secret Report No. ZP 8-038 "A Study of the F-102B as a Fighter Bomber Airplane", dated 29 April 1956.

Please sign this receipt and return to Preliminary Design Section, San Diego Engineering Department.

Convair, A Division of General Dynamics
Corporation.

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FORM 32-B-1

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

[redacted]

REPORT ZP 8-038
DATE Apr. 29, 1956
MODEL FB-102B
Revised May 23, 1956

TITLE

A STUDY OF THE F-102B
AS A
FIGHTER BOMBER AIRPLANE

Classification Changed to:
UNCLASSIFIED
Authorized by: [illegible] 2-1[illegible]-66
D D [illegible]
Reclassified by [illegible] Date
L. G. [illegible] [illegible]-1 4-4-67.

PREPARED BY T. M. Hemphill
R. C. Lingley

GROUP Preliminary Design

REFERENCE

CHECKED BY

APPROVED BY R. R. Lambert

NO. OF PAGES
NO. OF DIAGRAMS

REVISIONS

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WEIGHT SUMMARY................................. 5
DRAWINGS....................................... 6
PERFORMANCE SUMMARY............................ 9
MISSIONS.......................................10

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FB-102B FIGHTER BOMBER AIRPLANE

INTRODUCTION

This report presents a brief exploratory study of a fighter bomber version of the U.S.A.F. Interceptor, now in quantity production. The purpose of the report is threefold. First, it is to invite the attention of Air Force personnel to the performance of the F-102 equipped as a fighter bomber. Secondly, it is to provide performance estimates from which Air Force personnel can make analyses, comparing the F-102 with other airplanes under reasonably comparable circumstances. Finally, it is to provide a basis for discussions to develop the most suitable arrangement of the optimum armament and equipment.

The present F-102 program includes a plan of continuing development. F-102A airplanes are now in production. The first F-102B airplane will fly late this year and production will follow. Already an advanced type F-102 interceptor is under pre-design study.

The F-102B type to be flown in December of this year, will have more effective electronic fire control equipment, and a more powerful engine installed in it than that in the F-102A, and it will carry improved armament. Its performance as an airplane and as a weapon will be greatly improved.

The layout studies and performance estimates contained herein are based on the F-102B version of the airplane. The structural weight has been increased to provide for an 8.67 "g" maneuvering load factor at combat weight, prior to release of bombs.

This report is in two sections. The first consists of layout drawings of the airplane as a fighter bomber. The second shows its weight and performance characteristics.

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DRAWINGS

GENERAL ARRANGEMENT

The F-102 interceptor is an all weather supersonic fighter designed to destroy enemy bombers. Its conversion to a fighter bomber is relatively easy due to the favorable location of fire control equipment and armament. Drawing SD-56-08015 shows the general arrangement and exterior dimensions. They are the same as those of the interceptor.

A number of important characteristics of the F-102 are attributable to its general arrangement.

The delta wing shape does not stall in the normal sense, i.e., its lift curve does not drop abruptly with increasing angle of attack. Above 30° angle of attack, it gradually decreases. However, the attainable angles are arbitrarily made less than about 20°. This means that a pilot is able to make use of all the available lift coefficient without fear of a sudden loss of lift. Also, it is possible for him to maneuver at high altitudes without the sudden losses of considerable altitude.

Because of the above, the airplane will be very difficult to spin if it is spinable at all.

The tailless delta wing design does not utilize flaps and slots, hence its wing area is made greater than that of the wing on a conventionally designed airplane. The delta's greater wing area results in greater wing volume. This is turn permits carrying a large amount of fuel. Dragwise, the increased wing area is offset by the absence of a horizontal tail.

The absence of flaps, slots and a horizontal tail tends to simplify the pilot's job, simplifies maintenance, improves anti-icing and tends to make the whole airplane more rugged.

The delta wing shape, itself, is inherently rugged, closely approaching the ideal cantilever beam.

Armament, (rockets, missiles and bombs) released from the F-102 have followed smooth, stable projectories. It is believed that the vertically disposed armament bay doors at the mid fuselage location and the pressure field under the large chord wing, combine to produce a region of relatively

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Revised May 23, 1956

undisturbed [redacted] Apparently, this permits the released armament [redacted] enough to penetrate any region of disturbed flow which may exist without appreciable deviation from its intended course.

INBOARD PROFILE

The inboard profile drawing No. SD-56-08016 shows how the space available has been allocated to fighter bomber equipment. There is ample room for the radar, fire control, bombing, navigation, communications and other electronics equipment in the spaces ahead of, and immediately aft of the cockpit. These spaces are used for similar installations in the interceptor.

The rear armament compartment can house the new armament, internally stored. Rockets can be mounted in the armament doors as they are on the interceptor. The forward armament compartment is available for additional internally stored fuel.

The equipment selected for communications, short range navigation, all weather take-off and landing, identification and tail warning are standard items. An inertial system has been selected for long range navigation.

The bombing system utilizes a Convair SARTACK type high definition mapping and ranging radar and a General Electric bombing computer, similar to the E-30 system, together with a General Electric optical sight head and an infra-red tracker. The virtual image from the infra-red tracker appears on the optical sight combining glass.

The above equipment will provide a versatile all weather bombing system with loft, toss, over the shoulder, level and dive bombing capabilities under all conditions of visibility. Conventional bombs or special weapons can be released at all speeds, utilizing either the optical sight with radar ranging, or complete radar operation. Rockets or I.R. missiles may be fired at all speeds, utilizing either a visual or infra-red indication on the optical sight with radar ranging.

ALTERNATE INSTALLATIONS

A drawing of alternate possibilities, SD-56-08017, shows some of the fuselage space utilized for different purposes than previously indicated. In the sectional views at the top of the page, conventional bombs or missiles have been loaded into the armament bay and bombs loaded at the wing tank position.

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As a [redacted] could be carried internally by making provisions for them, in the forward armament bay, instead of fuel.

A trainer version of the interceptor is being designed by reshaping the fuselage, above the upper longeron, and installing a second cockpit. Thus the design for a two place version of the Fighter Bomber F-102 will be available. The second cockpit is shown in the side view.

To make a tanker for "buddy" missions, it is visualized that additional fuel and a drogue dispensing rig would be carried in the aft armament compartment. This is not shown on the drawing.

[redacted]

WEIGHT SUMMARY

	Lbs.	Lbs.	Lbs.
Wing		3,752.9	
Tail		687.0	
Fuselage		5,523.5	
Landing Gear		1,404.2	
Main	1,182.7		
Nose	221.5		
Surface Controls		480.9	
Nacelle Group		37.0	
Propulsion Group		8,197.3	
Engine	5,925.0		
Air Induction System	911.9		
Exhaust System	262.0		
Cooling System	45.8		
Lubricating System	50.7		
Engine Controls	28.1		
Starting System	70.6		
Fuel System	903.2		
Instruments		123.4	
Hydraulic & Pneumatic Systems		397.5	
Electrical		546.0	
Electronics		1,644.4	
Armament		405.0	
Furnishings		269.0	
Air Conditioning & Anti-Ice		320.3	
Auxiliary Gear		57.6	
WEIGHT EMPTY			23,846.0
Pilot (with gear)		270.0	
Unusable Fuel		234.0	
Oil-Engine (Including Trapped)		60.0	
Fuel		19,621.0	
Armament (Including 374 lbs. rockets)		2,174.0	
External Tanks & Pylons		550.0	
Cameras		45.0	
USEFUL LOAD			22,954.0
GROSS WEIGHT			46,800.0

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PERFORMANCE SUMMARY

P & W J75-JT4A-24 TURBOJET

F-102 FIGHTER BOMBER AIRPLANE
BASIC ARRANGEMENT PER DRAWING SD-56-08016

Weights

Design Take-Off	46,800 lbs.
Maximum Fuel	19,805 lbs.
Operating Weight Empty*	26,629 lbs.
Combat**	35,100 lbs.
Landing***	26,500 lbs.

Speeds (At Combat Weight of 35,100 lbs.)

Maximum at 35,000 ft. with Max. Power	M = 2.00
Maximum at 35,000 ft. with Mil. Power	M = 1.00
Cruise Speed, Optimum (w/wout external fuel tanks)	531 knots

Climb (At Combat Weight of 35,100 lbs.)

Maximum Rate of Climb at Sea Level	34,200 fpm
Combat Ceiling (M = .94)	51,200 ft.
Take-Off Ground Roll at Design Take-Off Weight	3,500 ft.

Landing Ground Roll at Landing Weight

With drogue chute	2,410 ft.
No drogue chute	3,250 ft.

Engine Ratings (P & W Spec #2604)

Normal Rated Power	13,500 lbs.
Military Power	15,500 lbs.
Military plus Afterburner Power	23,500 lbs.

* Design T.O. weight less all fuel and external tanks.
** Weight over target - Mission No. 3
*** Operating weight empty plus reserves.

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MISSIONS

In the missions shown on the following pages, the fuel allowances and reserves are as follows:

(1) The allowance for warm-up, take-off and acceleration to climb speed is the amount of fuel burned by the engine at sea level in 5 minutes of normal rated power operation, plus one minute of full afterburner operation. This is the standard take-off fuel allowance.

(2) Fuel held as a landing reserve is the amount of fuel burned by the engine in 15 minutes of operation at the speed for maximum endurance at cruising altitude, plus fuel for one missed approach and an instrument "go-around" plus 5% of the initial fuel.

(3) During "buddy" missions, the fuel held in the bomber for use at the transfer point is 2-1/2 percent of the initial fuel for both the bomber and the buddy airplane, plus fuel for 20 minutes of loiter at cruising altitude. The tanker airplane has a like reserve. The tanker unit carries sufficient fuel so that each airplane has a landing reserve of the amount of fuel needed for 15 minutes of operation at the speed of maximum endurance at cruising altitude, plus one missed approach and an instrument go-around, plus 2-1/2 percent of the initial fuel for both airplanes.

(4) The customary 5% has been added to the engine manufacturer's SFC's in the computations of all the missions.

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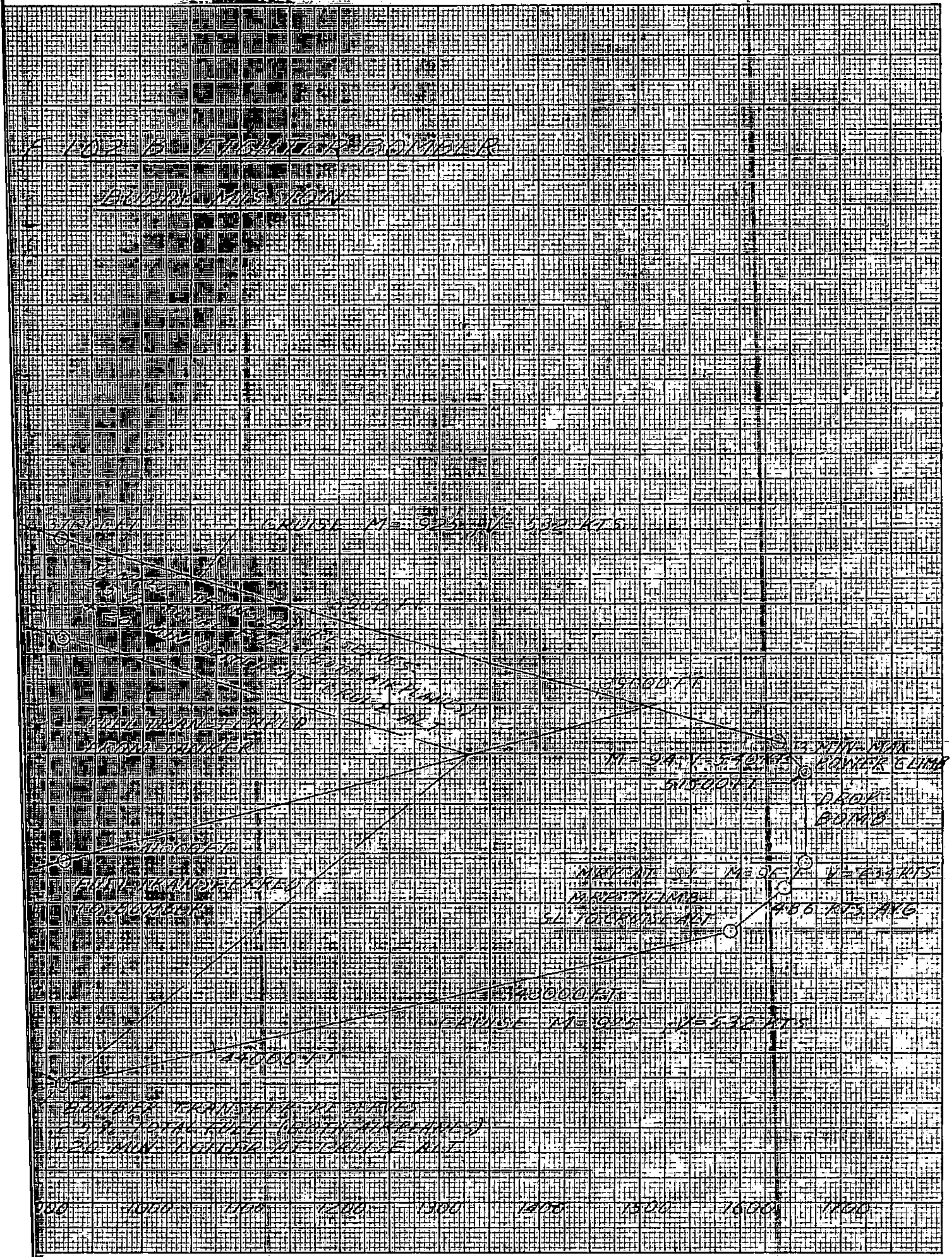
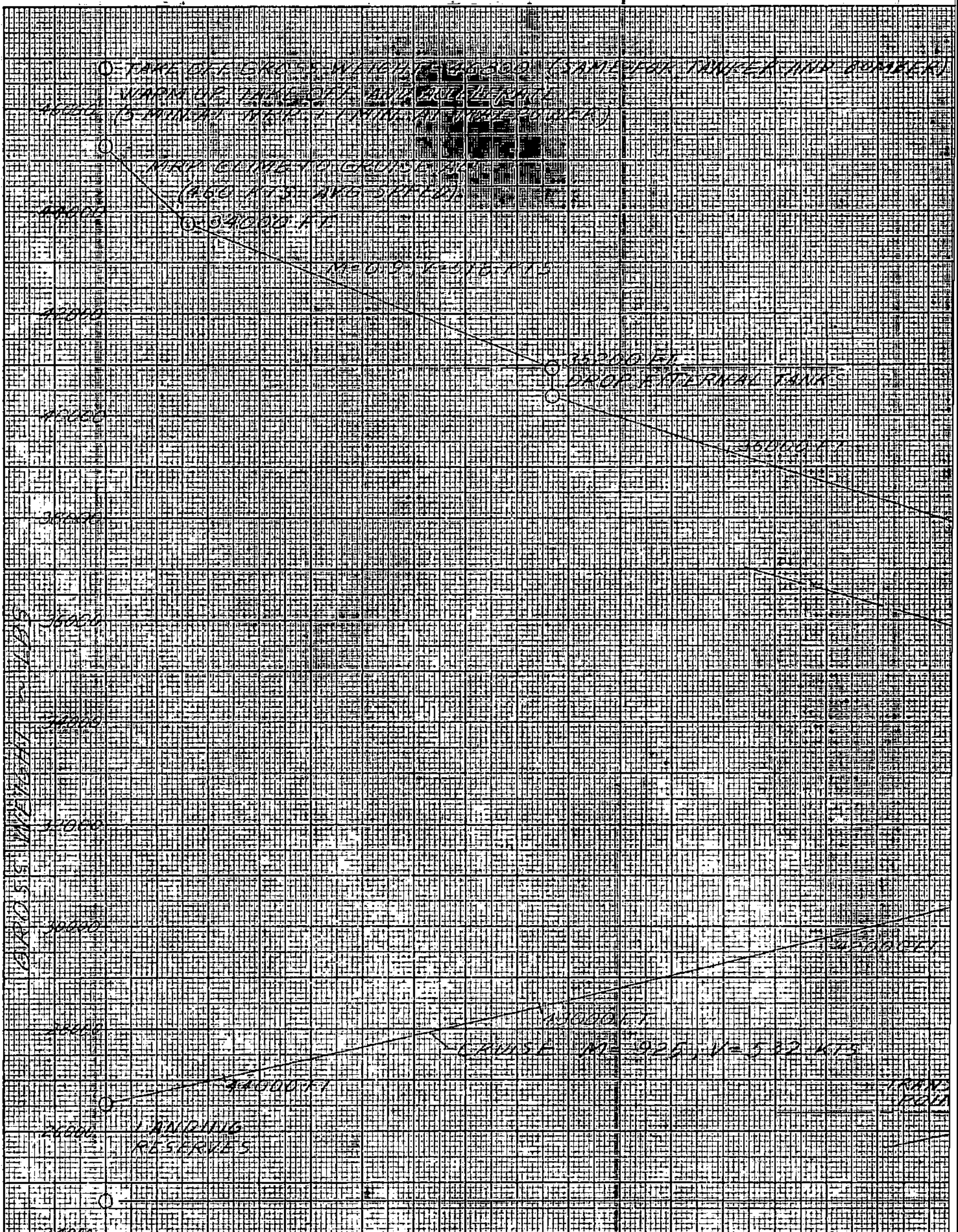
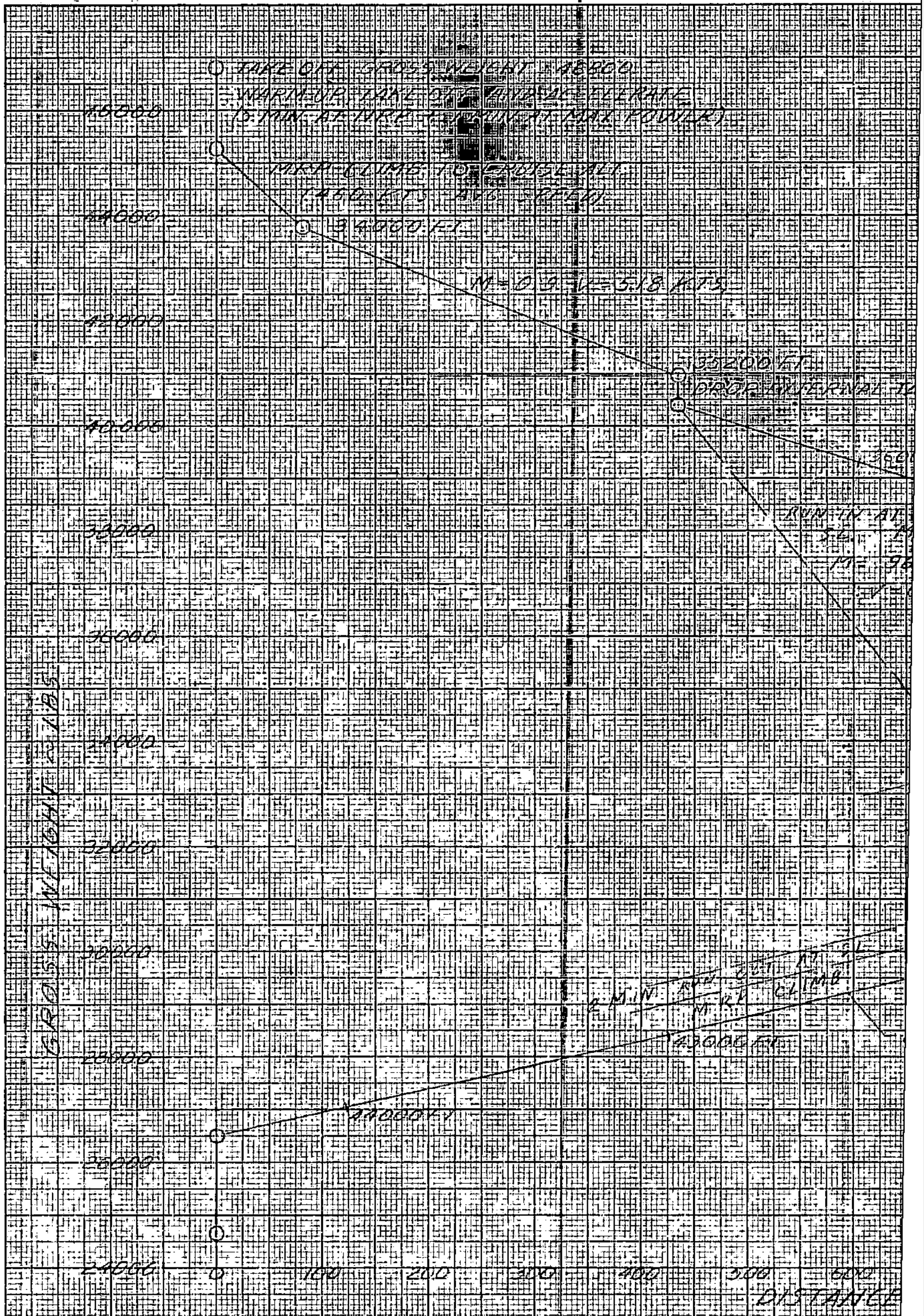
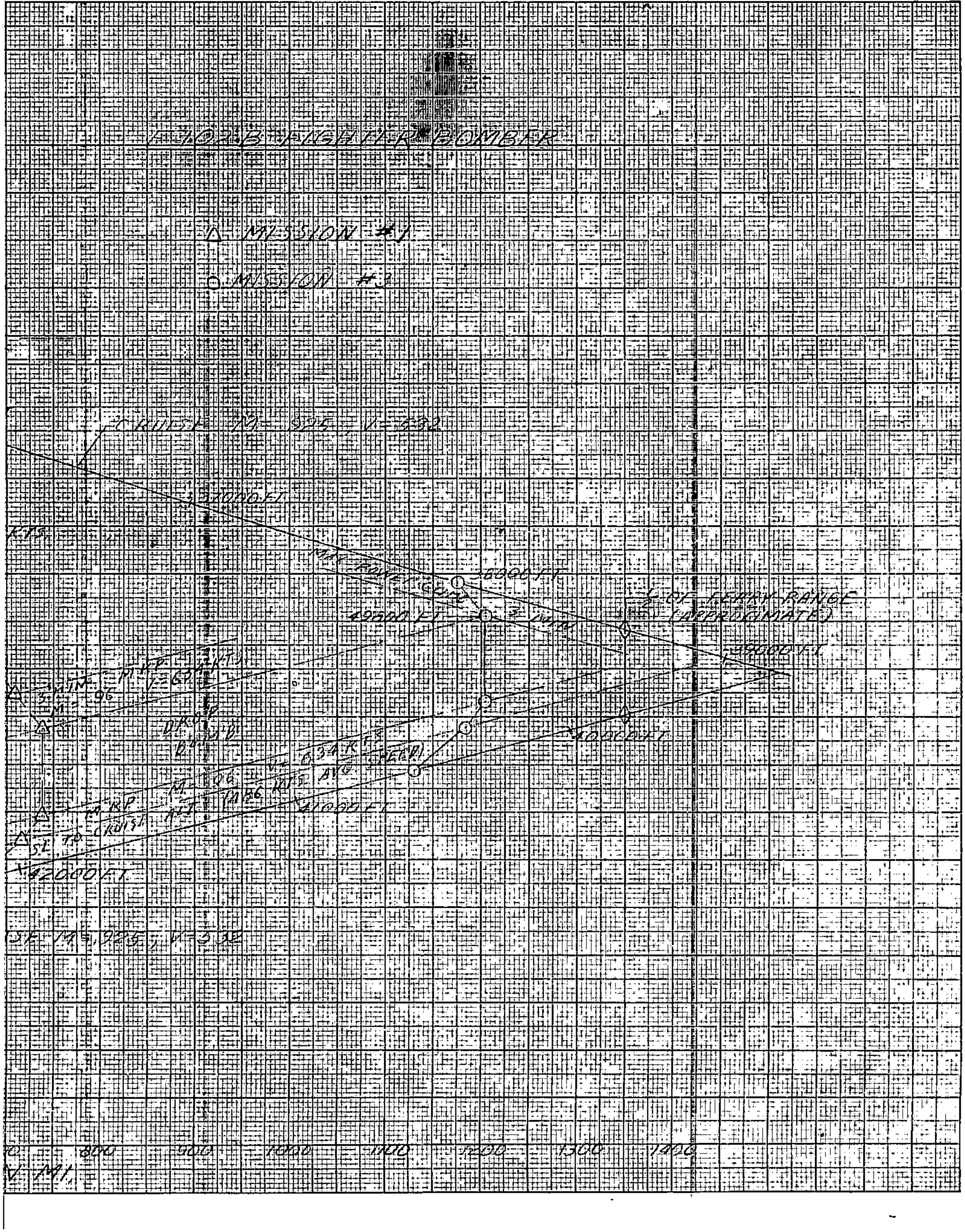

F-102-B FIGHTER BOMBER
BUDDY MISSION
37000 FT
CRUISE M=.925 V=532 KTS
49000 FT
FUEL TRANSFERRED FROM TANKER
3 MIN MAX POWER CLIMB
M=.94 V=540 KTS
51500 FT
DROP BOMB
FUEL TRANSFERRED TO BOMBER
MRT AT SL M=.96 V=634 KTS
MRT CLIMB SL TO CRUISE ALT
486 KTS AVG
43000 FT
CRUISE M=.925 V=532 KTS
44000 FT
BOMBER TRANSFER PT PLANE
2.5% TOTAL FUEL (BUDDY AIRPLANES)
20 MIN LOITER AT CRUISE ALT
900
1000
1100
1200
1300
1400
1500
1600
1700

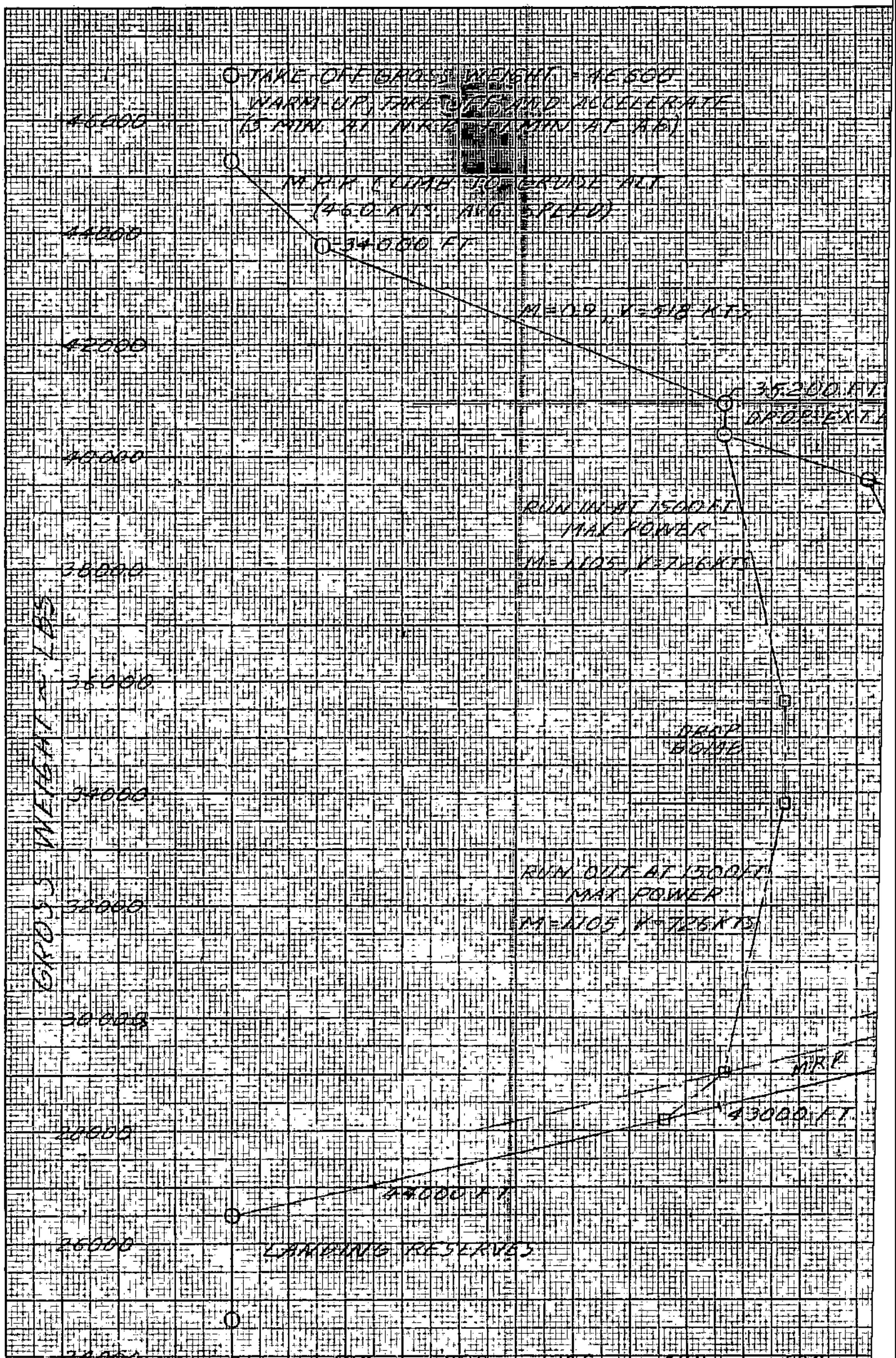
K+E 10 X 10 TO THE CM. 359T-14LG
KEUFFEL & ESSER CO.
ALBANENE ®
MADE IN U.S.A.



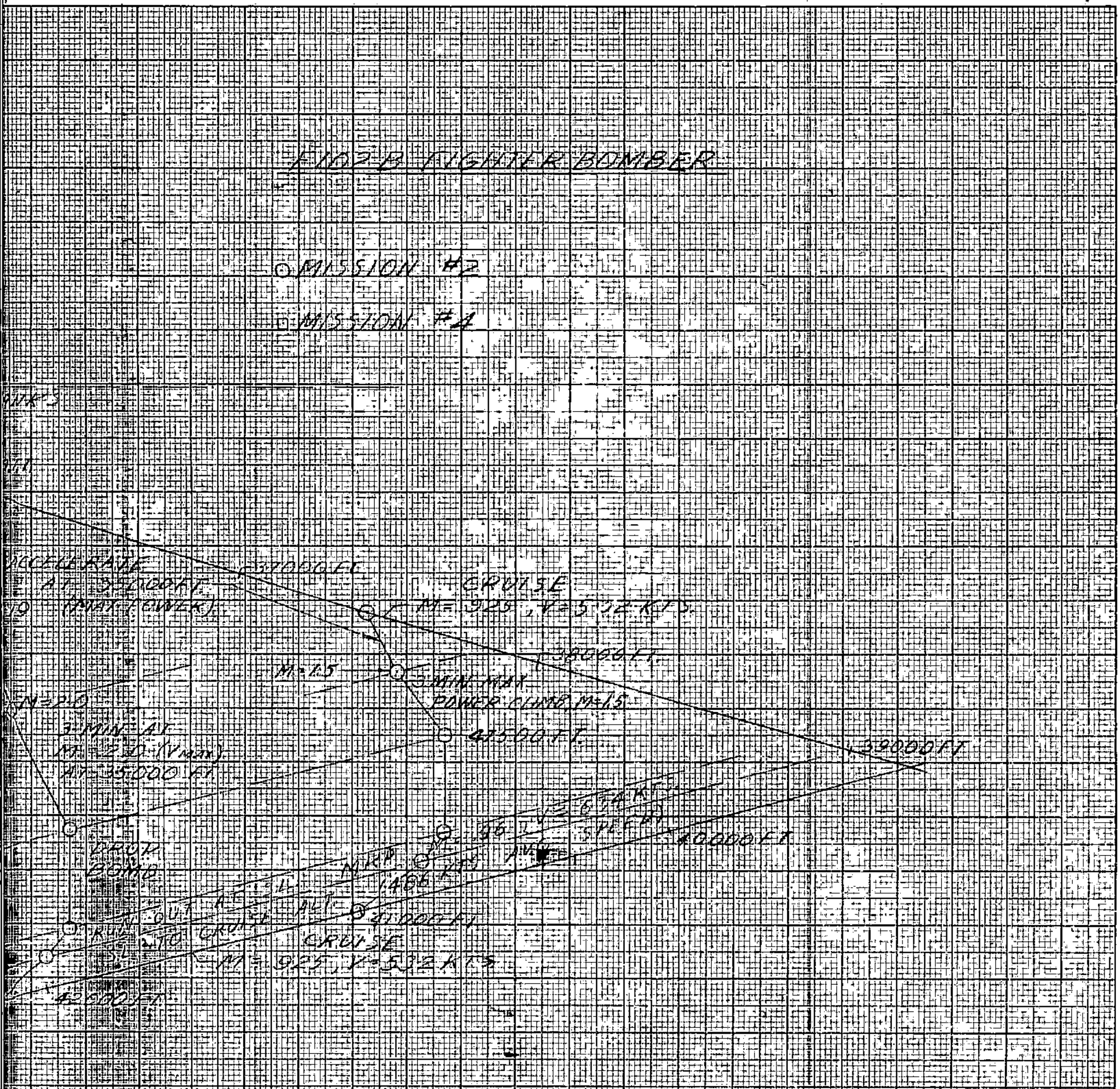
K+E 10 X 10 TO THE CM. 359T-14LG
KEUFFEL & ESSER CO.
ALBANENE ® MADE IN U.S.A.




F-102B FIGHTER BOMBER
△ MISSION #1
○ MISSION #3
CRUISE M = .925, V = 532
37000 FT.
KTS.
MAX POWER CLIMB
36000 FT.
49000 FT.
3 MIN.
½ OF FERRY RANGE (APPROXIMATE)
49000 FT.
5 MIN
M = .96
MRP
V = 634 KTS
DROP BOMB
V = 634 KTS
M = .96
AVG SPEED
1486 KTS
NRT
MRP
SL TO CRUISE
41000 FT.
40000 FT.
42000 FT.
OF M = .925; V = 532
0
800
900
1000
1100
1200
1300
1400
V MI.



K+E 10 X 10 TO THE CM. 359T-14LG
KEUFFEL & ESSER CO. MADE IN U.S.A.
ALBANENE ®



F102-B FIGHTER BOMBER